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EARTH-TO-GEOSTATIONARY ORBIT TRANSPORTATION FOR SPACE SOLAR POWER SYSTEM DEPLOYMENT

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ABSTRACT

Space solar power satellites have the potential to provide abundant quantities of electricity for use on Earth. One concept, the Sun Tower, can be assembled in geostationary orbit from pieces transferred from Earth. The cost of transportation from Earth is one of the major hurdles to space solar power. This study found that a two-stage rocket launch vehicle with autonomous solar-electric transfer can provide the transportation at prices close to the goal of $800/kg.

INTRODUCTION

The goal of this study was to examine the transportation of space solar power (SSP) elements from the Earth to the operational orbit, geostationary Earth orbit (GEO). The effort of this study continued and built on work on SSP transportation at Boeing performed in 1998[1] and 1999.[2] One of the findings in 1998 was that a rocket two-stage-to-orbit (TSTO) reusable launch vehicle (RLV) could deliver payloads to low Earth orbit (LEO) for a recurring cost of about $370/kg with a highly advanced rocket engine. The 1999 work used that information to help evaluate in-space transportation options.

In the 1999 work,[2] a reference transportation system was first developed and analyzed. The reference concept used autonomous solar electric propulsion from LEO to GEO, as illustrated in Fig. 1. The SSP element during transfer is shown in Fig. 2. The reference design used Hall thrusters with direct drive and delivered a specific impulse of 2000 s.

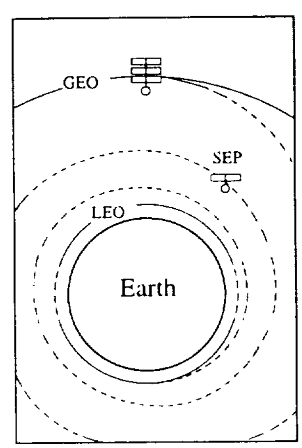

Figure 1. Illustration of autonomous solar-electric transfer from LEO to GEO.[2]

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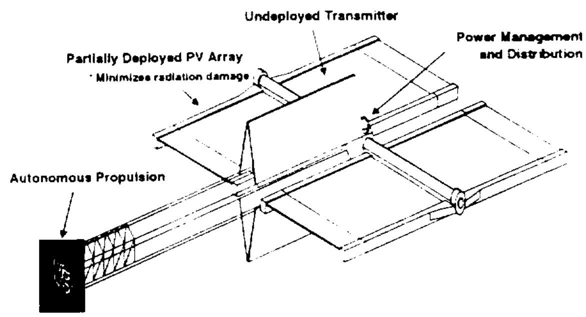

Figure 2. Sketch of autonomous transfer.[2]

In addition to the reference concept, several alternate in-space transportation options were evaluated.[2] Figure 3 shows the mass results. The options were all sized to deliver about 17,000 kg of useful SSP element mass from a 300-km circular equatorial orbit to GEO. Consideration of launch from sites other than equatorial led to the conclusion that equatorial launch was preferred. As might be expected, the chemical and solar thermal options resulted in large propellant mass. Figure 4 shows the recurring cost per flight for the same options. The large mass of the chemical and solar thermal options increased the launch costs. An improved solar-electric option with a specific impulse of 4000 s decreased the mass but increased the cost. This solar-electric option, with a specific impulse of 4000 s, requires about 44 percent more electric power over the solar-electric option with a specific impulse of 2000 s to meet the 90-day transfer time. The result is higher power system and thruster recurring cost and subsequent transportation cost. A hybrid option with a reusable chemical stage with an aerobrake for part of the transfer and autonomous completion of the transfer did not reduce costs, but further optimization of hybrid concepts could lead to cost savings, possibly using solar thermal propulsion with aerobraking. The tether option used a very conservative tether approach and should probably be considered further with other tether designs.

One conclusion of the 1999 work was that the transportation system using a TSTO RLV and autonomous transfer was a reasonable choice for SSP. Further details of the effort are provided in Ref. 2.

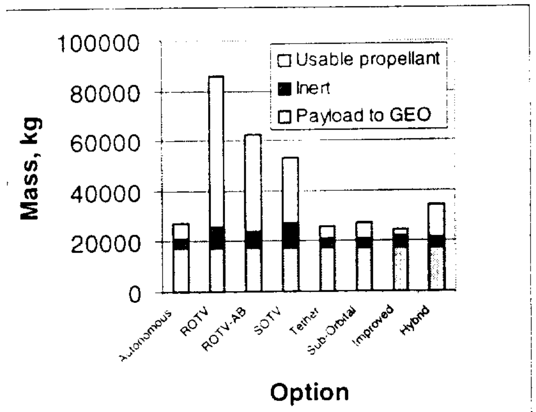

Figure 3. Comparison of mass in LEO of several in-space transportation options.[2]

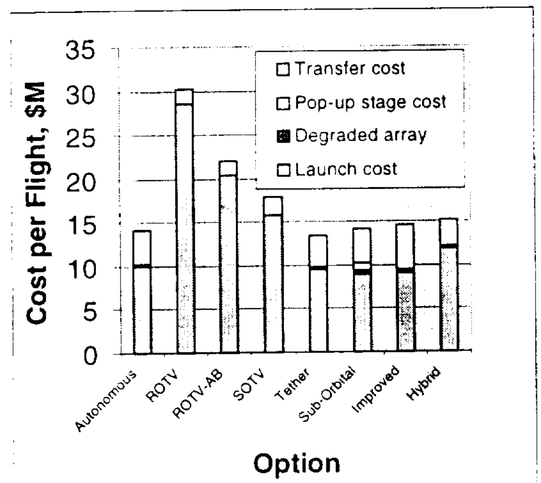

Figure 5. Comparison of recurring cost per flight of several in-space transportation options.[2]

POINT-OF-DEPARTURE VEHICLE

One of the important aspects of the current effort is selection of the launch vehicle concept as the point of departure. The first question to consider is reusability, and the conclusion is that the vehicle should be mostly reusable. While expendable or partly reusable vehicles are competitive for near term applications with a small number of launches, at the launch rate for SSP, the cost of expendable hardware is prohibitive. The 1998 work[1] showed that the reusable concepts provided lower recurring costs than the concept with an expendable hydrogen tank.

The next question that can be addressed is whether the Earth-to-orbit vehicle should go to

LEO. For launches that do not reach LEO, recovery of the orbiter is difficult. Downrange landing would require weather clearance at the landing site and return transportation. For launches beyond LEO, the size and mass of the orbiter increase, and the thermal protection to recover the orbiter would be more difficult. The conclusion is that LEO is a natural staging condition. One exception to this conclusion is a concept with a tether and transfer of the payload from the launch vehicle to the tether at a velocity low enough to permit the launch vehicle to fly back to the launch site. While such a concept is promising, examining it is not within the scope of this study.

Another question is whether the launch vehicle should include ascent airbreathing propulsion. While there are different opinions on this question, most detailed studies of both rocket and airbreathing concepts have concluded that airbreathing does have an economic benefit and would be justified only for unique missions requiring such capabilities as offset launch. Another factor to consider is that design of an airbreathing vehicle takes significant detailed design and aerodynamics work to have a believable result, and the current effort is too limited in scope to complete such work. The conclusion is that the vehicle for the current study should be rocket.

Another question of interest is whether the vehicle system should be a single-stage-to-orbit (SSTO) or two-stage-to-orbit (TSTO) design. The 1998 study[1] indicated a small economic advantage for SSTO at SSP launch rates. That work was done with very optimistic rocket engine data. Improvements in the TSTO concept have happened since that time. There is also a problem with center of gravity of vehicles as the payload increases. This problem is related to the square-cube effect: As vehicle size increases, the volume, mass, and thrust are related to the cube of the length while the areas are related to the square of the length. As a result, the engine mass increase forces the center of gravity aft. This problem limits the payload of SSTO vehicles to roughly 30,000 kg and TSTO vehicle to roughly 45,000 kg. Exact location of the limit would require more detailed design and aerodynamics analysis than can be accomplished within the scope of the current effort. The conclusion is that the TSTO should be selected for the current study.

A final consideration is the design of the rocket TSTO vehicle system. Work has been completed at Boeing on a rocket TSTO for second generation reusable launch vehicles (RLVs). It is possible that the high launch rates or the larger payload needs of SSP launch would drive the selection to another concept, but the differences, for the purposes of this study, would be small. Also, selection of the Boeing TSTO concept allows maximum use of the information developed in the detailed TSTO studies. The conclusion is that the Boeing TSTO concept should be the point-of-departure design for this SSP task. There is a difference between the technology level assumed in the Boeing SSP End-to-End Architecture Study[1] and that of the recent work; the Boeing TSTO concept has been developed with technology that is expected to be available within the decade, whereas the rocket engines in Ref. 1 assumed the existence of advanced materials.

LAUNCH VEHICLE ANALYSIS

Analysis of the Boeing TSTO concept was initiated during the 1999 effort.[2] Modification of the design to account for launch from an equatorial site and launch from Kennedy Space Center into due east and 51.6 deg. orbits was analyzed using trajectory optimization and sizing codes. The LEO payload for that work was 27,216 kg. For the current effort, the effect of increasing the payload has been examined. The effect of doubling the payload mass, while holding the payload density and ratio of length to diameter constant is shown in Fig. 5. The results are shown as a ratio to the values of the vehicle with payload of about 27,000 kg. The orbiter dry mass and the gross liftoff mass of the vehicle system increase in a nearly linear manner, but the lines would have a positive intercept if they were extended to no payload. The positive intercept is indicative of the fact that some mass elements are relatively fixed and would remain as the vehicle size is reduced.

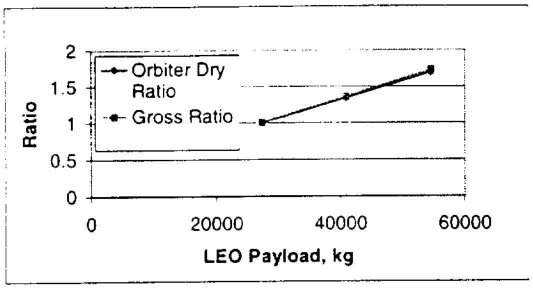

Figure 5. Effect of LEO payload mass on orbiter dry mass and system gross mass.

Another important effect that occurs as the payload is increased is the aft movement of the center of gravity of the orbiter. The center of gravity at landing with full payload is shown in Fig. 6. Vehicles of this type generally have an aft center of gravity that leads to the requirement for electronic stability augmentation. While the Boeing TSTO work has studied the problems of the vehicle with a payload of about 27,000 kg. stability work has not been accomplished for the more aft center of gravity locations. At some point, modifications to the vehicle to allow flight with the aft center of gravity would become so extreme that the design would not be competitive. Finding that limit is beyond the scope of this study. but estimates are that payloads of about 40,000 kg are probably acceptable.

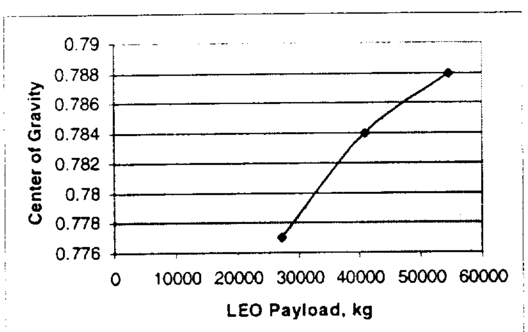

Figure 6. Effect of LEO payload mass on orbiter center of gravity at landing with full payload.

The LEO payload capability of the launch vehicle has an effect on the number of flights required to launch one SSP satellite. This analysis is based on the requirement to launch one satellite each year for 30 years. The useful mass of each satellite at GEO is 20.23 Gg. The resulting number of flights required each year, using the reference autonomous transfer, is shown in Fig. 7.

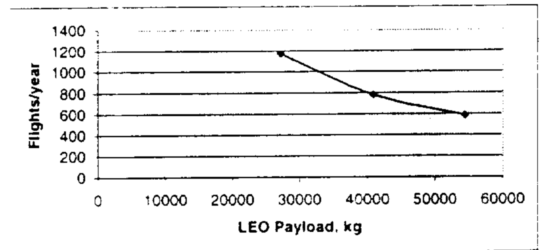

Figure 7. Effect of LEO payload mass on number of flights required/year for one satellite/year.

LAUNCH VEHICLE COSTS

The costs of the launch vehicles were calculated and are shown in Figs. 8 and 9. Again, the results are shown as a ratio to the results for the vehicle with a payload of about 27,000 kg. but in this case the life cycle cost (LCC) is the value used in forming the ratio for all costs. The results show that the life cycle costs have a minimum in the range of 30,000 to 40,000 kg payload, with a small increase of less than 3% for the largest payload considered. The LCC is dominated by operations costs because of the large number of flights. The initial costs (Fig. 9), development and production of the initial fleet, favor the smaller payloads but have minimal effect on the LCC.

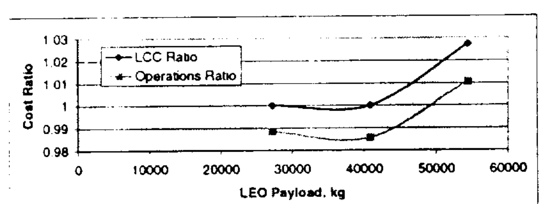

Figure 8. Effect of LEO payload mass on life cycle cost (LCC) and operations cost of launch system, shown as the ratio to the LCC of the vehicle with payload of about 27,000 kg.

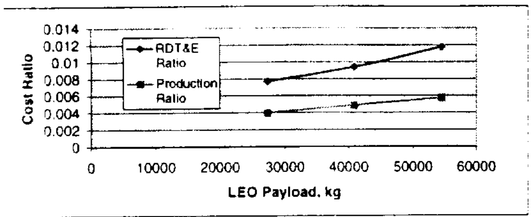

Figure 9. Effect of LEO payload mass on initial costs of launch system, shown as the ratio to the LCC of the vehicle with payload of about 27,000 kg.

Figures 10 and 11 show how the LCC changes if the flight rate changes for the vehicle with a payload of about 41,000 kg. The LCC increases nearly linearly with flight rate. The LCC per kg of payload shows some increase as flight rate drops because the initial costs are fixed.

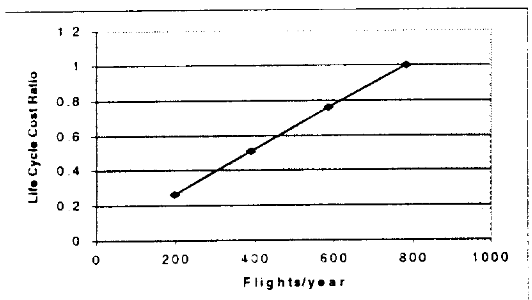

Figure 10. Effect of flight rate on LCC of launch system with payload of about 41,000 kg, shown as the ratio to the LCC of the vehicle with 782 flights/year.

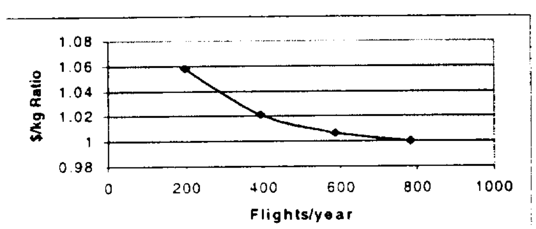

Figure 11. Effect of flight rate on LCC/kg of payload for launch system with payload of about 41,000 kg, shown as the ratio to the LCC/kg of the vehicle with 782 flights/year.

In considering the economics of SSP, the question that must be answered is whether a commercial company would be willing to invest in the project. In order to consider the transportation options in that light, the important question is what price the launch company would charge the SSP company for launches. The answer must be calculated using a business case analysis, with the launch company requiring a satisfactory internal rate of return (IRR) on the investment in the launch vehicle at the launch rate required by SSP. Results of calculations with various values of IRR are shown in Fig. 12. On the scale shown, the 3% difference in LCC with payload appears quite small. As the IRR increases to 40%, the effect of payload size becomes more important. This trend is the result of the reduction in number of flights and increase in initial costs with increased payload size. The preferred payload is on the smaller end when IRR is included in the analysis. This analysis does not reflect any extra costs required for assembly in space that might occur with smaller payloads. Even with the high number of flights in this analysis, including IRR shows that the initial costs are important, and commercial companies are likely to require IRR values approaching 40% for investments with the level of risk involved in SSP.

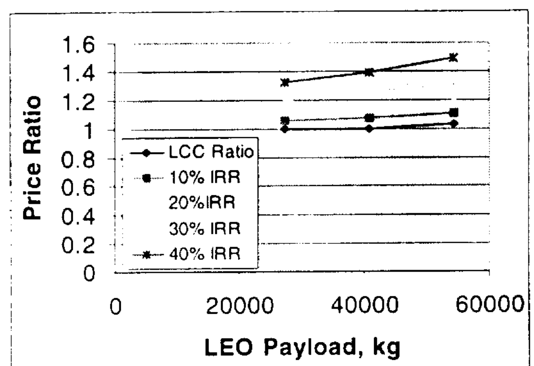

Figure 12. Effect of LEO payload mass on launch price, shown as a ratio to the LCC of the vehicle with payload of about 27,000 kg.

ORBIT TRANSFER ANALYSIS

The transfer from LEO to GEO was analyzed in the current effort assuming the autonomous solar-electric option, found to be a reasonable choice in Ref. 2. The transfer scheme is illustrated in Figs. 1 and 2, and the assumptions used for this analysis are presented in Table 1. Direct-drive Hall thrusters were selected to avoid the mass and cost of power conditioning.

1. Thruster specific mass of 3.4 kg/kW, based on 50 kW direct drive device and 170 kg mass, per NASA/TM-1999-209307, p7.

2. Propellant residual fraction of 0.03 assumed.
3. Propellant tank fraction of 0.1 assumed.

4. Initial orbit of 300 x 300 km /0 deg.
5. Final orbit of 35,786 x 35,786 km /0 deg.
6. Degraded power mass scaled using 6,240 kg array mass and 2,000 kW.
7. Thruster performance and efficiency values of 4,000 seconds and 0.65 respectively, are projected based on 25.4 kWe TM-50 thruster performance (specific impulse of 3,325 seconds, and efficiency of 0.62). Krypton propellant assumed for specific impulse of 2,000 seconds and efficiency of 0.44. Xenon/krypton propellant mixture assumed for specific impulse of 4,000 seconds.

Table 1. Assumptions for the solar-electric transfer analysis.

The solar-electric transfer analysis used the code SECKSPOT, which was developed in the mid 1970's under a contract from the NASA Lewis Research Center (Now Glenn Research Center, GRC). The program was written by Ted Edelbaum and associates at the MIT Lincoln Laboratory. GRC gave it a new name and incorporated new features that facilitate operation.

The program determines performance associated with a minimum trip time trajectory for electric propulsion spacecraft. It can simulate solar-array degradation caused by trapped radiation such as in the Van Allen belts and shadowing of the spacecraft by the central planet. It allows the user to study orbital transfers from one closed conic to another closed conic. As long as the eccentricity of either the initial or the final orbit is not too large (<.65), the program will usually converge. Although originally an Earth-centered program, it can simulate trajectories about any of the nine planets and the Earth's Moon.

The program computes optimal planet-centered trajectories using the techniques of optimal control and orbital averaging. Averaging assumes that of the six orbital elements used to describe a closed conic trajectory, only true anomaly varies quickly in time. By integrating over some number of orbits, one can quickly quantify the effect of several revolutions into a smaller group of terms. The optimal control problem is solved using these averaged values rather than instantaneous values.

The current version employs many modifications made over several years by Carl Sauer of the NASA Jet Propulsion Laboratory. These modifications included correcting errors and improving the convergence properties of the two-point boundary value solver. Modifications made at GRC include generalizing the program to work about planets other than Earth, adding solar array materials properties, and features to facilitate additional data post-processing.

The GRC point of contact is John P. Riehl. Boeing Rocketdyne Propulsion and Power received SECKSPOT from GRC in early 1998. The tool has since been renamed SEPSPOT by GRC, but Rocketdyne continues to use the original name.

The results for the transfer with thrusters that deliver a specific impulse of 2000 s are shown in Fig. 13 and Table 2. The initial thrust-to-weight ratio was maintained to achieve the desired 90-day flight time. The finite thrust losses and delta-velocity are constant for variations in initial mass. The effects of Earth shadow and power degradation due to Van Allen radiation belt transit were considered in the SECKSPOT analysis. Power requirements to achieve a 90 day flight time vary between 731 and 2,192 kW for an initial mass of 30,000 to 90,000 kg respectively. Power degradation varies between 378 and 1,135 kW, or about 52 percent of the initial power. Payload mass varies between 19,013 and 57,044 kg. Calculation of payload delivered to GEO accounts for the degraded power, tank, residual propellant and thruster mass values. Krypton propellant is assumed. The results indicate that the GEO payload mass, initial power, and power degradation scale linearly with initial LEO mass. Also, the intercepts at zero LEO mass are near zero.

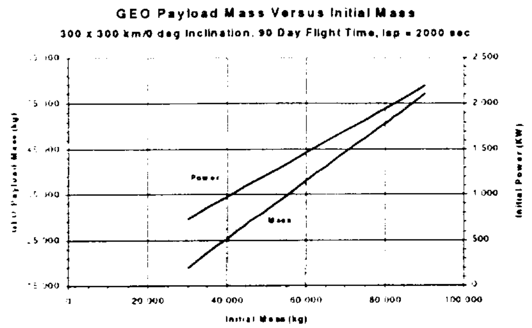

Figure 13. Power required and GEO payload as a function of LEO payload with 2000 s thrusters.

The results for the thruster with specific impulse of 4000 s are shown on Fig. 14 and Table 3. As with the specific impulse of 2000 s, the initial thrust-to-weight ratio was maintained to achieve the desired 90-day flight time, the finite thrust losses and delta-velocity are constant, and the effects of Earth shadow and power degradation due to Van Allen radiation belt transit were considered in the SECKSPOT analysis. Power requirements to achieve a 90-day flight time vary between 1,049 and 3,146 kW for an initial LEO mass of 30,000 to 90,000 kg respectively. The power levels are about 44% higher than the corresponding power levels with specific impulse of 2000 s. Power degradation varies between 541 and 1,622 kW, or about 52 percent of the initial power, the same percentage as with specific impulse of 2000 s. Payload mass varies between 20,864 and 62,597 kg. Calculation of payload delivered to GEO accounts for the degraded power, tank, residual propellant and thruster mass values, as before. A mixture of krypton and xenon is assumed to be needed for the propellant to allow the operation with a higher specific impulse. Krypton propellant is preferred, but only xenon propellant has been used. As before, the GEO payload mass, initial power, and power degradation scale linearly with initial mass, and the intercepts are near zero.

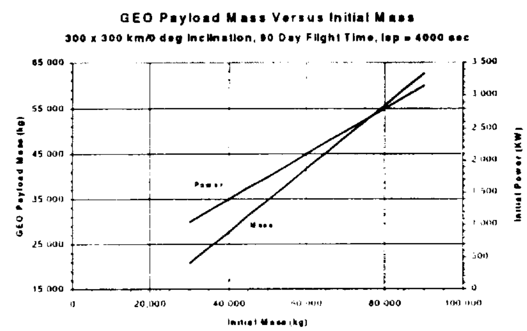

Figure 14. Power required and GEO payload as a function of LEO Payload with 4000 s thrusters.

Initial Mass kg	Initial Power kW	Final Power kW	Final Mass kg	Degraded Power kW	Degraded Mass kg	Thruster Mass kg	Tankage Mass kg	Propellant Mass Useable kg	Propellant Mass Residual kg	Payload Mass kg
30,000	731	353	23,538	378.3	1,180	2,485	666	6,462	194	19,013
35,000	853	412	27,461	441.4	1,377	2,900	777	7,539	226	22,181
40,000	975	471	31,384	504.5	1,574	3,315	887	8,616	258	25,349
45,000	1,097	529	35,307	567.6	1,771	3,730	998	9,693	291	28,517
50,000	1,219	588	39,230	630.7	1,968	4,145	1,109	10,770	323	31,685
55,000	1,340	647	43,153	693.4	2,164	4,556	1,220	11,847	355	34,858
60,000	1,462	705	47,076	756.6	2,360	4,971	1,331	12,924	388	38,026
65,000	1,584	764	50,999	819.7	2,557	5,386	1,442	14,001	420	41,194
70,000	1,706	823	54,922	882.8	2,754	5,800	1,553	15,078	452	44,362
75,000	1,827	882	58,845	945.5	2,950	6,212	1,664	16,155	485	47,535
80,000	1,949	940	62,768	1,008.6	3,147	6,627	1,775	17,232	517	50,703
85,000	2,070	999	66,691	1,071.4	3,343	7,038	1,886	18,309	549	53,876
90,000	2,192	1,058	70,614	1,134.5	3,540	7,453	1,997	19,386	582	57,044

Table 2. Results for 2,000 seconds specific impulse and 0.44 efficiency case.

Initial Mass kg	Initial Power kW	Final Power kW	Final Mass kg	Degraded Power kW	Degraded Mass kg	Thruster Mass kg	Tankage Mass kg	Propellant Mass Useable kg	Propellant Mass Residual kg	Payload Mass kg
30,000	1,049	508	26,574	540.9	1,688	3,567	353	3,426	103	20,864
35,000	1,224	593	31,003	631.1	1,969	4,162	412	3,997	120	24,341
40,000	1,399	678	35,432	721.3	2,250	4,757	471	4,568	137	27,817
45,000	1,572	761	39,861	810.8	2,530	5,345	529	5,139	154	31,303
50,000	1,747	846	44,290	901.0	2,811	5,940	588	5,710	171	34,779
55,000	1,922	931	48,719	991.2	3,093	6,535	647	6,281	188	38,256
60,000	2,097	1,016	53,148	1,081.4	3,374	7,130	706	6,852	206	41,733
65,000	2,272	1,100	57,577	1,171.6	3,655	7,725	765	7,423	223	45,209
70,000	2,447	1,185	62,006	1,261.8	3,937	8,320	823	7,994	240	48,686
75,000	2,622	1,270	66,435	1,352.0	4,218	8,915	882	8,565	257	52,162
80,000	2,796	1,354	70,864	1,441.9	4,499	9,506	941	9,136	274	55,644
85,000	2,971	1,439	75,293	1,532.1	4,780	10,101	1,000	9,707	291	59,120
90,000	3,146	1,524	79,722	1,622.3	5,062	10,696	1,059	10,278	308	62,597

Table 3. Results for 4,000 seconds specific impulse and 0.65 efficiency case.

The effect of specific impulse on useful delivered payload to GEO is shown in Fig. 15. The improved GEO payload for a given LEO initial mass is significant for the higher specific impulse, but the increased power required represents a significant cost.

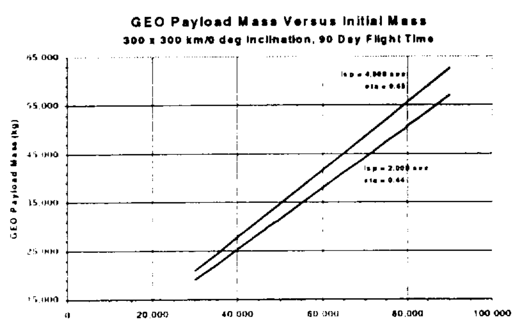

Figure 15. Effect of specific impulse on delivered GEO payload mass.

ORBIT TRANSFER COSTS

Cost analysis was completed on the autonomous transfer with the Hall thruster with specific impulse of 2000 sec. The results, shown in Fig. 16, indicate that launch costs, assumed to be $400/kg, are the most important part of the costs. The degraded array is not a significant direct contribution to the costs, but it does add to the launch costs. The initial costs, RDT&E, were used with the recurring costs to calculate the price that would be required if a commercial entity provided the transfer service. A 40% IRR was used, and a 25% markup was added to the recurring costs. The recurring costs did not include production support such as continuing engineering or financing of inventory in production, and the 25% factor represents markup to the recurring costs to cover these costs and profit. Because the costs are mostly recurring production rather than development of a reusable vehicle, the 40% internal rate of return was not a significant factor, but the 25% markup did have a noticeable effect.

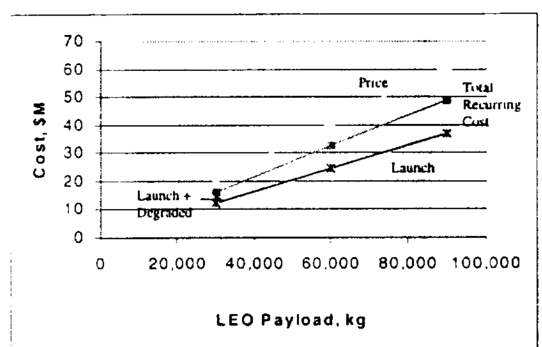

Figure 16. Commercial orbital transfer cost estimates vs. mass.

The costs shown in Fig. 16 were converted to cost per unit mass of useful payload in GEO, and the results are shown in Fig. 17. The results are shown to be insensitive to LEO payload. The launch of the GEO payload, assumed at $400/kg, is a major part of the costs. The launch of the additional mass required for the transfer adds about $230/kg. Other transfer costs, especially purchase of thrusters, add $228/kg for a total of $858/kg, close to the goal of $800/kg. The total cost decreases significantly if the price of launch is less than the \$400/kg assumed in these calculations.

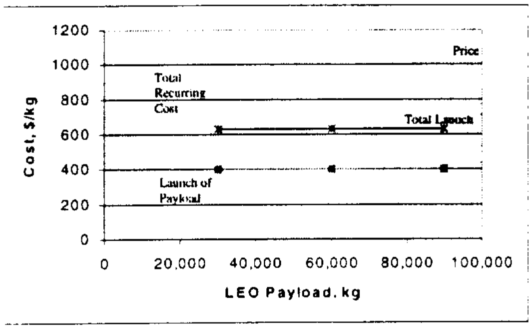

Figure 17. Orbital transfer costs per unit mass of useful payload in GEO.

CONCLUSIONS

The results of this study lead to the following conclusions:

1. A two-stage reusable rocket vehicle is a reasonable choice for Earth-to-orbit transportation. Other options may provide some cost reduction but could not be evaluated in this study.

2. Autonomous solar-electric propulsion from low earth orbit to geostationary orbit is a reasonable choice for the Sun Tower solar power satellites.

3. The cost of transportation is likely to be close to the goal of \$800/kg with the concepts studied.

ACKNOWLEDGEMENT

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