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Engineering Notes

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Optimized Low-Thrust Orbit Transfer for Space Tugs

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Introduction

RECENT advances in low-cost, commercial launch systems predict rapid insertion of relatively large payloads into low Earth orbit (LEO). At the same time, extensive work performed on radiation hardening of solar cells—specifically GaAs, InP, and InGaP/GaAs cells—has permitted repeated transfers through the Van Allen belt with minimal loss of conversion efficiency [1]. The combined impact of these two technological advances suggests that a low-thrust transportation system capable of ferrying payloads from LEO to geosynchronous Earth orbit (GEO) and beyond may provide an attractive alternative to chemical systems.

An electric space tug configuration would fill a need for an Earth–moon cargo vehicle system, capable of transferring payloads which do not require extensive on-orbit assembly or allow lunar surface assembly. The establishment of scientific research facilities or permanent outposts on the moon will require the transfer of such cargo to the moon's surface. A high power, reusable, solar electric propulsion system can be effectively optimized to deliver maximum cargo per year to the surface of the moon.

In this Note, the authors apply an optimization technique previously developed by Burton and Wassgren [2] to a reusable electrically propelled Earth–moon "space tug" transfer vehicle proposed by Spores et al. [3]. Both transfer mass and mission time are optimized, and the results show that significant reductions in the round-trip transfer time can be achieved. In addition, either the initial launch mass can be significantly reduced, while keeping the yearly delivered payload nearly constant, or the yearly delivered cargo can be significantly increased by appropriately increasing the available power.

Optimization Methodology

Burton and Wassgren derived a method that simultaneously optimizes transfer mass and mission time by maximizing the transfer mass ratio (TR), defined as

$$\mathrm{TR} = (m_t/m_0)/t_T \tag{1}$$

where t_T is the transfer time, and m_0 is the initial spacecraft mass and is defined as the sum of the propellant mass m_p, propulsion system mass m_{pow}, and the transferred mass m_t. It is assumed that the transferred mass includes the delivered payload mass as well as all masses which are not included under propellant or propulsion systems, such as communication, avionics, and structure.

For reusable, electric-powered space tugs considered in this Note, it is reasonable to assume that both thrust T and power P remain constant throughout the entire transfer time. In addition, because the electric thruster type (arcjet, magnetoplasmadynamic, ion, etc.) can be selected to maximize efficiency η at optimum u_e, it is assumed for simplicity that η is constant. With these assumptions, it is possible to write the round-trip transfer rate, from [2], as

$$\frac{\alpha(\Delta v)^2\mathrm{TR}}{2\eta} = \left(\frac{m_{\mathrm{pow}}}{m_0}\right)\left(\frac{\Delta v}{u_e}\right)^2 \frac{[e^{-\Delta v/u_e} - (m_{\mathrm{pow}}/m_0)e^{\Delta v/u_e}]}{1 - e^{-\Delta v/u_e} + (m_{\mathrm{pow}}/m_0)(e^{\Delta v/u_e} - 1)} \tag{2}$$

where u_e is the exhaust velocity, Δv is the characteristic velocity, and α is the powerplant specific mass (kg/W) that includes solar arrays and support structure, power processing units (PPU), and thrusters.

The resultant equation for the transfer rate is plotted as a function of powerplant mass ratio, m_{pow}/m_0 and $\Delta v/u_e$ and is shown in Fig. 1. The global maximum of the round-trip transfer rate, $\alpha(\Delta v)^2\mathrm{TR}/2\eta$, is equal to 0.0279 and occurs when $(\Delta v/u_e)_{\mathrm{opt}} = 0.432$ and $m_{\mathrm{pow}}/m_0 = 0.184$. Thus, for space tugs optimized for transfer rate, the optimum exhaust velocity is 2.31 times the mission Δv, and the propulsion system mass is 18.4% of the initial mass.

It is thus possible, given power plant specifications and desired Δv, to calculate an optimum transfer time and payload delivered for a constant thrust, electrical propulsion cargo transfer vehicle.

Transfer Rate Optimization of Earth–Moon Cargo Vehicle

The above-developed method for optimizing constant thrust electric transport vehicles is now applied to the specific configuration presented by Spores et al. The mission considered involves transfer of cargo from a low Earth orbit to a low lunar orbit (LLO) and the return of all reusable transfer vehicle components, which include the solar arrays, thrusters, and power processing units. Note that in [3], the authors present several configurations with different combinations of power I_{sp} and on-orbit mass. For demonstration purposes, only "configuration 1" will be considered in this Note, but the exact same procedure can be applied to the remaining cases yielding similar results. Configuration 1 consists of a solar electric propulsion cargo vehicle that uses four 150 kW Hall thrusters (620-kW end-of-life solar array including losses) that use xenon gas and provide an I_{sp} of 2500 s. The total initial mass of the spacecraft (m_0) in [3] is 77,456 kg, which allows for a delivery of 22,622 kg of payload to the surface of the moon in approximately 198 days (with a round-trip time of 237 days) and consumes a total of 25,502 kg of propellant.

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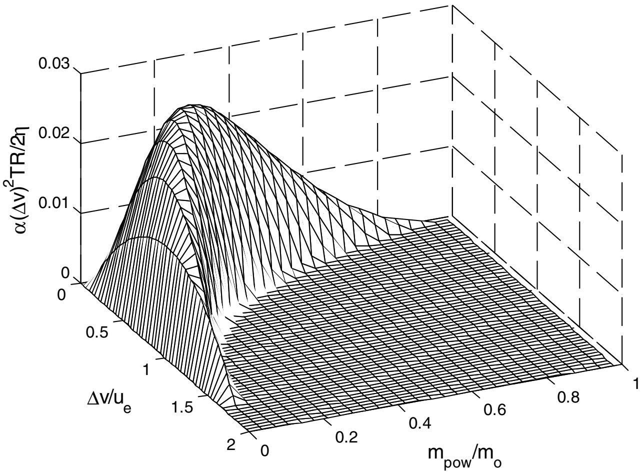

Fig. 1 Round-trip transfer rate for constant thrust as a function of powerplant mass ratio m_{pow}/m_0 and $\Delta v/u_e$. Maximum $\alpha(\Delta v)^2\mathrm{TR}/2\eta = 0.02790$ occurs at $m_{\mathrm{pow}}/m_0 = 0.184$ and $\Delta v/u_e = 0.432$.

The above initial mass was calculated in two steps. First, the descent propellant was calculated from the dry mass delivered to the surface and the scaling factors of a chemical system described by Spores et al. The sum of the dry mass delivered to the surface and descent propellant is simply the transfer mass m_t of the overall mission. Since both the mass of the propellant for the translunar injection and the dry mass of the orbital transfer vehicle (payload is not part of the orbital transfer vehicle) are given, the initial mass of the overall system can be calculated. In addition, the provided values of $\alpha = 0.0174$ kg/W, $\Delta v = 7835$ m/s, and $\eta = 0.63$ (assuming 97% PPU efficiency), allow calculation of the value of $\alpha(\Delta v)^2\mathrm{TR}/2\eta$ as 0.02211. The results of these calculations can be found in Table 1, which also includes corresponding optimum values obtained from the Burton–Wassgren optimization method.

It can be seen that the value of $\alpha(\Delta v)^2\mathrm{TR}/2\eta$ with the parameters proposed by Spores et al. falls below the optimal value of 0.0279. It is thus possible to rescale the mass fractions and maximize the mass transfer rate. Two different approaches were examined with either power or initial mass being kept in agreement with the proposed Spores et al. mission. The first analysis assumed limitations on the launch vehicle capability and held the initial mass constant, allowing use of the Atlas V launch vehicle as in the Spores et al. design. The second analysis assumed practical limits on the size of the solar array and held the power constant at 600 kW. The results of both analyses are shown in Table 2.

Discussion of Results and Conclusions

The optimization technique developed in [2] was used to optimize a LEO to a LLO cargo transfer mission by maximizing the mass transfer rate. Two separate constraints were applied to a specific configuration (configuration 1) presented by Spores et al. to optimize the mission. In the first approach, the initial mass was kept equal to the original mission and power was adjusted to maximize the transfer mass ratio. In the second approach, the power was kept at a level proposed in the original mission (600 kW) while the mass ratios were optimized. Both methods showed a significant reduction in the round-trip time with either an increase in the yearly delivered cargo or a reduction in the initial launch mass.

The optimized fixed initial mass analysis reduced the round-trip transfer time by 46% to 128 days and allowed a 26% increase in the yearly mass transferred. The power was scaled from 600 to 823 kW which corresponded, along with I_{sp} adjustment to 1850 s, to a thrust increase of 85%. This configuration is especially attractive if used with the refractive concentrator solar cell described by Spores et al. because it has one of the highest efficiencies and lowest required array areas, which makes it an excellent candidate for scaling to meet the new power requirements. Although this type of solar cell is limited to approximately 13 slow spiral transits through the Van Allen belt (seven outbound and six inbound) [3], the reduction in transfer time by 48% will limit the exposure time of these cells to harmful radiation and reduce their degradation, in effect allowing more transfers. It is thus possible to achieve a relatively large increase in the cargo rate delivery to the lunar surface without increasing the initial mass by appropriately scaling the power system.

The optimized fixed power analysis achieved the same round-trip transfer time of 128 days as the fixed initial mass case, but it also reduced the launch mass by approximately 27%. The reduction in the launch mass is considered an advantage for this type of mission when the emergence of rapid insertion, commercial launch systems is taken into consideration. Although the cargo delivered per trip to the lunar surface is reduced by half as compared to the Spores et al. configuration, the large reduction in the transfer time allows for nearly identical cargo rate delivery on an annual basis. As a result, this design offers an attractive alternative to larger but slower transfers by taking advantage of cargo that can be assembled on the lunar surface or requires shorter transfer time. The advances in radiation hardening of solar cells, whose lifetime was previously the limiting factor in reusability of such transfer vehicles, now allow for numerous trips through the Van Allen belt without significant loss of performance. It is thus advantageous for a fixed power configuration to scale down the initial mass and the specific impulse to achieve a shorter transfer time and take advantage of the reusable nature of the solar electric propulsion system, while preserving the annual cargo rate of the larger initial mass system.

Last, it is important to note that the optimal solution for sending the maximum mass on a cargo rate basis involves lowering the specific impulse of the cargo vehicle, not increasing it. It was assumed during the analysis that η was independent of the specific impulse. Although this may not be true for any given electric thruster, it is assumed that an electric thruster with the assumed efficiency at the required specific impulse is available. In practice, the thruster with the best performance at the required specific impulse will be chosen.

Table 1 Comparison of mass ratios for the optimal Burton–Wassgren and Spores et al. configurations

Method	m_{pow}/m_0	m_t/m_0	m_p/m_0	$\Delta v/u_e$	$\alpha(\Delta v)^2\mathrm{TR}/2\eta$
Spores et al.	0.134	0.536	0.329	0.319	0.02211
Optimized Burton– Wassgren	0.184	0.365	0.450	0.432	0.02790

Table 2 Comparison of original Spores et al. and optimized configurations

	m_0, t	P, kW	I_{sp}, s	m_t, t	T, N	Cargo delivered per trip, t	Lunar round-trip thrust time, days	Cargo rate, t/yr
Spores et al.	77.5	600	2500	41.5	30.9	22.6	237	34.8
Optimized (fixed m_0)	77.5	823	1850	28.3	57.3	15.4	128	43.9
Optimized (fixed power)	56.5	600	1850	20.7	41.7	11.2	128	32.1

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