A study has been performed to parametrically determine optimum performance parameters of an operational solar electric orbital transfer vehicle (SEOTV) capable of initial operating capability in the year 2000. The SEOTV concept is to be capable of near-earth transfer of payloads from low earth orbit (LEO) to geosynchronous earth orbit (GEO) and from LEO to global positioning system (GPS) orbit. The objective of this study was to search for the optimum specific impulse operating range that results in maximum vehicle payload fraction while minimizing vehicle life cycle costs with reasonable mission trip times. Results of the study were used to aid in propulsion system selection for the SEOTV concept. A parametric expression was developed that accounts for all variables in the problem including the variation of power conditioning unit efficiency, thruster efficiency, tankage fraction, and specific power of the power system with specific impulse. The expression was solved for various performance parameters of interest including limit trip time, required power to initial mass ratio, thrust to mass ratio, payload mass to power mass ratio, and payload fraction. Results indicate that for all performance parameters of interest, thrusters in the 1000 s to 1300 s specific impulse operating range showed optimum characteristics. Thrusters in this range correspond to arcjets operating on hydrogen propellant.

Nomenclature

c effective exhaust velocity, m/s

g_o gravitational constant, 9.8 m/s^2

I_sp specific impulse, s

m_f final vehicle mass, kg

m_o initial vehicle mass, kg

m_p propellant mass, kg

m_pl payload mass, kg

m_power power system mass, kg

P_req total power required for mission, W

t_s trip time (burn time), s

TF tankage fraction

\( \alpha_{pow} \) spec. power of pwr-dependent components, W/kg

\( \alpha_{pro} \) spec. power of the propulsion system, W/kg

\( \Delta V \) velocity increment for mission, m/s

\( \eta_{pcu} \) efficiency of power conditioning unit

\( \eta_t \) efficiency of thruster

Abstract

I. Introduction

The United States Air Force (USAF) has the requirement to launch the nation’s military satellites into their desired orbits and to develop and acquire the most cost-effective means to meet this requirement. A large portion of operational Department of Defense (DoD) missions focus on transfers between low earth orbit (LEO) and geosynchronous earth orbit (GEO) that will include orbit raising and plane changing. Recent studies\(^1,2\) have concluded that a solar electric orbital transfer vehicle (SEOTV) may reduce the costs, enhance space mission survivability, and increase the reliability of transferring satellites. An SEOTV may allow significant savings in both launch costs and upper stage operational costs by enabling the use of smaller, less expensive first-stage vehicles such as Delta II or Atlas II to transfer satellites of similar mass that currently require Titan-class boosters.

Many papers\(^3,4\) discuss the various advantages and disadvantages of electric propulsion versus chemical propulsion for performing these missions. Several other works\(^5,6,7,8\) address comparisons between the different types of electric propulsion systems with regard to performance of orbit transfer. In general, the approach used was to vary the rocket equations and apply different mass distributions and reach conclusions on the basis of trip time and payload delivered without special regard to optimizations. Still other works\(^9,10,11,12\) did attempt to optimize performance parameters for the orbit transfer mission under the assumption of constant specific power (W/kg) of the power and propulsion system. More recent works by Auweter-Kurtz\(^13\) et. al. did characterize the variation of specific power during the mission and the optimization process, but under the assumption that the power conditioner and the specific power of the power system did not change while varying specific impulse. Recent systems analysis work on EOTVs and ELITE by the Aerospace Corporation\(^14\) did account for the variation of specific power during a mission and developed an expression for the vehicle power density accounting for all power-dependent components.

This objective of this paper is to build on these valuable works by performing an SEOTV parametric systems analysis that accounts for all variables in the problem while searching for the optimum specific impulse operating range that results in maximum payload fraction with reasonable trip times. Special emphasis is given to the LEO to GEO mission and to electric propulsion performance characterizations that are possible in an operational SEOTV designed for use in the 2000 to 2005 timeframe. This study assumes a constant specific impulse, continuous thrust orbit transfer mission using solar power. Because the SEOTV is envisioned to have initial operating capability by the year 2000, candidate electric propulsion systems

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have been narrowed, for the purposes of this paper, to include ammonia arcjets, hydrogen arcjets and ion engines using argon, krypton and xenon propellants. Many past studies concentrated on very high constant power levels (30-100 kW) or more. Because of the time constraint, we are studying solar power systems with end of life (EOL) powers ranging from 10 to 25 kWe.

Quantitative comparisons are based on several performance parameters including limit vehicle trip time, required power to vehicle initial mass ratio, thrust to mass ratio, payload mass to power system mass ratio and payload fraction. All performance parameters are plotted as a function of specific impulse. This purpose of this phase of the study is primarily to perform engine/propulsion system selection for an operational SEOTV for the year 2000. The variation of power conditioning unit efficiency, thruster efficiency, tankage fraction and the specific power (W/kg) of the power and propulsion system with specific impulse are all accounted for. In this sense, the results presented differ from the previous studies. Other overall criteria for the operational SEOTV are to transfer 2,500-15,000 lbs. from LEO to GEO and 3,000 lb from LEO to GPS orbit. The maximum allowable trip time for the missions in this study is set at one year (365 days) duration.

The work presented represents part of an on-going USAF Operational Solar Electric Orbital Transfer Vehicle (SEOTV) Concept Design Study being performed to assess life cycle cost, performance capabilities, vehicle optimization, and to identify needed technology development in power and propulsion systems as well as vehicle GN&C issues. The SEOTV is envisioned to be used with today’s expendable launch vehicles and their derivatives.

II. Parametric System-Level Analysis

The objective of the parametric systems analysis for the SEOTV Concept Design Study is to develop a methodology to determine performance parameters that aid in power and propulsion system selection. The results are used to guide power and propulsion system requirements definition. The requirements defined at the system level can then be passed down to the subsystem level to guide development efforts and identify needed technology advancements for an operational SEOTV. Estimated life cycle costs and performance of expendable, expendable-integral, fully integral, and reusable SEOTV's can then be assessed. The least complex operations that enable the SEOTV to out-perform advanced chemical stages for less cost can then be identified.

The approach used was to conduct a concentrated literature search on all parametric systems analyses and optimization studies available directly related to the problem. Based on the previous work, a rocket equation expression was developed accounting for all variables in the case of a separately-powered vehicle. The expression was then solved for several performance parameters of interest. The expression developed relies on accurately modelling the thruster efficiency, PCU efficiency, tankage fraction, and specific power of the power and propulsion system variation with specific impulse. It was also important to consider the full range of electric propulsion system specific impulse (i.e 600s < Isp < 4600s) applicable to the LEO to GEO mission and account for degradation of the solar power supply during the optimization process and the mission analysis. Past studies only considered operation in the specific impulse regime where engine efficiency increases with specific impulse (i.e ion, MPD engines) and constant available power (i.e., a nuclear power supply).

Definition of Parameters - Electric Propulsion Orbit Transfer

This section develops a means for deriving a rocket equation expression that relates all performance parameters of interest for the case of a separately powered SEOTV. For both chemical and separately powered vehicles, the simplest rocket equation relation can be expressed as (see nomenclature)

\[ \Delta V = c \ln \left( \frac{m_0}{m_f} \right), \quad \text{where} \quad c = g_0 I_{sp} \]  

In this study we will consider the initial vehicle mass to be made up of simply the propulsion system (including power source), payload, and propellant. No further mass breakdown is necessary for the purposes of this paper and its trends and results. Therefore, the initial vehicle mass in LEO can be expressed as

\[ m_0 = m_{pl} + m_{pro} + (1 + TF)m_p \]  

Substituting equation 2 (initial and final mass) into equation 1 results in a rocket equation expression for a separately-powered vehicle where the power system is considered part of the propulsion system.

\[ \Delta V = c \ln \left( \frac{1 + TF + c^2/2 \alpha_{pro} \cdot \alpha_{thr}}{m_{pl}/m_0} + TF + c^2/2 \alpha_{pro} \cdot \alpha_{thr} \right) \]  

A very important part of equation 3 is the dependence of \( \alpha_{pro} \) with specific impulse. \( \alpha_{pro} \) the specific power of the power and propulsion system was defined by Auweter-Kurtz et. al. and includes the efficiency of the power conditioning unit and the thruster. It also includes the specific power of the solar power system and the rest of the power-dependent components. In effect, \( \alpha_{pro} \) accounts for the specific power of the power system components (array, deployment system, batteries, cover glass, switching, cables, etc.) and the specific power of the propulsion system power-dependent components (PCU, thruster, cables, gimbals, etc.). The definition of \( \alpha_{pro} \) is shown in equation 4.

\[ \alpha_{pro} = \eta_{pcu} \cdot \eta_{thr} \cdot \alpha_{pow} \]  

Equation 4 shows that \( \alpha_{pro} \) is strongly dependent on specific impulse since \( \eta_{pcu}, \eta_{thr}, \text{and } \alpha_{pow} \) all vary with specific
impulse. Many studies\textsuperscript{1,4,5,6,11} have realized the importance of the characterization of thruster efficiency with specific impulse. Not only the magnitude, but the slope and shape of the efficiency versus specific impulse curves for various engines are important. However, the author is aware of no papers that account for the variation of PCU efficiency with \( I_{sp} \) in both the arcjet and ion engine specific impulse regimes. The author is aware of only one study\textsuperscript{14} that has attempted to model the variation of specific power of the power dependent components with \( I_{sp} \). Jones\textsuperscript{3} did characterize the specific mass \((\text{kg/kW})\) of various electric propulsion systems and included it in the analysis.

Almost all past studies cite the required propellant for any mission irrespective of the propulsion system type which is often referred to as the "rocket equation" given in equation 5.

\[
m_p = m_o \left( 1 - e^{-\Delta V/c} \right)
\]  

(5)

Many references would then go on to use the Edelbaum \( \Delta V \) equation\textsuperscript{15}, the equation for mass flow rate, and solve for one-way or round trip times. This paper, however, will stress the characterization of the specific payload mass \((\text{payload mass normalized by total vehicle mass})\), or payload fraction as the chief performance parameter of interest. When the fundamental rocket equation shown in equation (1) is solved for payload fraction, significant differences arise in the form of the expressions. As noted in a recent paper\textsuperscript{14}, the payload fraction for a chemical propulsion OTV can be expressed as a function of only two major parameters, namely specific impulse and velocity increment;

\[
m_p = m_{pl}(I_{sp}, \Delta V)
\]

When equation (1) is solved for such a system, the analyst would find that the payload fraction for chemical systems can be written as

\[
\frac{m_{pl}}{m_o} = 1 - \frac{1}{\lambda_p}(1 - e^{-\Delta V/c})
\]

(6)

where \( \lambda_p \) denotes the propellant fraction or propellant mass divided by the "burnout" mass or dry mass of the vehicle minus payload \((m_p/(m_o-m_{pl}))\) for a chemical system.

With SEOTV's however, the payload fraction \((m_{pl}/m_o)\) is found to be a function of 7 major parameters;

\[
m_{pl} = m_{pl}(I_{sp}, \Delta V, I_b, \eta_t, \eta_{pcu}, \alpha_{pow}, TF)
\]

In the case of electric propulsion then, not only is the energy of the mission \((\Delta V)\) important as is the specific impulse, but now the amount of payload is determined by the trip time taken, the efficiencies of the propulsion system, the specific power of the power and propulsion system and the tankage fraction of the propellant tank. When the rocket equation (3) is solved for payload fraction we get:

\[
\frac{m_{pl}}{m_o} = 1 - \frac{c^2}{2\alpha_{pow}I_b}(1 - e^{-\Delta V/c}) - (1 + TF)(1 - e^{-\Delta V/c})
\]

(7)

### Propulsion System Characterization

The previous section discussed the characterization of the rocket equation accounting for all specific impulse-dependent parameters. To correctly compare candidate propulsion systems, the thruster efficiency, PCU efficiency, and propulsion system specific power \((\text{W/kg})\) must be accurately modelled and characterized. For the purposes of this study, the characterizations will reflect thruster, PCU and other component masses and performance values estimated for the year 2000 IOC. The propulsion system components must therefore be capable of being tested and qualified several years before.

With this assumption, an extensive database search was conducted on thrusters, PCU's, and other components in the propulsion system. It is the author's opinion that only small increases in thruster efficiency are possible in the next ten years. Most likely, thruster efficiency increases of less than 5-15% are expected above today's (and 1960s) state of the art thrusters. Rather, the emphasis should be on increasing thruster lifetimes necessary for the long-duration missions of an SEOTV.

Under the assumption that thruster efficiencies will not increase over 15% relative to thrusters tested in the last three decades, a large literature search was performed on thruster data from the 1960's to present. Careful attention was paid to past tests with regard to validity and reproducible data. Based upon the various references of thruster test data from the past three decades, as well as recent 1990 and 1991 data, a plot of electric propulsion thruster efficiencies vs. specific impulse was generated and is shown in Figure 1. Note that a noticeable gap exists in demonstrated arcjet and ion engine performance in a range of specific impulse from approximately 1600 s to 2500 s. Note also from the Figure 1 the lower limit of specific impulse for the different ion engine inert gas propellants and the efficiency variations existing at the same \( I_{sp} \). As will be shown later in the study,

![Figure 1. Thruster efficiency as a function of specific impulse](image-url)
Engines in this "gap" region of specific impulse will not affect overall performance parameter optimum results, contrary to past studies. Because of recent xenon gas availability concerns, this study will assume use of the krypton ion engine performance shown in Figure 1 for all subsequent performance calculations.

Thruster efficiency variation with Isp has the most dramatic effect on the rocket equation expression results when plotted as a function of Isp. However, the variation of PCU efficiency with Isp must also be characterized. Therefore, a database search of tested PCUs was conducted the results of which are shown in Figure 2. There are a limited number of PCUs that have been developed in the US to characterize this behavior. In general, arcjet PCUs have slightly higher overall efficiency compared to the more complicated PPUs designed for various ion engines.

As mentioned earlier, the representative tankage fractions for the various propellants used in each engine Isp operating regime must be accounted for. Table 1 lists the tankage fractions used in this paper. The numbers are assumed to include support structure, hardware, and plumbing.

<table>
<thead>
<tr>
<th>Propulsion system</th>
<th>Tankage Fraction</th>
</tr>
</thead>
<tbody>
<tr>
<td>NH₄ Arcjet</td>
<td>.12</td>
</tr>
<tr>
<td>H₂ Arcjet</td>
<td>.15</td>
</tr>
<tr>
<td>Kr Ion</td>
<td>.18</td>
</tr>
</tbody>
</table>

Power System Characterization
As mentioned in Section 1, most past studies assumed a constant specific power of the SEOTV power system both during the mission analysis and during the optimization process. Contrary to this assumption, however, the power level at the end of the mission may vary greatly from that when the mission began. This was true for interplanetary missions because of increasing distance from the sun, and it is very true in the LEO to GEO mission where an SEOTV using photovoltaic solar arrays slowly spirals through the Van Allen Radiation belts. The arrays will be subject to high energy trapped protons (peaking at about 6000 km altitude) and to high energy trapped electrons (peaking at approximately 20,000 km). Mission studies reveal that depending on the type of solar cell and cover glass thickness chosen, power degradation levels reaching 60% of BOL power are possible. Therefore, the assumption of constant specific power (W/kg) of the solar arrays is invalid for the purposes of this paper and its results.

In fact, the specific power of the power system effectively changes with mission trip time as well. For example, a vehicle performing a 300-day mission will spend much more time in the radiation belts than one lasting 100 days and therefore see a much greater power degradation level.

The mass of the power system will not change during the mission, but the amount of available power will certainly change, and therefore the available specific power (W/kg) will decrease with increasing trip time. Characterizing this effect is beyond the scope of this paper since here we are interested in how specific power varies with specific impulse.

To illustrate this point, a number of missions were analysed with a tool capable of modelling the power degradation during missions of different duration and with different propulsion systems (to vary the specific impulse). Figure 3 presents results of this analysis assuming a 6 mils cover glass solar array utilizing Si cells. Figure 3 shows how the specific power of the solar array power system (s pow) can vary with increasing trip times. Because this paper is dealing with parameters that vary with Isp, this effect is not included in this study. However, Figure 3 also shows that the required specific power is also very dependent on Isp. To complete the same mission (equal payload, ΔV, trip time) it is shown that a much larger s pow is required for EP engines of the higher Isp regime (i.e. ion). It is this dependency of the required s pow that is included in this optimization study.

![Figure 2. Power conditioning unit (PCU) efficiency as a function of specific impulse](image)

![Figure 3. Specific power (W/kg) of the power and propulsion systems as a function of specific impulse and trip time](image)
III. Calculation of Performance Parameters

Having characterized separately each element of the expression developed in equation 4 for the specific power of the propulsion and power system and their variation with specific impulse, we can now proceed with the optimization process accounting for all variables and determine the optimum \( I_{sp} \) operating range that results in maximum payload fraction and other performance parameters for the SEOTV LEO-GEO mission.

We start with the rocket equation expression (3) for a separately-powered vehicle developed in Section 1.

\[
\Delta V = c \ln \left( \frac{1 + TF + c^2/2 \alpha_{pow} I_{sp}}{(m_P/m_o) + TF + c^2/2 \alpha_{pow} I_{sp}} \right)
\]

**Trip Time**

If equation (3) is solved for trip time \( t_b \), the resulting equation is shown below:

\[
t_b = \frac{c^2}{2 \alpha_{pro}} \left[ \frac{(1 - e^{-\Delta V/c})}{1 - (m_P/m_o) - (1 + TF)(1 - e^{-\Delta V/c})} \right]
\]

Note that the burn time (or trip time) expression in equation 8 is a function of \( \alpha_{pro} \), \( \eta_{pmod} \), \( \eta_{pcu} \), and \( \alpha_{pow} \), and the tankage fraction \( TF \). Note also that trip time is a strong function of the payload mass fraction \( (m_P/m_o) \). Equation 8 gives the burn time for a constant specific impulse mission but does not, however, necessarily assume a constant power level during the mission. The variation of \( \alpha_{pro} \) with specific impulse can account for variable power.

A simplifying assumption can be made to the equation for burn time in equation 8 by allowing the payload mass to approach zero. In this case, the payload fraction would also be zero, and the resulting equation would be an expression for an SEOTV limit trip time using various electric propulsion systems. Figure 4 illustrates the SEOTV limit trip times vs. specific impulse for a variety of specific power values. As shown, the shortest limit trip times are realized for hydrogen arcjet propulsion systems in the specific impulse range from 900 s to 1200 s. This result is shown to hold true for any value of \( \alpha_{pow} \). For smaller values of \( \alpha_{pow} \) (20 - 30 W/kg) such as with today's available technology, limit trip times for arcjet systems can be as much as 40% lower than for even optimum ion engine systems. Figure 4 also shows that the choice of \( I_{sp} \) for arcjet propulsion systems can dramatically affect the SEOTV limit trip time because of the steep slope of the trip time curves. For ion engines, however, a non-optimum \( I_{sp} \) selection affects the limit trip time by less than 10%.

**Thrust to Mass Ratio**

A second parameter of interest to the electric propulsion systems engineer is the thrust-to-mass ratio. For various electric propulsion systems it is possible to compare performance at the same specific impulse by comparing the thrust-to-mass ratios for the different systems. The thrust to mass ratio is found to be a function of propulsion system efficiencies, specific impulse and the specific power of the power and propulsion system as shown in equation 9.

\[
\text{Thrust} = \frac{2 \eta_{pmod} \eta_{pcu} \alpha_{pow}}{8 \alpha_{pro}} \frac{2 \alpha_{pro}}{c}
\]

Figure 5 plots the thrust-to-mass ratio in N/kg vs. specific impulse for a variety of specific power values. Figure 5 shows that the highest values of thrust to mass exist for ammonia arcjet systems at very low values of specific impulse (e.g., 600 s) that fall off sharply as \( I_{sp} \) is increased to about 850 s. Hydrogen arcjets exhibit greater thrust to mass ratios in their "normal" operating range (\( \approx 1000 \text{ s} \) - 1200 s) when compared to ammonia arcjet systems at 800 s. Figure 5 also shows that, in general, arcjet propulsion systems have at least a factor of 2 greater thrust-to-mass ratio than ion engines for all values of specific power plotted. It is also interesting to note that the thrust-to-mass ratios for ion engine systems vary only slightly for specific impulse values ranging from 1600 s to 4500 s, while arcjets fall off very quickly with increasing specific impulse.
Required Input Power
To illustrate the magnitude of the power required for orbit transfer of a vehicle from LEO to GEO, an equation was derived that can calculate the ratio of required power to initial mass of the vehicle based on the specific impulse and allowable trip time as parameters. The expression is shown in equation 10.

\[
P_{\text{req}} = \frac{m_0}{2 I_b \eta_i \eta_{\text{pro}}} \left(1 - e^{-AV/c}\right)
\]  

Equation 10

The analyst may then chose a value for the initial vehicle mass \(m_0\), \(\Delta V\), and the propulsion system and get a value for the required power to perform the mission. Figure 6 plots required power as a function of specific impulse for a LEO-GEO mission (\(\Delta V = 6000 \text{ m/s}\)) assuming an initial vehicle mass in LEO of \(m_0 = 5000 \text{ kg}\). The lowest values of required power are shown to exist for ammonia arcjets at low specific impulse (= 600 s) and for hydrogen arcjet systems in the 900 s to 1200 s Isp range. Note that as allowable trip time decreases, the amount of required power for the ion propulsion systems increases more rapidly. To compare the different propulsion systems, we can show an example where for instance, to complete a 100-day mission, an 800 s ammonia arcjet requires 40 kW, a 1000 s hydrogen arcjet requires = 30 kW, and a 3800 s xenon ion engine requires over 100 kW. As trip time is relaxed, however, say towards one year, all EP propulsion systems have power requirements for equal missions that grow closer together.

![Figure 6. Power required vs. specific impulse for a range of trip times to GEO](image)

If the total vehicle mass \(m_0\) is taken to the left side of equation 10 we can find a series of values of power required to initial mass ratio for a variety of propulsion systems and allowable trip times. This is shown as equation 11.

\[
\frac{P_{\text{req}}}{m_0} = \frac{c^2}{2 I_b \eta_i \eta_{\text{pro}}} \left(1 - e^{-AV/c}\right)
\]  

Equation 11

Figure 7 plots the ratio of input power required to initial vehicle mass as a function of specific impulse for a range of allowable trip times starting at 100 days and extending to 365 days. Trends are identical to Figure 6 curves. Note that the required power to mass ratio increases quite rapidly for arcjets relative to ion engines shown by the steep slope of the arcjet curves.

![Figure 7. Power required to initial SEOTV mass ratio vs. specific impulse for a range of allowable trip times to GEO](image)

Payload Mass to Power Mass Ratio
An interesting parameter to the SEOTV designer is the ratio of the payload mass delivered to the power system mass required for the mission. This parameter provides insight into the trades involved in propulsion system selection on both a performance and cost basis. For both performance and cost reasons the designer would want to maximize payload mass while minimizing power system mass and therefore would seek to maximize this ratio which has been developed as a parameter. The expression developed from the rocket equation (3) for this ratio is presented as equation 12.

\[
\frac{m_{\text{pl}}}{m_{\text{pow}}} = \frac{1 + c^2/2 \alpha_{\text{pro}} I_b (1 - e^{-AV/c})}{c^2/2 \alpha_{\text{pro}} I_b (e^{AV/c} - 1)}
\]  

Equation 12

Figure 8 represents the payload mass to power system mass ratio \(m_{\text{pl}} / m_{\text{pow}}\) as a function of specific impulse for several values of specific power. The figure indicates that very large spikes occur in the low Isp region of ammonia arcjets and the low to moderate Isp values for hydrogen arcjets. In general, the figure shows that arcjet propulsion systems exhibit a factor of 2 to 3 greater \(m_{\text{pl}} / m_{\text{pow}}\) ratio than ion engine propulsion systems, at least for the trip time slice of 180 days shown. This is important since a larger ratio here means more payload delivered with a smaller power system to complete the mission, which directly correlates to lower EOTV costs.
The payload mass to power mass parameter defined in equation 12 was modified slightly to provide an even more meaningful set of results in which not only measure payload weighted by power system mass fraction, but to also make comparisons with chemical propulsion.

The expression developed used the excess payload mass (net payload fraction greater than an advanced chemical upper stage) normalized by the required power system mass (Δm_{pl} / m_{pow}). The program used the curves for payload fraction discussed below for all three propulsion systems, and subtracted out the advanced chemical payload fraction (assumed to be 17%). When the Δm_{pl} was weighted by power system mass fraction and plotted against specific impulse (see Figure 9), significant benefits were realized for H2 arcjets. An excess payload to power mass ratio of zero would correspond to performance equal to the chemical stage. The largest excess payload to power fraction peaks occurred quite steadily at 1200 s Isp for all values of \( \alpha_{pow} \), as shown. Arcjets were again, in general, greater than the ion systems for this parameter.

Payload Mass Fraction
A most important parameter for the EOTV designer and for the electric propulsion promise as a whole is the payload mass fraction, or payload mass normalized by total vehicle mass defined earlier in the paper as \( (m_{pl} / m_{p}) \). It is important that the payload mass fraction for an EOTV far exceed that for planned advanced chemical stages. The equation for payload fraction can be derived directly from equation (3) and is shown here as equation 13.

\[
\frac{m_{pl}}{m_{p}} = 1 - \frac{c^2}{2 \alpha_{pro} \kappa_h} (1 - e^{-\Delta V/c}) - (1 + T_F)(1 - e^{-\Delta V/c}) \quad (13)
\]

It is seen that the normalized payload mass is a function of \( \Delta V, I_{sp}, I_{th}, T_F, \) and \( \alpha_{pro} (\eta_{PCU}, \alpha_{pow}) \). Figure 10 plots the payload mass fraction of EOTVs with various propulsion system characteristics as a function of specific impulse for increasing specific powers. The figure shown assumes a LEO-GEO mission (\( \Delta V = 6000 \text{ m/s} \)) with a 180-day transfer time for illustrative purposes. The curves in the figure indicate that hydrogen arcjet propulsion systems with specific impulse ranging from 1000 s to 1300 s offer the largest payload fraction for specific powers up to 25 - 30 W/kg. For specific powers above 35 W/kg, ion thrusters offer an ever-increasing payload fraction. Available power system technology and its power density will guide propulsion system selection. It is therefore important to determine both today’s and 9 to 10 year projections for the values of specific powers (W/kg) of all the power-dependent components included in the power and propulsion subsystems.

IV. Conclusions
A system-level parametric study on SEOTVs for LEO to GEO missions has been performed. A rocket equation expression was developed for separately-powered vehicles that enables the analyst to account for the variation of thruster efficiency, PCU efficiency, and specific power of...
the power and propulsion systems with specific impulse. A
number of performance parameters were defined based on
the rocket equation expression developed, including trip
time, required power to initial mass ratio, thrust to mass
ratio, payload mass to power system mass ratio, and pay-
load fraction.

Results indicate that with today’s power system technology,
vehicle limit trip times \((m_p \rightarrow 0)\) for arcjet propulsion
systems may approach 40% less than for ion engine systems.
The shortest limit trip times are realized for hydrogen arcjet
propulsion systems in the 900 s to 1200 s range of specific
impulse. Hydrogen arcjets in general have at least a factor
of 2 greater thrust to mass ratio than ion engines for all
values of specific power \((20 - 55 \text{ W/kg})\) considered. The
required power for a given mission was shown to be
minimized for ammonia arcjet systems at low Isp \((\approx 600 \text{ s})\)
while hydrogen arcjets in the 1000 s to 1200 s range
indicated less power required than ammonia arcjets at 800
s. As trip time becomes more demanding \((180 - 220 \text{ days})\)
it was shown that arcjet propulsion systems can require less
than half of the power required for ion engine systems to
perform the same missions. The ratio of payload mass to
to power system mass was shown to be maximized for low
specific impulse ammonia arcjets and low to moderate Isp
gydrogen arcjets. The arcjet propulsion systems exhibited
a factor of 2 to 3 greater payload to power ratio. The
payload mass fraction was maximized with hydrogen arcjet
systems \((1000 \text{ s to } 1300 \text{ s})\) for specific powers up to 25-30
W/kg. Above 35 W/kg, ion propulsion systems offer an
ever increasing payload fraction. When considering all
performance parameters in combination, results indicate
that hydrogen arcjet thrusters operating in the 1000 s to
1300 s specific impulse range showed the optimum char-
acteristics.

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