Comparison of Potential Electric Propulsion Systems for Orbit Transfer

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Electric propulsion concepts are compared on the basis of trip time for the LEO to GEO mission. Resistojet, arcjet, MPD, pulsed inductive, and ion engine thruster concepts are included. A simple procedure is used that provides the optimum specific impulse for minimum trip time independent of power supply or payload mass. The solution for trip time and propellant mass for the constant power, continuous low acceleration orbit transfer problem (one way and round trip) is also presented. The influences of mission ΔV, thruster efficiency, specific impulse, power level, propulsion and power system mass and payload mass can easily be discerned. The resistojet with its high thrust to power ratio provides the fastest trip time. There is no single minimum trip time specific impulse. The minimum trip time specific impulse depends on the particular technology being considered.

Nomenclature

g	$= 9.8$ m/s^2
I_{sp}	= specific impulse, s
M_0	= initial vehicle mass, kg
M_{pay}	= payload mass, kg
M_{pp}	= propellant mass, kg
M_{pow}	= power supply mass, kg
M_{pro}	= propulsion system mass, kg
$\dot{m}$	= propellant flow rate, kg/s
P	= total system power, W_e
TF	= tankage fraction
TT	= trip time, s
V	= circular orbit velocity, m/s
α_{pow}	= power system specific mass, kg/W_e
α_{pro}	= propulsion system specific mass, kg/W_e
ΔV	= mission specific energy, m/s
$\Delta\theta$	= inclination change, deg
η	= total system efficiency
η_{ppu}	= power processing efficiency

Introduction

IT seems likely that both NASA and the United States Air Force (USAF) will be expanding their activities in near Earth space. This assertion is based on the near operational status of the Space Shuttle,[1] NASA's current emphasis on space station planning,[2] and the USAF's increasing realization of the military value of near Earth space (the high ground) as evidenced by the creation of a space command.[3] An inescapable part of this increased near-Earth space activity will be transportation of payloads between orbits.

There is some controversy[4-9] as to the most appropriate role and extent of that role for electric (vs chemical) propulsion systems for near-Earth orbit transfer missions. Although there is controversy, there can be no question as to the utility of a careful consideration of potential electric propulsion systems, with the aim of identifying those that may best exploit their opportunity when called upon to perform an orbit transfer mission. Therefore, the objective of this paper is to compare several potentially developable electric propulsion systems for a constant power, constant specific impulse, continuous-thrust orbit transfer mission. The primary quantitative basis of comparison will be trip time. Other bases of comparison are life cycle cost, payload capability, technology readiness, etc.

There are many papers in the literature[5,7-13] dealing with electric propulsion systems for orbit transfer. Generally these papers deal with state of the art technology and do not evaluate electric propulsion technologies which are presently in the exploratory reserach and development phases. This paper attempts to improve upon the previous work by 1) evaluating all of the primary electric propulsion technologies currently under development in the United States (save one special class), and 2) using similar levels of technology extrapolation for those systems not well developed at the present time rather than only preliminary data, and 3) illustrating those features of the generic electric propulsion orbit transfer system/mission which will impact the choice of an electric propulsion technology and (hopefully) assist the electric propulsion technologist in better directing/focusing his work.

The propulsion technologies included in this paper are: 1) argon magnetoplasmadynamic (MPD), 2) argon pulsed inductive (PIT), 3) xenon ion, 4) argon ion, 5) mercury ion, 6) hydrogen electrothermal (arcjet and resistojet), and 7) the J series solar electric propulsion system (SEPS) mercury ion thruster system. SEPS is included in order to give the reader a point of reference. Teflon pulsed plasma thrusters are being developed for auxiliary propulsion functions and are clearly not competative (in their present form) for the orbit transfer mission.

Orbit Transfer with Electric Propulsion

This section develops a means by which one can take the fundamental variables of interest to the electric propulsion technologist and calculate the orbit transfer trip time. This section also illustrates how ΔV, M_{pay}, M_{pow}, and power affect the trip time. The purpose is not to develop a highly accurate trip time equation for all circumstances, but only to provide a tool for the quick evaluation of competing technologies.

An expression for the trip time of a continuous thrusting electric propulsion system is simply the propellant mass divided by the mass flow,

$$\mathrm{TT} = \frac{M_{pp}}{\dot{m}} \tag{1}$$

The required propellant for any mission irrespective of the propulsion system is given by the "rocket equation."

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$$M_{pp} = M_0(1-e^{-x}), \qquad x = \frac{\Delta V}{gI_{sp}} \tag{2}$$

The equivalent field free velocity increment, ΔV, for very low acceleration propulsion systems for orbit transfer was derived by Edelbaum.[14]

$$\Delta V = \left[V_2^2 + V_1^2 - 2V_2V_1\cos\left(\frac{\pi\cdot\Delta\theta}{2}\right)\right]^2 \tag{3}$$

In Eq. 3, V_1 and V_2 are the circular velocities of the beginning and final orbits and $\Delta\theta$ is the change in inclination. The value of ΔV for a nominal LEO to GEO mission (500-35,860 km plus a 28.5 deg plane change) is 5.847 km/sec. Most workers in the field (including this author) round this to 6 km/s to be conservative.

The statement for the initial vehicle mass best suited for the purposes of this paper is given in Eq. 4.

$$M_0 = M_{\rm pay} + M_{\rm pow} + M_{\rm pro} + (1+{\rm TF})M_{pp} \tag{4}$$

The power supply and the propulsion system are assumed to contain all their own structure, thermal control, and subsystem control devices. The tankage fraction is assumed to account for the propellant tank and distribution and control devices.

Reference 4 would add to this initial mass an estimate of vehicle avionics, integrating structure, auxiliary propulsion devices, secondary power system, and a reserve. These features give a more accurate representation of a true vehicle but do not contribute to the purpose of this section, i.e., providing a quick way to assess various electric propulsion technologies. The reader may adjust $M_{\rm pay}$ for the items that Eq. (4) does not explicitly account for after the trip time has been determined. Equations (2) and (4) may be used to obtain an expression for the propellant mass, Eq. 5.

$$M_{pp} = \frac{(M_{\rm pay}+M_{\rm pow}+M_{\rm pro})(1-e^{-x})}{[1-(1+{\rm TF})(1-e^{-x})]} \tag{5}$$

Rearranging the equation for electric propulsion system efficiency [Eq. (6)] gives the propellant flow rate and Eq. (7) the sought after trip time expression.

$$\dot{m} = \frac{2\eta P}{(gI_{sp})} \tag{6}$$

$$TT = \frac{(gI_{sp})^2(1-e^{-x})(M_{\rm pay}+M_{\rm pow}+M_{\rm pro})}{2\eta P[1-(1+TF)(1-e^{-x})]} \tag{7}$$

Equation (7) gives the trip time for a constant specific impulse, constant power orbit transfer mission with an unconstrained initial mass. This formulation of the problem assumes constant power, i.e., a nuclear power supply. Reference 15 presents the same trip time relation, except for the tankage fraction, modified to apply for a solar power supply. Reference 15 modifies Eq. (7) to attempt to take into account atmospheric drag at LEO, steering losses, solar array degradation, time spent in the shadow with no thrust, and thrusters that require a significant warm up time after coming out of the shadow. One might also wish to apply modifications for an oblate Earth, gravity gradient losses, and sun and moon perturbations.

Again, for the purposes of this paper, these adjustments only serve to complicate and conceal the desired result without altering the fundamental relationships and trends. The drag, steering, and perturbation effects may increase the trip time less than 5%. However, the losses associated with a solar powered system compared to an equal mass and power nuclear powered system can result in a factor of 2 greater trip time.

Although Eq. (7) may be "ideal" trip time, it contains those features of the mission which one would expect to see, i.e., payload mass, power supply mass, power supply power, propulsion system mass, total efficiency, specific impulse, and mission ΔV. Equations (5) and (7) are rewritten below in a form better suited to nomographs.

$$\mathrm{TT} = \frac{(gI_{sp})^2(1-e^{-x})\left[\frac{M_{\rm pay}}{\eta P} + \frac{\alpha_{\rm pow}+\alpha_{\rm pro}}{\eta}\right]}{2[1-(1+{\rm TF})(1-e^{-x})]} \tag{8}$$

$$M_{pp} = \frac{M_{\rm pay}\left(1+\frac{M_{\rm pow}+M_{\rm pro}}{M_{\rm pay}}\right)(1-e^{-x})}{[1-(1+{\rm TF})(1-e^{-x})]} \tag{9}$$

After calculating the trip time and required propellant, one can go back and use Eq. (4) to find the initial vehicle mass. Equations 8 and 9 are plotted in nomograph form in Figs. 1 and 2 in order to illustrate the fundamental trends and serve as a tool for estimating the one way trip time or propellant requirement. An example illustrating their use is included in Figs. 1 and 2. Important points to recognize from Fig. 2 are that the propellant mass: is independent of system efficiency and trip time for a given mission and system; decreases as specific impulse increases; increases as ΔV increases; and increases as payload, power, and propulsion system masses increase.

All of these points, except maybe the first, are intuitive and not surprising. In a similar manner, the important points to recognize for the trip time nomograph, Fig. 1, are that the trip time: increases with increasing ΔV; increases as the ratio of payload mass to thrust power $M_{\rm pay}/\eta P$ increases; increases as the sum of the power and propulsion specific mass divided by system efficiency increases; is always lower for a higher system efficiency; becomes effectively independent of power and propulsion system specific mass as the payload to thrust power ratio increases past about 0.1 kg/W_e; and becomes effectively independent of the payload to thrust power ratio as that ratio drops below about 0.001 kg/W_e.

The first three points are again intuitive, while the last two simply mean that the power and propulsion system mass have no effect on a payload-dominated vehicle and vice versa. Important as all the previous points about Fig. 1 are, so is the fact that Fig. 1 tells you nothing about the optimum (lease trip time) specific impulse because the relationship between system efficiency and specific impulse has not yet been specified. This will be discussed later. It should be sufficient at this point to realize that one cannot trust blanket statement regarding the LEO to GEO trip time with any electric propulsion technology without at least implying the mass of the payload and power supply, and the power level.

Using the same basic starting points, we can derive an equation for the round trip time. The same assumptions (constant power, constant specific impulse, unconstrained initial mass) apply, along with several new assumptions, i.e.,

Table 1 Tankage fractions for hydrogen, argon, xenon and mercury

Propellant	TF	Propellant mass, kg	Reference
H_2	0.12	20,000	17
H_2	0.14	10,000	17
H_2	0.16	1,765	18
H_2	0.18	N/A	4
Hg	0.01	N/A	16
Ar	0.05	10,000	16
Ar	0.1	1,000	16
Ar	0.04	N/A	4
Xe	0.09-0.14	~3,000	16

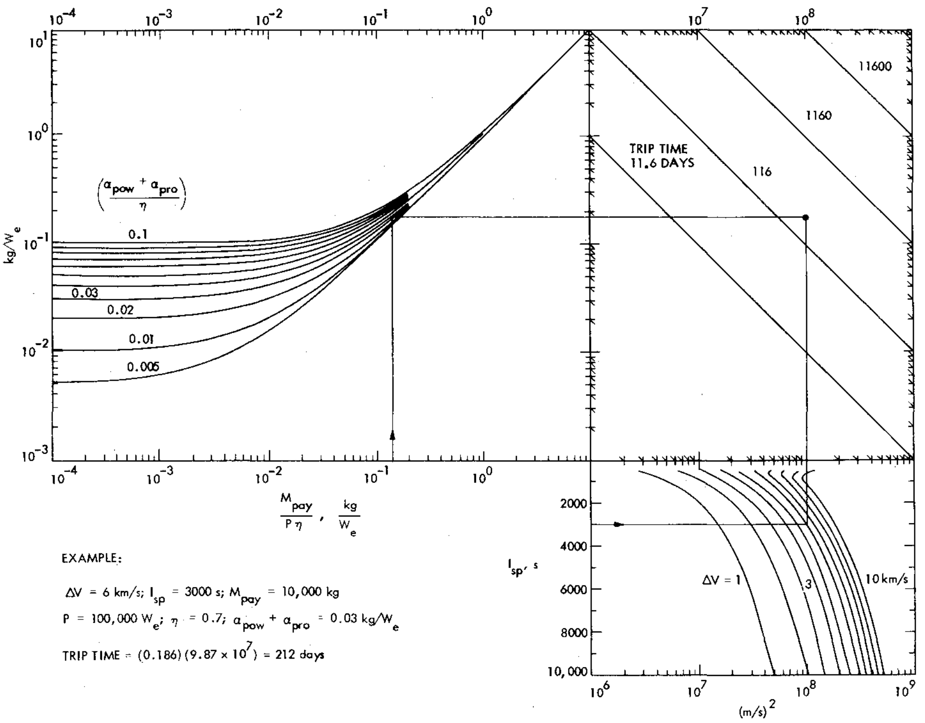

Fig 1 Nomograph for one way trip time, TF = 0.05.

the payload is dropped off at the end of the first leg, the ΔV for the up and down leg are the same, the specific impulse is the same for both up and down legs, and there is no loitering at the end of the first leg. Using the assumptions and the same procedure as was used in deriving the one way trip time [Eq. (7)], an expression for the round trip time can be obtained and is presented in Eq. (10).

$$\mathrm{TT} = \frac{(gI_{sp})^2}{2\eta} \times \left\{ \frac{(1-e^{-2x})\left[\alpha_{pow} + \alpha_{pro} + \frac{M_{pay}}{P}\left[1-(1+\mathrm{TF})(1-e^{-x})\right]\right]}{\left[1-2(1+\mathrm{TF})(1-e^{-x}) + (1+\mathrm{TF})(1-e^{-x})^2\right]} - \frac{M_{pay}}{P}(1-e^{-x}) \right\} \tag{10}$$

In Figs. 1 and 2 the tankage fraction was held constant at 0.05 to simplify the calculations. The propellants of interest for this paper are mercury, argon, xenon, and hydrogen. The tankage fraction is a function of propellant type and the mass of the propellant. Table 1 summarizes the results of previous propellant tank conceptual designs.

Roughly, the tankage fractions for mercury, argon, xenon, and hydrogen may be thought of as 0.01, 0.05, 0.09 and 0.15, respectively. Tankage fraction has the largest impact on missions where the propellant mass is high, i.e., low I_{sp} high ΔV missions. For the worst case ($I_{sp} = 1000$ s, $\Delta V = 10$ km/s) the maximum impact on the trip times and propellant masses shown in the nomographs (where TF = 0.05) is +20% for a TF = 0.15 and +8% for TF = 0.09 and −8% for TF = 0.01. For a more reasonable mission of $I_{sp} = 3000$s and $\Delta V = 6$ km/s, the maximum variations are +5%, +2%, and −2% for TF = 0.15, 0.09, and 0.01, respectively.These corrections are generally within the accuracy with which one can read the charts and in any case, do not change the fundamental trends and relationships.

Propulsion System Characterization

The previous section presented the trip time and required propellant equations for a one way orbit transfer mission and the equation for the round-trip time. The nomographs can be used to obtain first order estimates of the trip time and initial vehicle mass for any electric propulsion system. Nomographs for the round trip propellant, trip time for the up leg (payload leg) of a round trip, and the ratio of round trip to up leg trip time are presented in Ref. 19. In order to use this method to compare potentially developable electric propulsion systems, these systems must be characterized. Specifically the propulsion system efficiency and specific mass must be modeled.

Table 2 Models for propulsion system characteristics

System efficiency model: $\eta = A + B\mathcal{L}\mathfrak{N}I_{sp}$
System specific mass model: $\alpha_{pro} = CI_{sp}^{D}$

Thruster system	Modeling constants			
	A	B	C,kg/kW$_e$	D
H_2 Electro-thermal	–[a]	–[a]	5.0	0
Ar Ion	−2.024	0.307	4,490	−0.781
Xe Ion	−1.776	0.307	123,1000	−1.198
Hg Ion	−0.765	0.181	82,870	−1.136
Ar MPD	−0.591	0.126	7	0
Ar PIT	−1.99	0.32	7	0
SEPS	$\eta = 0.605$, $\alpha_{pro} = 28$ at 3000 s			

[a] Individual data points from text were used.

Figures 3 and 4 present the system characterizations for each of the seven electric propulsion systems compared in this paper. Figure 3 presents the total thrust system efficiency vs specific impulse. If the data source presented both thruster and power processing efficiency, then the power processing efficiency (assumed independent of specific impulse) is listed separately on Fig. 3.

The electrothermal curve is a combination of resistojet and arcjet technology. The curve is made up of four data points: 1) a 3 kW$_e$, 840 s, 75% efficient H_2 resistojet[20]; 2) a 30 kW$_e$, 1010 s, 54% efficient H_2 arcjet[18]; 3) a 30 kW$_e$, 1520 s, 44% efficient H_2 arcjet[18]; and 4) a 200 kW$_e$, 2120 s, 35% efficient H_2 arcjet.[18]

The electrothermal power processor efficiency comes from Ref. 4. The thruster efficiency vs specific impulse curves for the argon and xenon ion thruster come from Ref. 5, which is based on two previous papers, Refs. 21 and 22. These

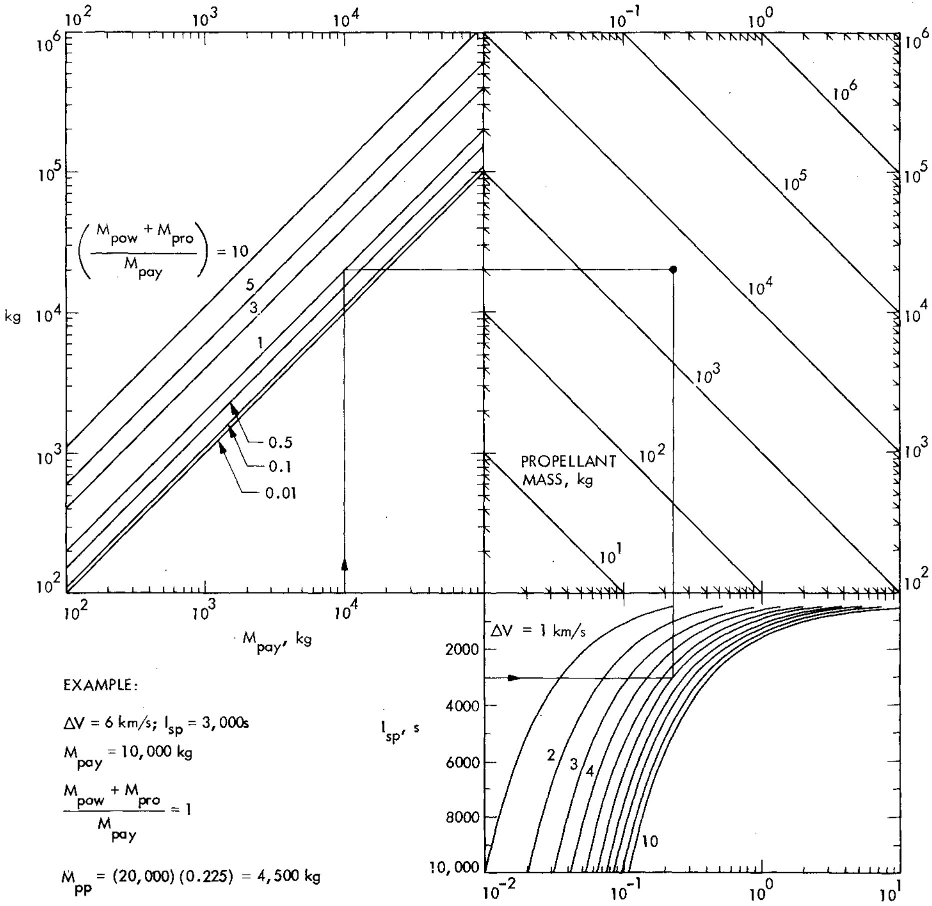

Fig. 2 Nomograph for one way propellant requirement, TF = 0.05.

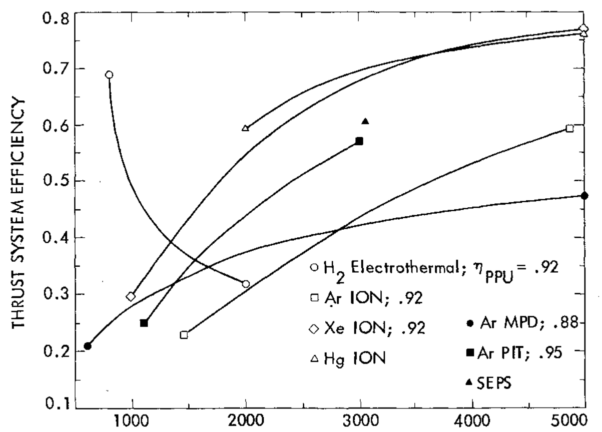

Fig. 3 Thrust system efficiency vs specific impulse.

projections are for ion thrusters using three grids and optimized in size for any specific impulse. The data used for this paper models thrusters from 0.57 to 1.14 m in diameter, consuming slightly less than 2 kW_e at about 1000 s to over 70 kW_e at about 5000 s. The power processing efficiency is estimated from an excellent ion thruster system model presented in Ref. 16. The mercury ion thruster performance was obtained from a projection made in 1980 by NASA Lewis Research Center personnel. The projections are for 30 cm, three grid engines consuming up to about 18 kW_e. The argon MPD thruster efficiency projections[23] are for a set of devices each optimized for a small range of specific impulse. These thrusters are projected to consume about 700-3000 kW_e depending on the specific impulse. The projections of Ref. 23 assume the thruster would operate in a pulsed mode from an

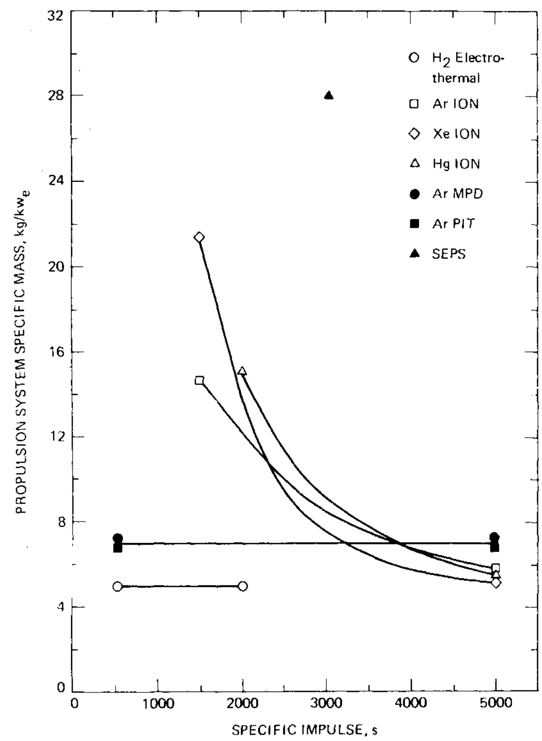

Fig. 4 Thrust system specific mass vs specific impulse.

energy storage/pulsed power supply, if the prime power was significantly below that required for the thruster to operate efficiently. The power processing efficiency also comes from Ref. 23. The argon pulsed inductive thruster performance projection is based upon data for a 1 m device.[24,25] The projection was made by the author based upon potential efficiency improvements coming from using a thyratron switch and better propellant distribution before the discharge. The power processing efficiency is high because the data of Refs. 24 and 25 contain the inefficiency due to the switch and residual energy in the capacitor bank. The efficiency data for a SEPS type engine (30 cm, two-grid, *J* series Hg ion thruster) was taken directly from Ref. 26, which presents the system efficiency for a single detailed system design capable of being throttled from 2000 to 3000 s.

Leaving aside the SEPS *J* series system, which was included only as a reference to "real" system, an attempt was made to use thruster efficiency projections with a similar amount of optimism or required R&D time. The earliest point at which these projections might be realized would be 1985, and perhaps not until 1990.

As described above, the MPD and the advanced ion thruster projections are for an optimized set of thrusters each for a small range of specific impulse, while the pulsed inductive thruster (PIT) projection is for a fixed size, and the electrothermal and SEPS projections are actual data or designs.

Figur 4 presents the specific mass models for the electric propulsion systems examined in this paper. Propulsion system specific mass is defined (for the purposes of this paper) as the total propulsion system mass divided by the total electric power supplied to the system. The propulsion system mass includes thrusters, gimbals, valves, switches, cables, heaters, radiators, primary and auxiliary power processors, system controllers, and integrating structure. Propellant tanks are not included. In all cases the power processors and their radiators and the thrustors dominate the system mass.

For Figure 4, the specific mass of the SEPS system again comes from Ref. 26. The specific mass of the advanced xenon and argon ion thruster systems was calculated using the thruster projection of Ref. 5 and the system mass model of Ref. 16. The specific mass of the advanced 30 cm mercury system came from the earlier Lewis Research Center (LeRC) projections which were based upon Ref. 16. The electrothermal system specific mass was assumed to be 2 kg/kW$_e$ for the power processor, 1.9 kg/kW$_e$ for the power processor radiator, and 1.1 kg/kW$_e$ for the thruster and everything else. These numbers are based, in part on Ref. 4. The MPD system specific mass of 7 kg/kW$_e$ was based upon Ref. 23, which presents a specific mass model for a 1×10^{-3} s pulse length system that is a function of both input power and specific impulse. For power levels above 100 kW$_e$ the variation with power and specific impulse were small and the value of 7 kg/kW$_e$ was judged to be accurate enough for the purposes of this paper. In a similar manner 7 kg/kW$_e$ was judged to be an appropriate specific mass for the PIT based upon Ref. 24, where again the variations of specific mass with power and specific impulse for power levels above about 100 kW$_e$ were negligable for the purposes of this paper.

These projections for specific mass include no explicit assumptions of thruster or other system component lifetime. No redundancy is included for any component.

Throughout this section statements have been made concerning the power level for which the projections of Figs. 3 and 4 are appropriate. These statements can be summarized by saying that all the projections presented in this section are adequate for the purpose of this paper for power levels between 100 and 1000 kW$_e$. The ion thruster system projections are probably good for power levels as low as 10 kW$_e$ for values of specific impulses below about 2500 s.

The projections were analytically modeled. Table 2 presents the models for the system efficiency and system specific mass projections used in the next section.

Performance Based Comparison

This section uses the round trip time as calculated from Eq. (10) as the primary basis for comparing the seven electric propulsion systems whose characteristics were presented in the previous section.

As can be seen from Eq. (10), the trip time is a function of I_{sp}, η, ΔV, TF, α_{pow}, α_{pro} and M_{pay}/P. Using I_{sp} as the independent variable leaves only α_{pow}, M_{pay}/P, and ΔV as parameters, since η and α_{pro} have been modelled as functions of I_{sp} in the previous section. The tankage fraction is set equal to 0.05 for all of the calculations in this section except for the H_2 electrothermal which has been given a tankage fraction of 0.15.

All the trip time results will be presented as a function of I_{sp} with ΔV, α_{pow} and M_{pay}/P as parameters. Figure 5 presents the results for the argon ion system, $\alpha_{pow} = 40$ kg/kW$_e$, $M_{pay}/P = 110$ kg/kW$_e$ and three different ΔV. The reader

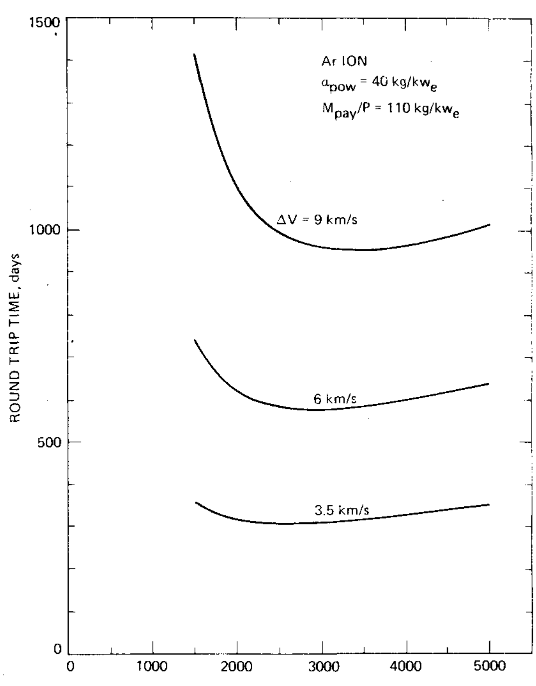

Fig. 5 Argon ion propulsion system trip time vs ΔV.

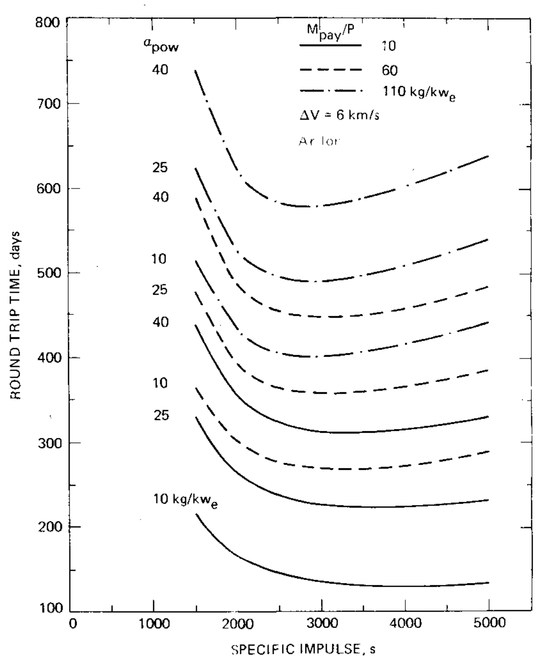

Fig. 6 Argon ion propulsion system trip time vs power supply specific mass and payload to power ratio.

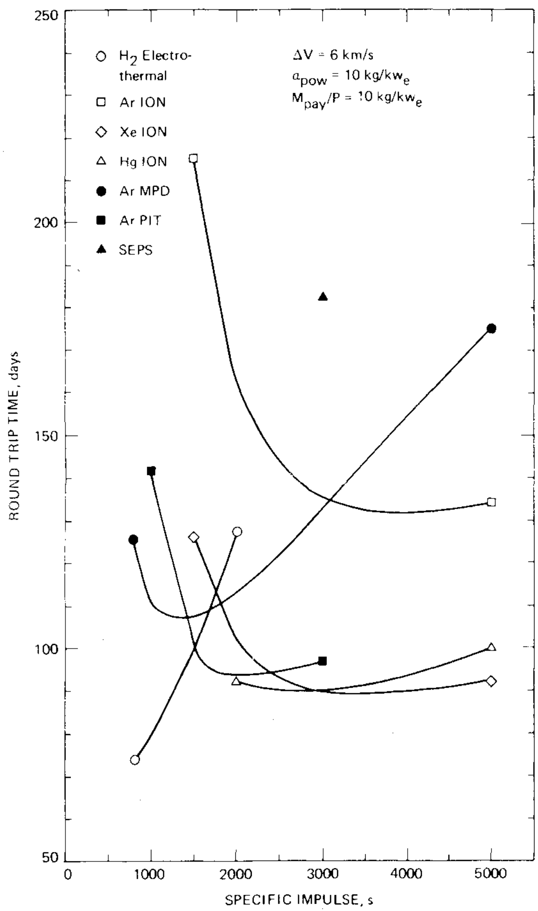

Fig. 7 Propulsion system trip time comparison for $\alpha_{pow} = 10$ kg/kW$_e$ and $M_{pay}/P = 10$ kg/kW$_e$.

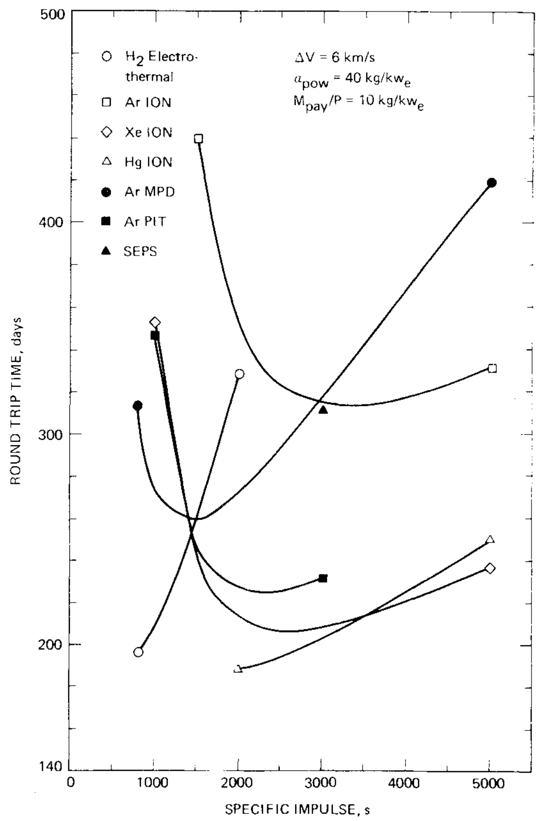

Fig. 8 Propulsion system trip time comparison for $\alpha_{pow} = 40$ kg/kW$_e$ and $M_{pay}/P = 10$ kg/kW$_e$.

may be assured that the argon ion is only a representatative system and the trends to be discussed relative to Figs. 5 and 6 are common for all electric propulsion systems. The results show a minimum round trip time of 579 days at 3000 s for the $\Delta V = 6$ km/s mission. Again, both power and specific impulses are held constant throughout the mission. The payload is dropped off at the end of the first leg and ΔV is equal to 6 km/s for both the up and down leg of the trip. With a payload to power ratio of 110 kg/kW$_e$ and assuming 100 kW$_e$, this gives a 11,000 kg payload. At 3000 s, the argon ion system has a system efficiency of 0.435 and a specific mass of 8.5 kg/kW$_e$. The initial vehicle mass of 21,200 kg is comprised of the 11,000 kg payload, 850 kg of argon ion propulsion system, 4000 kg of power supply, a propellant mass of 5060 kg, and propellant tanks 253 kg. At 5000 s the round trip time is 640 days for the same 100 kW$_e$, 4000 kg power supply and 11,000 kg payload, but the initial mass is only 18,500 kg. The trip time would be the same for a payload of 55,000 kg and a power of 500 kW$_e$. In figures to be presented the payload-to-power ratio is set equal to 10 kg/kW$_e$ which along with 110 kg/kW$_e$ is felt to provide acceptable limits on that parameter. The reader may use Eq. (10) and insert any chosen ratio.

The most critical technology parameter not directly controlled by the electric propulsion technologist is the power supply specific mass. Figure 5 uses 40 kg/kW$_e$ for α_{pow} and later figures will use 10 kg/kW$_e$. An α_{pow} of 40 kg/kW$_e$ is representative of 100 kW$_e$, 7 yr life nuclear reactor power system using a fast-spectrum, highly enriched UO_2 fueled heat pipe reactor and state of the art thermoelectric (Seebeck effect) converters.[27] An α_{pow} of 10 kg/kW$_e$ is an optimistic estimate for high power (~ 250 kW$_e$) reactor/thermoelectric power systems. To achieve an α_{pow} of 10 kg/kW$_e$ with the reactor design of Ref. 27, more efficient thermal to electric converters such as the AMTEC[28] or rotating machinery conversion systems will probably be necessary. Finally, an α_{pow} of 10 kg/kW$_e$ may be very difficult to come by with any of the aforementioned technologies at 100 kW$_e$, but becomes increasingly achievable as the power level increases past about 500 kW$_e$ using rotating machinery or magnetohydrodynamic (MHD) conversion.

Referring again to Fig. 5, one can see that the trip time increases dramatically as ΔV increases, as would be expected. A ΔV of 3.5 km/s is representative of a mission to GEO starting from an altitude of 10,000 km (above most of the radiation belts) and 28.5 deg inclination, while 8 km/s is representative of a LEO to low lunar orbit mission. Figure 6 presents round trip times for the argon ion propulsion system covering the full range of α_{pow} and M_{pay}/P. Higher values of α_{pow} and M_{pay}/P lead to longer trip times. This can be explained by the fact that both α_{pow} and M_{pay}/P are mass-to-power ratios for large portions of the initial vehicle mass. Mass to power ratio can be thought of as mass to thrust ratio, which is the inverse of acceleration. Therefore larger α_{pow} and M_{pay}/P imply lower acceleration and consequently larger trip times.

In both Figs. 5 and 6, there is cleraly an optimum value (least trip time) of specific impulse. The existence of an optimum value of specific impulse, be it a cost or trip time optimum, has been discussed previously.[9,10,12,15,29] The fact that a minimum trip time value of specific impulse exists is not apparent from the form of the trip time Eqs. (10) and (9). However, a minimum trip time specific impulse could have been postulated on the basis that propellant mass and therefore initial vehicle mass, for the constant power problem, steadily decreases as specific impulse increases, while the thrust-to-power characteristic for the electric thruster is maximum at a low value, 1000-3000 s, of specific impulse. The exact value of the minimum trip time specific impulse is a subtle function of all of the input variables (ΔV, α_{pow}, α_{pro}, M_{pay}/P, and η).

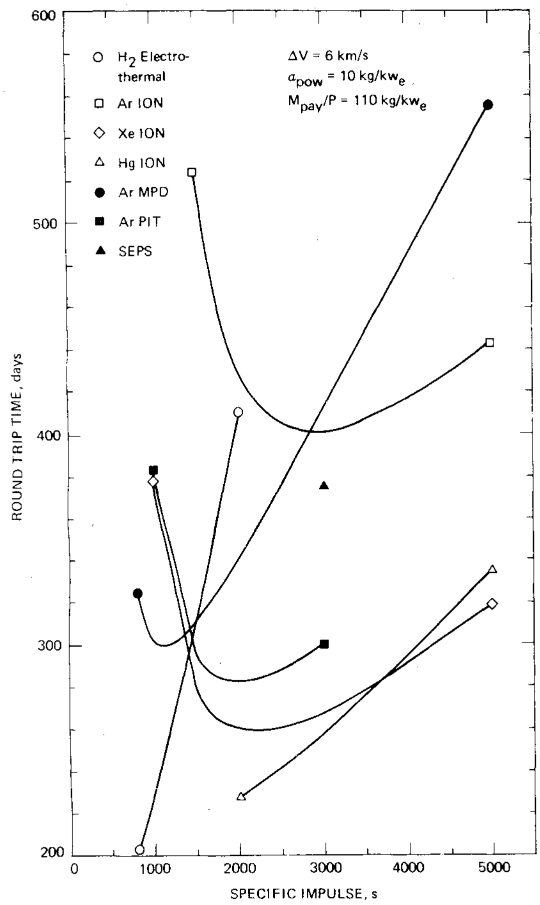

Fig. 9 Propulsion system trip time comparison for $\alpha_{pow} = 10$ kg/kW$_e$ and $M_{pay}/P = 110$ kg/kW$_e$.

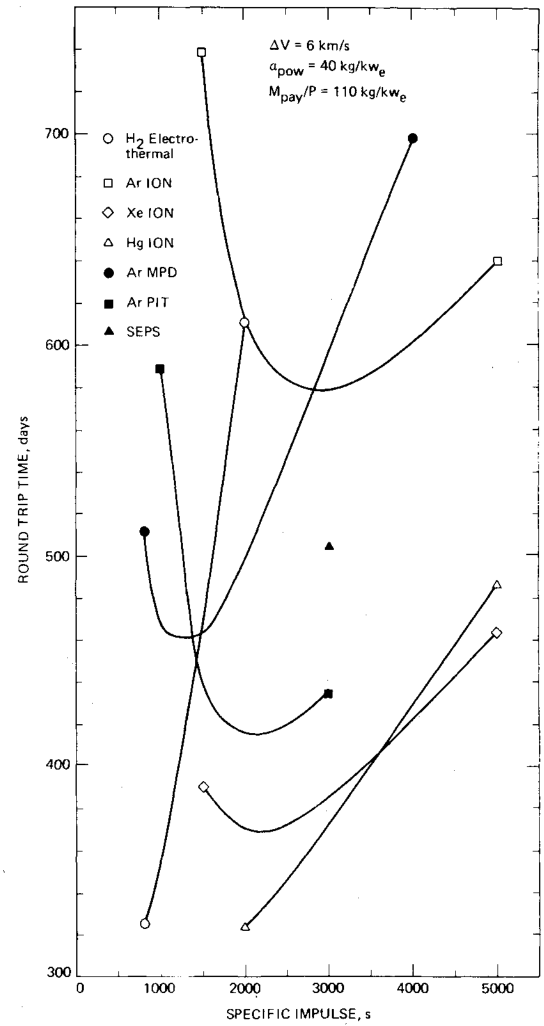

Fig. 10 Propulsion system trip time comparison for $\alpha_{pow} = 40$ kg/kW$_e$ and $M_{pay}/P = 110$ kg/kW$_e$.

Figures 5 and 6 show that for the argon ion thruster system used in this paper, the minimum value of specific impulse lies between 2500 and 4000 s. However, figures yet to be presented for different systems show very different values of the optimum specific impulse.

One can also observe four trends relative to the optimum specific impulse from Figs. 5 and 6. First, the value of the optimum specific impulse is directly related to ΔV, i.e., larger values of ΔV shift the value of the optimum specific impulse to larger values and vice versa. Second, the optimum is less distinct for lower values of ΔV. Third, the optimum value of specific impules moves to lower values as α_{pow} and M_{pay}/P increase. Finally, the optimum is less distinct for lower values of α_{pow} and M_{pay}/P.

Figures 7-10 present the round trip results for a ΔV of 6 km/s using the limiting values of α_{pow} (10 and 40 kg/kW$_e$) and payload-to-power ratio (10 and 110 kg/kW$_e$). The reader should note the trends stated earlier that are common for all systems. The optimum value of specific impulse varies for each thruster system as follows: H_2 electrothermal <800 s, argon ion 3000-4000 s, xenon ion 2000-3500 s, mercury ion <2000-3000 s, argon MPD 1000-1500 s and argon PIT about 2000 s. The reader should keep in mind that all of the results and conclusions concerning specific thruster systems are very dependent upon the performance and system characteristics, Figs. 3 and 4. In general, Figs. 7-10 show that the electrothermal thruster, mercury ion and xenon ion systems have the shortest trip times followed by the argon PIT, argon MPD, and argon ion. This general result is nearly independent of the power supply specific mass or payload-to-power ratio.

Summary and Conclusions

The purpose of this paper was to compare different electric propulsion technologies on the basis of their trip time for an orbit transfer mission. The potentially developable electric propulsion systems considered were argon, xenon, and mercury ion thrusters, argon MPD thrusters, argon pulsed inductive thrusters, and hydrogen resistojet and arcjet thrusters.

The constant power, constant specific impulse unconstrained initial mass orbit transfer problem was chosen as the framework for the propulsion system comparison. The constant power assumption implies a nuclear power supply or a solar array/energy storage power supply. Equations were developed for the one-way and roundtrip time for an orjbit transfer mission. The equations do not include all the finest details, since a highly accurate estimate of trip time was not the primary goal of the paper. The trip time equations do include all the fundamental variables, i.e., ΔV, payload mass, specific mass of the power supply, specified mass of the propulsion system, specific impulse, total system efficiency, and power level. The one way trip time and propellant requirement equations are plotted in nomograph form in order to give one a tool with which to make quick first order trip time estimates and visualize what influence changes to the initial mission or technology parameters have on the trip time. Other bases of comparison besides trip time such as costs, payload capability, technology readiness could have also been used.

Propulsion system specific mass and efficiency projections were made as a function of specific impulse for the six systems being considered. These projections are the key ingredient in the performance comparisons. An attempt was made to use projections with a similar amount of optimism. The projections are only appropriate for systems in post 1985 to post 1990 time frame and for input power levels between about 100 and 1000 kW_e.

The trip-time calculations show that the round trip time increases as all the variables increase, except total system efficiency with which it has an inverse relationship. An optimum (shortest trip time) value of specific impulse does exist but takes on a different value for each system and mission ΔV. As ΔV increases, the optimum specific impulse increases and becomes more distinct. As M_{pay}/P and α_{pow} increase, the optimum specific impulse decreases and becomes more distinct.

As characterized in this paper, the electrothermal thrusters show the best trip time performance at low values of specific impulse, and the mercury and xenon ion thrusters show the best performance at larger values of specific impulse. Although the argon electromagnetic thrusters (PIT and MPD) do not show performance advantages relative to the xenon and mercury ion thrusters, they do exhibit shorter trip times than the argon ion thruster. This would be significant if mercury and xenon propellants were eliminated due to contamination and price considerations.

In conclusion, one should note that there is no single value for the minimum trip time specific impulse. The minimum value is dependent upon the mission and system characteristics. Additionally, it should be kept in mind that the minimum trip time specific impulse may not be the optimum specific impulse for the overall mission. If the minimum trip time specific impulse is too low, the payload ratio will be too small. This will force launch costs high (due to a large propellant mass), and the specific impulse will be driven to larger values.

Besides thruster efficiency, a low specific mass power supply is the most important factor in electric propulsion (as it was 15 years ago[30]). The significance of low specific mass power supplies cannot be over emphasized.

Finally, one should be very careful when interpreting the trip time comparisons. The trip time results presented in this paper include both strengths and weaknesses for all three types of thrusters.

Perhaps the most important product of this work is a technique to compare alternative technologies. Statements regarding the relative worth of technologies can be made with more accuracy for a specified mission. This accuracy will be increased when the technologies are further characterized, i.e., electromagnetic thruster performance should be measured for xenon and mercury propellants where they may show performance improvements compared to argon propellant and ion thruster performance using new approaches to low specific impulse high efficiency operations[5,31] should be demonstrated.

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