JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 30, No. 6, November–December 1993

Mass Savings Domain of Plasma Propulsion for LEO to GEO Transfer

Edgar Y. Choueiri,* Arnold J. Kelly,† and Robert G. Jahn‡
Princeton University, Princeton, New Jersey 08544

A parametric model is used to study the mass savings of plasma propulsion over advanced chemical propulsion for lower-Earth-orbit to geosynchronous-Earth-orbit transfer. Such savings are characterized by stringent requirements of massive payloads [$\mathcal{O}(10)$ metric tons] and high-power levels [$\mathcal{O}(100)$ kW]. Mass savings on the order of the payload mass are possible but at the expense of longer transfer times (8–20 months). Typical of the savings domain is the case of a self-field magnetoplasmadynamic (MPD) thruster running quasisteadily, at an I_s of 2000 s, with 600 kW of input power, raising a 50 metric ton satellite in 270 days. The initial mass at LEO will be 65 ton less than a 155 ton LO_2/LH_2 advanced chemical high thrust spacecraft. An optimum I_s can only be found if the cost savings associated with mass savings are counterbalanced by the cost losses incurred by longer transfer times. A simplistic cost model that illustrates the overall trends in the optimization yielded an optimum I_s of about 2200 s for a cost effective baseline MPD system.

Nomenclature

C	= cost
C_{C_u}	= yearly cost of "lost life"
C_{save}	= cost savings
C_{tt}	= cost loss during transfer time
c_{chem}	= specific cost of chemical propulsion stage
c_{MPD}	= specific cost of plasma propulsion stage
f_t	= tankage mass factor
f_{wear}	= wear factor
g	= acceleration of gravity
I	= yearly interest rate on the cost of transfer time
I_s	= specific impulse
m	= mass
m_{I_s}	= specific impulse dependent mass
$\tilde{m}$	= specific mass
$\dot{m}$	= mass flow rate
P	= power
t_{trans}	= transfer time
u_e	= exhaust velocity
ΔV	= mission delta V
η_t	= thrust efficiency
η_0	= overall efficiency of propulsion system
ψ	= specific cost of MPD propulsion system relative to that of chemical system

Subscripts

f	= final (at GEO)
fix	= fixed
otv	= orbit transfer vehicle
p	= propellant
pay	= payload
pp	= power processing system
ps	= power source
tc	= thermal control system

Received Feb. 23, 1991; revision received Aug. 18, 1992; accepted for publication Sept. 26, 1992.

*Graduate Student; currently Research Associate, Electric Propulsion and Plasma Dynamics Laboratory.

†Senior Research Staff, Electric Propulsion and Plasma Dynamics Laboratory. Member AIAA.

‡Professor of Aerospace Sciences, Electric Propulsion and Plasma Dynamics Laboratory. Fellow AIAA.

I. Introduction

IN anticipation of the eventual availability of high-power electric sources in space, a sizable literature concerning conceptual and detailed mission studies involving plasma propulsion has evolved over the past two decades. Missions, such as lower Earth orbit (LEO) to geosynchronous Earth orbit (GEO) orbit transfers and orbit raising,[1–15] station keeping at GEO,[11,14] manned Mars expedition,[4,5,11] lunar supply shuttles and orbiters,[4,6,16] Neptune orbiter,[6] large structure drag makeup, and asteroid and comet rendezvous,[11] have been proposed and studied with varying degrees of rigor. In these mission studies the self-field magnetoplasmadynamic (MPD) thruster's propulsion capabilities have been compared to those of ion thrusters,[1,6,7,11,14,17] arcjets,[1,2,8,10,11,13,14] and advanced chemical rockets.[1,9,12] Power levels ranging from 50 kW to 10 MW have been considered using both quasisteady and steady-state thrusters.

In the present work we address, through a multiparameter optimization study, the potential mass and cost savings advantages of an MPD propelled orbit transfer vehicle (OTV) over a vehicle propelling the same payload using advanced chemical propulsion. The chosen mission benchmark is that of LEO to GEO orbit raising of a commercial satellite. The goal of the study is to quantitatively define the savings domain of MPD propulsion in this context and illustrate the various dependencies using near-term performance data and technologies.

The mass savings domain is defined as the region of parameter space for which the use of plasma propulsion leads to a substantial reduction in required OTV mass at LEO over that required by advanced chemical propulsion.

There have been few studies that aim at a detailed investigation of this domain and its dependencies.[1,9,12] It is generally known from such studies that more massive payloads and higher power levels tend to bolster the mass savings of the MPD OTV. It is also known that such mass savings are penalized by the much longer transfer times imposed by the relatively low thrust MPD systems.

In the present work, the initial mass of an MPD OTV at LEO required for a transfer to GEO is expressed as a function of key parameters under various assumptions. These assumptions include projected values and dependencies for efficiencies and specific masses of various system components. The MPD-OTV mass is then subtracted from that of an advanced chemical system, and the resulting parametric dependencies of the mass savings are analyzed to define trends and optimizing prescriptions.

In particular we show that the one-sided criterion of minimum transfer time adopted in previous studies[12] (which almost inevitably leads to the choice of a minimum I_s) can be improved on the grounds that the major advantage of MPD propulsion is its mass savings and that the relative changes in the already long transfer times are not so drastic as to restrict I_s to its minimal value. Indeed, we show that for a typical case of the mass savings domain a factor of three increase in mass savings can be achieved by raising I_s from 1000 to 2000 s without increasing the transfer time by more than 10%.

Since in most of the previously mentioned studies, the increase of both the mass savings and the transfer time is found to be monotonic with the (increasing) specific impulse an optimization can only be reached if the penalties imposed by the long transfer times are quantified and made to counterbalance the advantages of mass savings. This can ultimately be achieved through a cost optimization model where both the cost savings engendered by the mass savings and the cost losses incurred by longer transfer times are represented. The last section describes a tentative cost analysis that illustrates one possible method for such a multidimensional optimization problem. In this particular case, the goal is to optimize the specific impulse (for a set payload mass and available power level) for maximum cost savings. Various simplifying assumptions are made along with cost extrapolations from industrial publications.

II. Some Relevant Aspects of Magnetoplasmadynamic Propulsion

Before we proceed with the mass savings study we need to consider some of the relevant aspects of MPD propulsion, such as the mode of operation (steady or pulsed), the choice of propellant, and the scaling of performance with the specific impulse.

Steady-state operation is more advantageous from the point of view of system complexity since it precludes the need for massive power conditioning systems and pulse forming networks, Unfortunately, steady-state self-field MPD thrusters operated at low power suffer from low thrust efficiencies. Measured efficiencies of a self-field device operating steadily in the electromagnetic acceleration regime with argon, at power levels in the 10–30 kW range, never exceeded 10%. The inefficiency of the low-power self-field MPD thruster is attributed to the increasing fraction of the input power consumed by the anode as the power is decreased. In the megawatt-class device this fraction is about 15%,[19] whereas, for the kilowatt device, this fraction is typically as high as 90%.[18]

It is possible, however, to take advantage of the relatively higher performance of megawatt-class thrusters through quasisteady pulsed operation. The dc bus line of a spacecraft power supply can be used to power an energy storage system, like a pulse forming network (PFN), which is discharged at a finite duty cycle. Megawatt-level power is available at short pulses that are long enough for most electromagnetic and plasma phenomena to reach a steady-state level. Excluding heat transfer transients, the transient time scale of all of the relevant processes is usually below 100 μs so that flat 1-ms long pulses can be considered quasisteady.

Quasisteady self-field MPD thrusters were first developed as a laboratory convenience to study high-power acceleration while bypassing the stringent power and vacuum support that steady-state operation entails.[20,21] Until recently, the prospect of using the pulsed MPD thruster for spacecraft propulsion was deemed prohibitive due to the high mass and limited lifetime of the associated power conditioning systems. Recent advances in dielectric film and capacitor technology[22,23] have yielded an order-of-magnitude increase in the energy density of energy storage systems over those available in the early 1970s, and the lifetime problem is all but solved. Cathode erosion which has been the major life-limiting factor for pulsed MPD thrusters has recently been shown in the laboratory to be reduced by more than three orders of magnitude by dispensing low work function metals through the cathode.[24] Thus the quasisteady operation mode of the MPD thruster has recently emerged from its laboratory origins as a real propulsion option. The first space test of an MPD thruster for spacecraft propulsion, scheduled for 1993, will be for a megawatt-class self-field device operated quasisteadily from a 1-kW solar power source.[25-27]

Therefore, we shall assume in our study that the MPD propulsion system operates in the quasisteady pulsed mode. This will allow us to consider power levels as low as 25 kW as well as benefiting from the relatively higher efficiency of megawatt-level operation. It must also be noted that when power levels above 500 kW are considered the option of steady-state operation may become more plausible since tolerable thrust efficiencies can be achieved without a pulse forming network thus saving the mass of the pulsed power conditioning stage. This option, however, is not considered in the present study.

Furthermore, we choose argon for propellant for the sake of continuity with previous studies. The overwhelming majority of studies assumed argon as a propellant since thruster performance with argon is the most thoroughly documented. Most of the studies mentioned used the performance data of the so-called Princeton benchmark thruster operated quasisteadily at megawatt level with argon, published in 1983,[28] as a basis for mission evaluation. The dependence of the thrust efficiency η_t on the specific impulse I_s adopted in this study is that for a pulsed quasisteady self-field MPD thruster operating with argon, as projected by reference[12] for the mid-1990s. This dependence is plotted in Fig. 1.

Finally, we note that the specific masses (mass per unit power) of the various support systems for an MPD OTV are not deterministic parameters at the present time, and the outcome of an analysis can be strongly dependent on them. It has been shown,[7] for instance, that the dependence of the specific mass on the specific impulse must be taken into account in optimization studies. This dependence is still not very well characterized for MPD propulsion and is sometimes totally neglected by some authors. We shall represent this dependence in the formulation of the model in the following with a specific impulse dependent mass m_{I_s} as projected by Ref. 12 and plotted in Fig. 1.

III. Formulation

The propellant mass m_p expended for an incremental velocity ΔV, for a spacecraft with constant rocket exhaust velocity u_e, is[29]

$$m_p = m_f(e^{\Delta V/u_e} - 1) \tag{1}$$

where m_f is the final spacecraft mass. On the other hand, by equating the kinetic power in the exhaust to the available electric power multiplied by the overall efficiency, the propellant mass flow rate $\dot{m}$ of an electric thruster can be written as

$$\dot{m} = 2\eta_0 P/u_e^2 \tag{2}$$

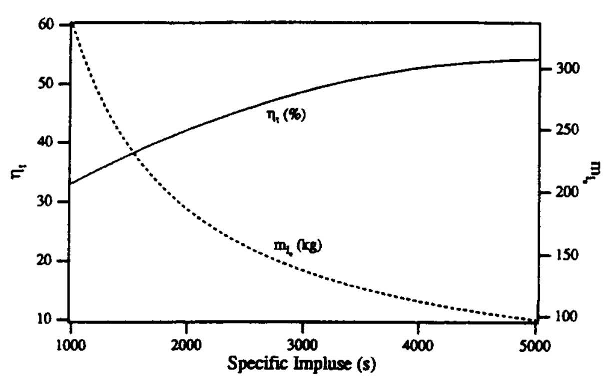

Fig. 1 Thrust efficiency η_t and specific impulse dependent mass m_{I_s} vs the specific impulse as projected by Ref. 12.

where P is the electric power delivered by the power source onboard and η_0 is the overall efficiency of converting this power into thrust power in the exhaust jet. The overall efficiency η_0 is the product of the efficiency of the thruster η_t and that of the power processing apparatus η_{pp}. The latter efficiency was fixed at 88%, as suggested in Ref. 30. The final mass m_f in Eq. (1) is idealized as the sum of the masses of the payload, the empty propellant tank, and the MPD propulsion system. The latter mass can be further broken down into more specific items as shown in Fig. 2 where the associated symbols are also defined. The payload mass m_{pay} will be one of the parameters of the study, and the breakdown of the other masses follows that done in Ref. 31. Specifically, the tankage mass factor f_t is 5%, meaning that m_t is $0.05m_p$, the fixed mass m_{fix} is 43 kg, and the specific impulse dependent mass m_{I_s} is given by the curve in Fig. 1. The specific mass $\bar{m}_{\text{pp}}$ of the power processor is projected at 2 kg/kW and that of the thermal control system $\bar{m}_{\text{tc}}$ at 3 kg/kW. The most crucial specific mass is that of the power source $\bar{m}_{\text{ps}}$, which was estimated as a projected baseline between various power source and energy conversion system options. The energy conversion options are photovoltaic, solar collector/dynamic, fuel cell, turbogenerator, thermoelectric, dynamic Rankine, dynamic Brayton, thermoionic, and magnetohydrodynamic. An extrapolation from current technologies[31] has led to specific masses ranging from 10 to 40 kg/kW. A baseline for this analysis was drawn at 20 kg/kW.

Using the mass breakdown described along with Eq. (1), the final mass at GEO can be expressed as

$$m_f = \frac{m_{I_s} + (\bar{m}_{\text{pp}} + \bar{m}_{\text{tc}} + \bar{m}_{\text{ps}})P + m_{\text{fix}} + m_{\text{pay}}}{1 - f_t(e^{\Delta V/u_e} - 1)} \tag{3}$$

and the total initial mass of the OTV at LEO is given by the well-known rocket equation

$$m_{\text{otv}} = m_f e^{\Delta V/u_e} \tag{4}$$

For a continuous thrust trajectory, the transfer time for a certain ΔV increment can be obtained by dividing Eq. (1) by Eq. (2) and using Eq. (3) to get

$$t_{\text{trans}} = \frac{m_p}{\dot{m}} = \frac{(I_s g)^2[m_{I_s} + (\bar{m}_{\text{pp}} + \bar{m}_{\text{tc}} + \bar{m}_{\text{ps}})P + m_{\text{fix}} + m_{\text{pay}}](e^{\Delta V/gI_s} - 1)}{2\eta_{\text{pp}}\eta_t P[1 - f_t(e^{-\Delta V/gI_s} - 1)]} \tag{5}$$

The ΔV requirement for a low-thrust spiral trajectory from LEO to GEO is a function of the initial acceleration (initial thrust to weight ratio). For the payload masses (5–50 metric

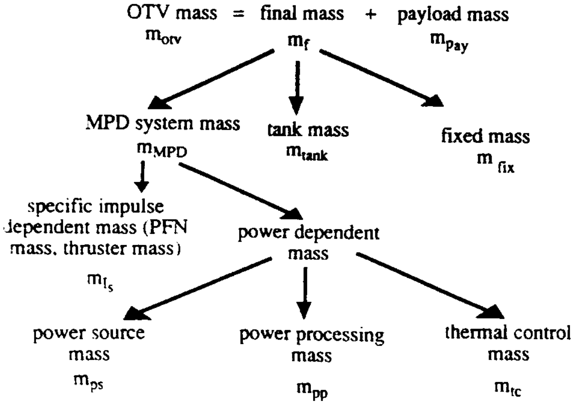

Fig. 2 MPD-OTV mass breakdown.

Table 1 Initial mass at LEO of LO_2/LH_2 OTV after Ref. 31

Payload mass, kg	Initial mass of LO_2/LH_2 OTV, kg
5,000	20,000
10,000	35,000
20,000	65,000
50,000	155,000

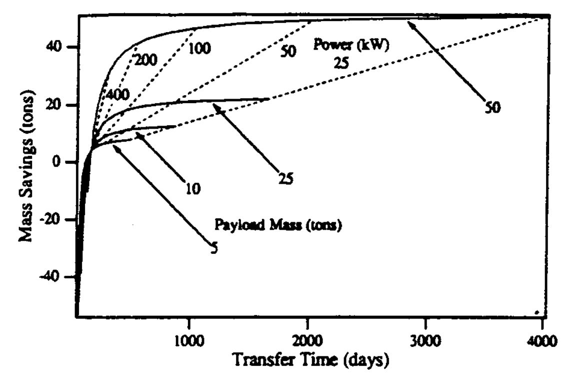

Fig. 3 Mass savings using an MPD OTV at I_s = 1000 s and low-power levels.

ton) and power levels (50 kW–1 MW) considered in this parametric study, the initial thrust to weight ratios are low enough so that ΔV can be approximated by its asymptotic value at zero initial acceleration. This ΔV was determined by a low-thrust spacecraft dynamics study of reference[32] and is about 6600 m/s for the LEO to GEO mission.

In Eq. (5), m_{I_s} and η_t are both functions of I_s as plotted in Fig. 1. All of the other constants have been fixed except for I_s, m_{pay}, and P which are left as parameters. When power levels higher than 500 kW are available for the propulsion system, the MPD thruster can be operated efficiently in a steady-state mode. Steady-state operation can substantially enhance the mass savings capabilities of the MPD OTV as the various power conditioning and pulsed forming networks of the pulsed system are no longer needed. However, only pulsed MPD thrusters are considered in the present study.

For fixed power level and payload mass, the transfer time was shown in Ref. 12 to be slightly increasing with the specific impulse. It was concluded in that reference that an "optimum" specific impulse is that for which the transfer time is minimum. This one-sided criterion trivially leads to an optimum I_s equal to the lowest attainable specific impulse which, for the MPD thruster operated with argon, is not more than 1000 s. It will be shown that a factor of three increase in mass savings can be achieved by raising I_s from 1000 to 2000 s without increasing the transfer time by more than 10%. Therefore, the choice of an optimum I_s should be made taking both transfer time and mass savings into account.

Ultimately, an optimization should rest on the compromise between the cost savings associated with mass savings and the cost losses incurred by longer transfer times. Such an optimization is attempted in Sec. V.

IV. Parametric Study of Magnetoplasmadynamic Orbit Transfer Vehicle Mass Savings over Chemical Orbit Transfer Vehicle

Estimates of mass savings are obtained by subtracting the initial mass at LEO of an MPD OTV from that of an OTV propelled by advanced high-thrust chemical rockets. Table 1 shows the initial mass at LEO of an OTV propelled by an LO_2/LH_2 rocket as a function of payload mass, as estimated in Ref. 31. First, the specific impulse is fixed at 1000 s whereas m_{pay} and P are left to vary as parameters. Later, the role of the

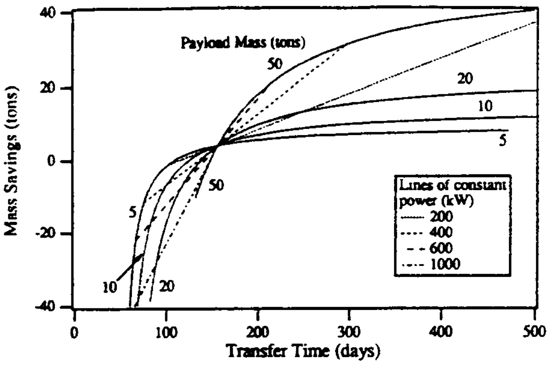

Fig. 4 Mass savings using an MPD OTV at I_s = 1000 s and high-power levels.

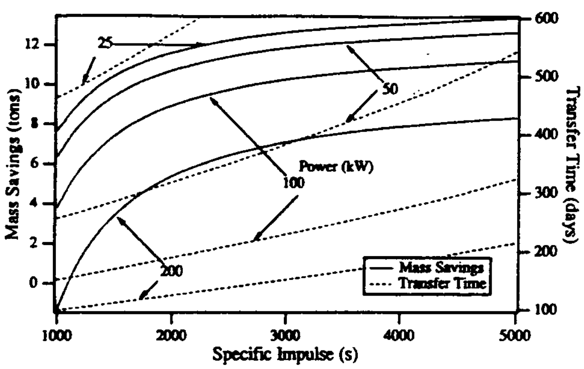

Fig. 5 Mass savings using an MPD OTV for a 5-ton payload.

specific impulses is studied by fixing the payload mass at 50 metric tons.

A. Mass Savings at I_s = 1000 s

The resulting mass savings are calculated and plotted in Fig. 3 as a function or transfer time where the straight dotted lines represent lines of constant power and negative mass savings imply that the MPD OTV is not competitive. In this figure the low to moderate power range was considered with 25 kW ≤ P ≤ 400 kW. First, it is evident that greater mass savings are conditioned by more massive payloads. Mass savings exceeding 40 metric ton are attainable with 50-ton payload. Transfer times, however, are excessively long at these power levels and are typically a few years. For a fixed payload mass, both mass savings and transfer time increase with decreasing power. The asymptotic behavior of the curves is such that decreasing the power beyond a certain level (e.g., 200 kW for a 50-ton payload) would incur strong penalties on the transfer time with little further gains in the form of mass savings. Of course, for the case of a commercial satellite, these penalties and gains must be weighed in terms of cost as will be later attempted in the cost analysis of Sec. V.

The case of higher power levels (400 kW ≤ P ≤ 1 MW) is illustrated in the plot of Fig. 4 which shows more tolerable transfer times. A typical case from that figure is that of a 50-ton payload propelled to GEO in 7 month by a 600-kW MPD OTV weighing 20 tons less than the corresponding 155-ton LO_2/LH_2 OTV. It is also evident, from the same figure, that for a fixed payload mass there exists a power level above which the MPD OTV becomes noncompetitive due to negative mass savings. Moreover, this maximum power level increases with payload mass. For the 50-ton payload case considered the maximum power level is little higher than 1 MW.

B. Mass Savings as a Function of the Specific Impulse

To study the parametric role of the specific impulse, the payload mass was fixed, and the specific impulse was varied between 1000 and 5000 s. The plots in Figs. 5 and 6 show the results for the cases of 5- and 50-ton payloads, respectively, which in turn correspond to the cases of low- and high-power ranges. These plots demonstrate that a judicious increase in specific impulse can be quite advantageous for mass savings without large transfer time penalties. For instance, if the case of the 50-ton payload propelled by a 600-kW MPD OTV is reconsidered, an increase in the specific impulse from 1000 to 2000 s yields a factor of three increase in mass savings (from 20 to 60 ton) although the transfer time only increases from 7 to 9 month. The capability of increasing mass savings without large transfer time penalties, by raising the specific impulse, deteriorates with decreasing available power.

V. Specific Impulse Optimization for Maximum Cost Savings

Ultimately, for a commercial satellite, any optimization of the specific impulse should be based on the compromise between mass savings cost gains and transfer time cost losses. The simplistic model discussed here aims at illustrating, albeit qualitatively, the issues conditioning such an optimization. To make the cost model more tractable, the following assumptions were made:

1) The MPD and LO_2/LH_2 propulsion technologies are both well developed.

2) A one-way trip to GEO is the specified mission.

3) The OTV is placed in LEO using the Space Shuttle, and the cost figures for the required Shuttle trips are included in the quoted system specific costs.

4) Mission operation costs at LEO are assumed to be the same for both OTVs and, therefore, cancel out from the cost savings analysis.

5) Prelaunch system readying time costs are neglected (although typical chemical cryogenic stages of the kind considered here take more than 2 months of vertical processing and launch integration according to Ref. 31).

The idealization of the cost behavior is easier for heavy payloads since the assumption of constant OTV specific cost (cost per kilogram of payload) for the LO_2/LH_2 OTV becomes better for more massive payloads.[10] For a 50-ton payload the specific cost of the LO_2/LH_2 OTV is about 17,000 $/kg. From Table 1 the OTV initial mass for the chemically propelled 50-ton payload is 155 tons, so that the specific cost of the chemical propulsion stage c_{chem} can be roughly set at 8100 $/kg.

A model for the cost savings C_{save} can be written as

$$C_{save} = c_{chem}(m_{i_{chem}} - m_{pay}) - c_{MPD}(m_{i_{MPD}} - m_{pay}) - C_{tt} \quad (6)$$

where $m_{i_{chem}}$ and $m_{i_{MPD}}$ are the initial masses for the chemical and MPD OTVs obtained from Table 1 and Eq. (4), respectively, c_{MPD} is the specific cost of the MPD propulsion stage; and C_{tt} represents the cost losses incurred by the long transfer time of the MPD OTV. The transfer time cost losses of the

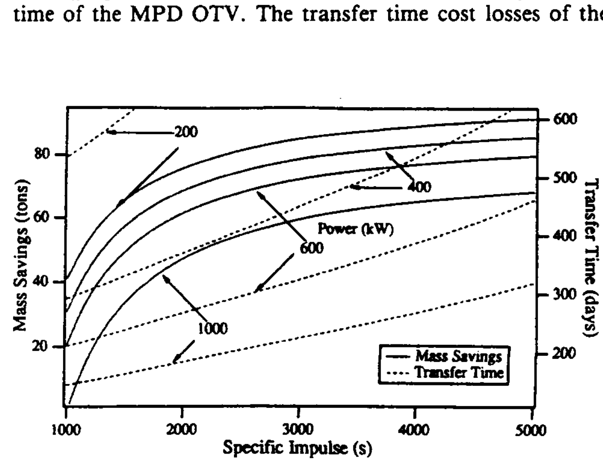

Fig. 6 Mass savings using an MPD OTV for a 50-ton payload.

chemical OTV are negligible compared to those of the MPD OTV. A model for C_{tt} can be written as

$$C_{tt} = t_{trans}\frac{(C_m + C_{ll})}{365}\left(1+\frac{I}{365}\right)^{t_{trans}} \tag{7}$$

where t_{trans} is the transfer time in days obtained from Eq. (5) and C_m is the yearly cost for monitoring the slowly ascending OTV. The monitoring can be kept at a minimum cost by assigning the operation control to a routinely run multimission control system. An estimate for C_m is made by adding the salaries of two full-time control personnel to a 100-k$ margin for emergency monitoring requesting dedicated radar and staff service (C_m = 160 k$). The parameter C_{ll} in Eq. (7) represents the yearly cost of "lost life" of the slowly ascending inactive satellite which can be expressed as $C_{ll} = f_{wear} \times$ commercial yearly income, where f_{wear} is a wear factor larger than unity accounting for the losses due to thermal stresses the satellite endures during its slow trajectory in and out of the Earth shadow, as well as other degrading effects due to the long exposure to the LEO environment. Assuming that the 50-ton satellite is an important commercial satellite, the commercial yearly income is set at 200 M$, and f_{wear} is conservatively set at 3. The term $[1 + (i/365)]^{t_{trans}}$ in Eq. (7) represents the "cost of money" invested in the ascent to GEO. The yearly interest rate on the cost of the transfer time I in this term is set at 10%. Additional costs that may result from the higher insurance premium of slow transfer maneuvers were neglected.

The only unspecified term in the Eq. (7) model is the specific cost of the MPD propulsion stage C_{MPD}. Any estimate would be highly uncertain. Since the value of C_{MPD} is the most unknown item of the cost model, it is left as a varying parameter. This is done by introducing the nondimensional parameter $\psi \equiv C_{MPD}/c_{chem}$, which represents the specific cost of the MPD propulsion stage relative to that of the chemical one. The final cost savings model becomes

$$C_{save} = c_{chem}[(m_{i_{chem}} - m_{pay}) - \psi(m_{i_{MPD}} - m_{pay})] - t_{trans}\frac{(C_m + C_{ll})}{365}\left(1+\frac{I}{365}\right)^{t_{trans}} \tag{8}$$

The cost savings for a 50-ton payload lift to GEO at an available power of 600 kW is plotted in Fig. 7. The parameter ψ was varied between 0.25 and 2 to represent a range of possible relative cost of MPD propulsion technology. For each value of ψ in that figure there exists an optimum I_s for which the cost savings are maximum. The optimum specific impulse decreases with decreasing ψ. It can also be concluded from the same figure that for this particular mission and under the assumptions of the model, the MPD OTV becomes cost competitive only when the specific cost of an MPD propulsion stage is at most as expensive as that of an advanced chemical stage. For ψ = 1, for instance, the optimum specific impulse is at 2200 s corresponding to about 30 M$ of savings.

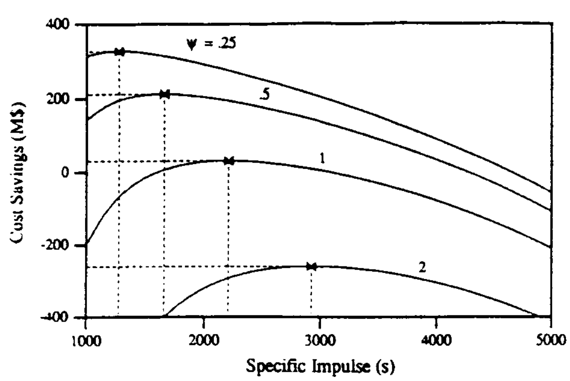

Fig. 7 Cost savings using an MPD OTV for a 50-ton satellite lift to GEO with 600 kW of power; optimal specific impulse marked on each curve.

VI. Conclusions and Final Remarks

The initial mass of an MPD OTV at LEO required for a transfer to GEO was modeled as a function of key parameters using projected values and dependencies for the efficiencies and specific masses of the related systems. This initial mass was then subtracted from that of an OTV with advanced (LO_2/LH_2) chemical propulsion.

Even though the results were based on an optimistic projection of high-power MPD propulsion performance (η = 40% at I_s = 2000 s), the domain in which pulsed self-field MPD systems become competitive for LEO to GEO transfers is still characterized by massive payloads [$\mathcal{O}(10)$ metric ton] and high-power levels [$\mathcal{O}(100)$ kW] well above those foreseen in the near future. Mass savings on the order of the payload mass can be achieved but under a penalty of 8- to 20-month-long transfer times. A parametric sensitivity analysis also showed that the specific impulse of the MPD system should not be kept to a minimum as was proposed in previous studies aiming to keep the transfer time penalty to a minimum. Indeed, a judicious increase in the specific impulse was shown to substantially benefit mass savings by a factor of three with a relatively small increase in the required transfer time.

Only by quantifying the penalties imposed by such long transfer times in terms of cost it becomes possible to estimate an optimum specific impulse for maximum cost savings. An example of how such an optimization can be done within the parameter space of interest yielded an optimum specific impulse of 2200 s.

The need for propelling heavy payloads, the establishment of expandable space stations, the possibility of remote energy transfer by laser or microwave beaming, the realization of reusable propulsion stage designs, the possibility of efficient steady-state MPD thruster operation, and the rapid evolution of related technologies would shed a more optimistic light on the use of MPD propulsion for LEO to GEO transfers.

Acknowledgments

This work was supported by NASA under Contract NASA-954997, by grants from Rocket Research, Inc. and DOE/Princeton Plasma Physics Laboratory, and by the Air Force Office of Scientific Research Grant AFOSR-91-0162.

References

[1]Birkan, M., and Micci, M. M., "Survey of Electric Propulsion Applicability to Near Earth Space Missions," International Electric Propulsion Conference (IEPC) Paper 88-65, Oct. 1988.

[2]Lyon, W. F., "KREM: A Model for Evaluating Orbital Transfer Vehicles Utilizing Nuclear Electric Propulsion," IEPC Paper 88-124, Oct. 1988.

[3]Coomes, E. P., King, D. Q., Cuta, J. M., and Webb, B. J., "The Pegasus Drive: A Multi-Megawatt Nuclear Electric Propulsion System," AIAA Paper 87-1038, May 1987.

[4]King, D. Q., and Sercel, J. C., "A Review of the Multi-Megawatt MPD Thruster and Current Mission Applications," AIAA Paper 86-1437, June 1986.

[5]King, D. Q., "Steady-State Operation and Power Requirements of the Multi-Megawatt MPD Thruster," *1985 IEEE International Conference on Plasma Science*, IEEE Catalog 85CH2199-8, June 1985, p. 20.

[6]Sercel, J., and Krauthamer, S., "Multimegawatt Nuclear Electric Propulsion; First Order System Design and Performance Evaluation," AIAA Paper 86-1202, June 1986.

[7]Auweter-Kurtz, M., Kurtz, H. L., and Schrade, H. O., "Optimization of Propulsion Systems for Orbital Transfer with Separate Power Supplies Considering Variable Thrust Efficiencies," AIAA Paper 85-1245, July 1985.

[8]Selph, C., and Perkins, D., "An Analysis of Electromagnetic Thrusters for Orbit Raising," IEPC Paper 84-80, July 1984.

[9]Rudolph, L. K., "Design and Benefits of Pulsed MPD Thruster Orbit Transfer Vehicles," IEPC Paper 84-81, July 1984.

[10]Rudolph, L. K., and King, D. Q., "Electromagnetic Thrusters for

Spacecraft Prime Propulsion," AIAA Paper 84-1446, June 1984.

[11]Vondra, R., "A Review of Electric Propulsion and Mission Applications," IEPC Paper 84-82, July 1984.

[12]Rudolph, L. K., and Hamlyn, K. M., "A Comparison between Advanced Chemical and MPD Propulsion for Geocentric Missions," AIAA Paper 83-1391, June 1983.

[13]Jones, R. M., "A Comparison of Potential Propulsion Systems for Orbit Transfer," AIAA Paper 82-1871, Nov. 1982.

[14]Kaufman, H. R., and Robinson, R. S., "Electric Thruster Performance for Orbit Raising and Maneuvering," AIAA Paper 82-1247, June 1982.

[15]Dailey, C. L., and Loveberg, R. H., "Shuttle to GEO Propulsion Trade-Offs," AIAA Paper 82-1245, June 1982.

[16]Toki, K., Shimizu, Y., and Kuriki, K., "Application of MPD Thruster System to Interplanetary Missions," AIAA Paper 85-2026, Sept. 1985.

[17]Ray, P. K., "Characterization of Advanced Electric Propulsion Systems," AIAA Paper 82-1246, June 1982.

[18]Myers, R. M., "Energy Deposition in Low Power Coaxial Plasma Thrusters," Ph.D. Thesis, Mechanical and Aerospace Engineering Dept., Princeton Univ., Princeton, NJ, June 1989.

[19]Saber, A. J., "Anode Power in a Quasi-Steady MPD Thruster," Ph.D. Thesis, Mechanical and Aerospace Engineering Dept., Princeton Univ., Princeton, NJ, May 1974.

[20]Clark, K. E., "Quasi-Steady Plasma Acceleration," Ph.D. Thesis, Mechanical and Aerospace Engineering Dept., Princeton Univ., Princeton, NJ, May 1969.

[21]Clark, K. E., and Jahn, R. G., "Quasi-Steady Plasma Acceleration," *AIAA Journal*, Vol. 8, No. 2, 1970, pp. 216–220.

[22]Dehlefsen, R., and Ennis, J., "Capacitors for Pulsed Electric Propulsion," AIAA Paper 88-125, Oct. 1988.

[23]Kuriki, K., Harada, H., Gohnai, T., Yoshida, T., Ijichi, K., and Obara, H., "Metalized Plastic Film Capacitor for MPD Thruster," IEPC Paper 84-27, July 1984.

[24]Chamberlain, F. R., Kelly, A. J., and Jahn, R. G., "Electropositive Surface Layer MPD Thruster Cathodes," AIAA Paper 89-2706, July 1989.

[25]Kuriki, K., Toki, K., and Shimizu, Y., "Advanced Technology Experiment Onboard Space Flyer Unit (SFU)," Inst. of Space and Astronautical Science, Electric Propulsion Experiment (EPEX) Tech. Rept. EXP-R-i003-0, Tokyo, May 1987.

[26]Kunii, Y., Moriai, T., Sasaki, H., Okamura, T., Harada, H., Ijichi, K., and Kuriki, K., "Development of the Electrical Power Subsystem for the Electric Propulsion Experiment Onboard the Space Flyer Unit (SFU)," AIAA Paper 87-1040, May 1987.

[27]Kunii, Y., Moriai, T., Okamura, T., Yoshida, T., Ijichi, K., and Kuriki, K., "Development of Control and Monitor Subsystem for the Electric Propulsion Experiment Onboard the Space Flyer Unit (SFU)," AIAA Paper 87-1041, May 1987.

[28]Burton, R. L., Clark, K. E., and Jahn, R. G., "Measured Performance of a Multi-Megawatt MPD Thruster," *Journal of Spacecraft and Rockets*, Vol. 20, No. 3, 1983, pp. 299–304.

[29]Jahn, R. G., *Physics of Electric Propulsion*, McGraw-Hill, New York, 1966, pp. 2–11.

[30]King, D. Q., and Rudolph, L. K., "100 kWe MPD Thruster System Design," AIAA Paper 82-1897, Nov. 1982.

[31]Rudolph, L. K., Hamlyn, K. M., Ogg, G. M., Davis, H. P., and Stump, W., "MPD Thruster Definition Study," Martin Marietta Denver Aerospace for the Air Force Rocket Propulsion Lab., Tech. Rept. AFRPL TR-84-046, Denver, CO, June 1984.

[32]Burt, C., "The Dynamics of Low-Thrust Spacecraft Maneuvers," *The Aeronautical Journal of the Royal Aeronautical Society*, Vol. 72, Nov. 1968, pp. 925–940.

Antoni K. Jakubowski
Associate Editor