Projected Lunar Cargo Capabilities of High-Power VASIMR^{TM} Propulsion

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Abstract: A lunar cargo architecture based on solar-powered VASIMR^{TM} plasma propulsion is considered. Performance in terms of the mass of cargo delivered to the lunar surface is presented as a function of specific impulse. A principal advantage of the VASIMR^{TM} over other electric propulsion technologies for this application is its use of abundant and inexpensive argon as propellant. While it is generally believed that solar electric propulsion offers significant economic advantages over chemical propulsion to a large-scale lunar exploration program, the cost of solar photovoltaic power will be a critical factor in achieving real cost savings. Solar electric power cost will strongly affect the choice of thruster technology and optimal specific impulse.

Nomenclature

α	=	specific mass, kg/kW
CDV	=	Cargo Delivery Vehicle
DL	=	descent/lander
IMLEO	=	initial mass in low Earth orbit
Isp	=	specific impulse
LEO	=	low Earth orbit
LLO	=	low lunar orbit
LOI	=	lunar orbit insertion
OTV	=	Orbital Transfer Vehicle
PV	=	photovoltaic
SEP	=	solar-electric propulsion
SLA	=	Stretched Lens Array
SLASR	=	Stretched Lens Array Square Rigger
TL	=	trans-lunar
VASIMR	=	VAriable Specific Impulse Magnetoplasma Rocket

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I. Introduction

A NUMBER of nations – the US, China, Russia, India and the states of the European Community – have expressed the intention in recent years of embarking on long-term programs of human exploration of the Moon. Such programs would involve the transfer of large quantities of material from the Earth to the Moon's surface. In addition to piloted vehicles, such programs would also require a cargo capability to transport the infrastructure and ongoing shipments of supplies needed to support human activity on the Moon.

While chemical rockets could provide the needed cargo capability, mission analysts have realized for decades that the high specific impulse afforded by electric propulsion offers greater lunar cargo capability. A 1966 study conducted for NASA by General Electric[1] examined the capabilities of both nuclear and solar-powered electric lunar cargo systems employing ion engines. At that time, it was assumed that space nuclear powerplants would be able to achieve specific masses of 3 – 10 kg/kW, while space photovoltaic power would achieve only 10 – 20 kg/kW. Largely for this reason, the 1966 study concluded that nuclear power was a better choice for lunar cargo than solar. In the intervening years, development of space nuclear power stalled while space photovoltaic array performance reached 3 kg/kW, as exemplified by the Stretched Lens Array Square Rigger (SLASR) system developed by Entech and ATK[2]. Recent progress suggests that the SLASR technology may reach 2 kg/kW within a few years. Our proposed lunar cargo system therefore baselines the SLASR technology as its power source.

A lunar cargo system based on the xenon Hall thruster was described in 2005 by R. Spores[3]. This study included consideration of the effect of specific impulse from 2500 to 3500 seconds, the range within reach of contemporary Hall thruster technology. The present study considers the effect of specific impulse over a very wide range, from 3,000 up to 20,000 seconds, since the VASIMR™ system has demonstrated specific impulse capability up to 12,000 seconds[4] and is believed to be capable of good performance at much higher values. The VASIMR™ system considered here also assumes argon propellant. Recent experimental work[5] conducted by Ad Astra indicates that an argon-based VASIMR™ engine will have satisfactory efficiency (~65%) at power levels above 100 kW and specific impulse above 4,000 seconds. The abundance and low cost of argon - typically 1/20th that of xenon - is an important factor to consider in evaluating the long-term sustainability of in-space propulsion technologies.

II. Proposed Architecture

The architecture of the system considered here is illustrated in Fig. 1. Implementation of the system begins with the launch of the Orbital Transfer Vehicle (OTV). This consists of a cluster of VASIMR™ engines, a solar array, and a relatively small propellant tank needed for the return flight from the Moon (without cargo). Mass of the OTV is modeled as being directly proportional to power. The array is assumed to be based on the present SLASR technology with a specific mass of 3 kg/kW. A detailed point design study (unpublished) conducted by Andrew Petro at NASA Johnson Space Center with input from Oak Ridge National Laboratory estimated the specific mass of a megawatt-class VASIMR™ engine to be approximately 1.3 kg/kW. To account for the remaining OTV hardware, the present study assumed an overall specific mass for the OTV of 6 kg/kW. To examine the sensitivity of the OTV specific mass on the system performance, selected calculations were repeated assuming 12 kg/kW.

The Cargo Delivery Vehicle (CDV) is the assembly consisting of the cargo, the descent lander (DL) that takes the cargo from low lunar orbit down to the surface, a tank containing sufficient argon propellant for the OTV's outbound and return flights, and structure holding these components together. The mass of the CDV is set at 100 mT, approximately the projected capability of the heavy lift launch vehicle NASA is planning to develop to support its return to the Moon. 5 mT of this total is allocated for structure and other hardware, leaving 95 mT for cargo, the DL, and the argon propellant plus its tank (tankage fraction was set at 0.12). In the Spores study[3], the ratio of DL mass to cargo mass was set at 0.961. If LOX/LH2 propulsion with a specific impulse of 450 seconds is assumed for the DL and 2000 m/s is allocated for landing from a 100 km LLO, it would seem that this ratio could be as low as 0.81. However, for margin and ease of comparison with Spores' study, we also set this ratio to be 0.961.

The operational scenario is as follows. The OTV is launched into low Earth orbit. This is followed by launch of the CDV, which then makes rendezvous and docks with the OTV. At this point, a small portion of the propellant in the main tank launched with the CDV is transferred to a tank within the OTV so that the OTV has propellant for the return flight. The docked assembly then spirals up from low Earth orbit, makes the transition to a lunar capture trajectory, and then spirals down to the 100 km altitude low lunar orbit. At this point, the OTV separates from the CDV and begins the return flight to low Earth orbit. The cargo and DL separate from the CDV and make the descent to the surface. The remainder of the CDV – the 5 mT of structure and the now-empty argon propellant tank – is abandoned in low lunar orbit.

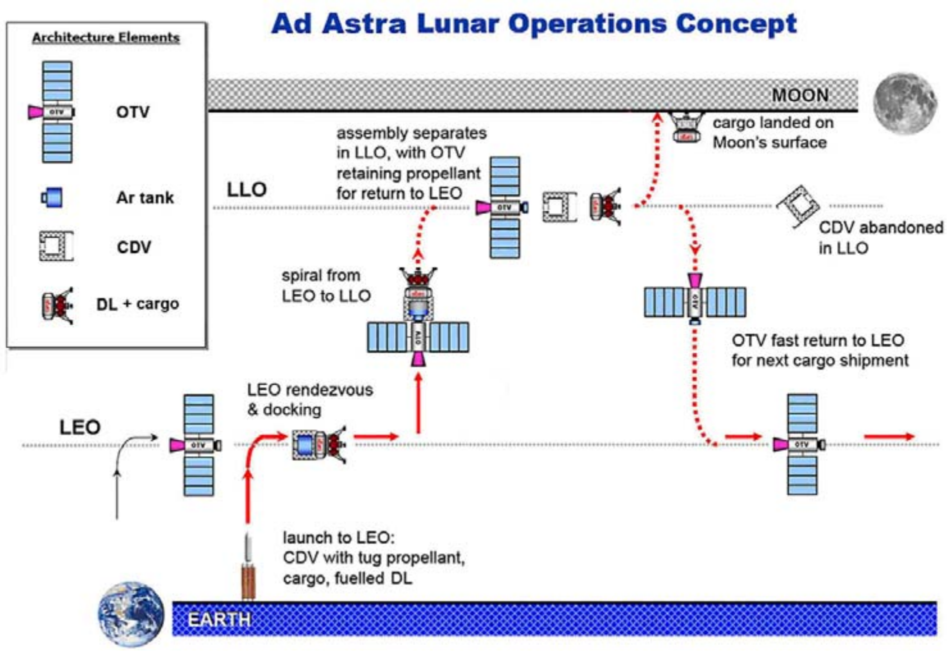

Figure 1. Architecture of Lunar Cargo System. The solar-powered VASIMR™ OTV (orbital transfer vehicle) is launched to low Earth orbit, followed by launch of a CDV (cargo delivery vehicle) for each round trip to low lunar orbit and back.

III. Comparison Chemical System

For comparison with the performance of the electric system, the cargo that could be delivered to the lunar surface was calculated for a 450 s specific impulse all-chemical delivery system. This scenario is illustrated in Fig. 2. Delta-v's were rounded for simplicity of presentation, but fall within actual ranges, based on numbers from the Apollo program. Using the same performance for the LOX/LH2 descent lander as in the solar electric system, the all-chemical system delivers 18 mT of cargo to the lunar surface, of the 100 mT initial mass lofted into low Earth orbit.

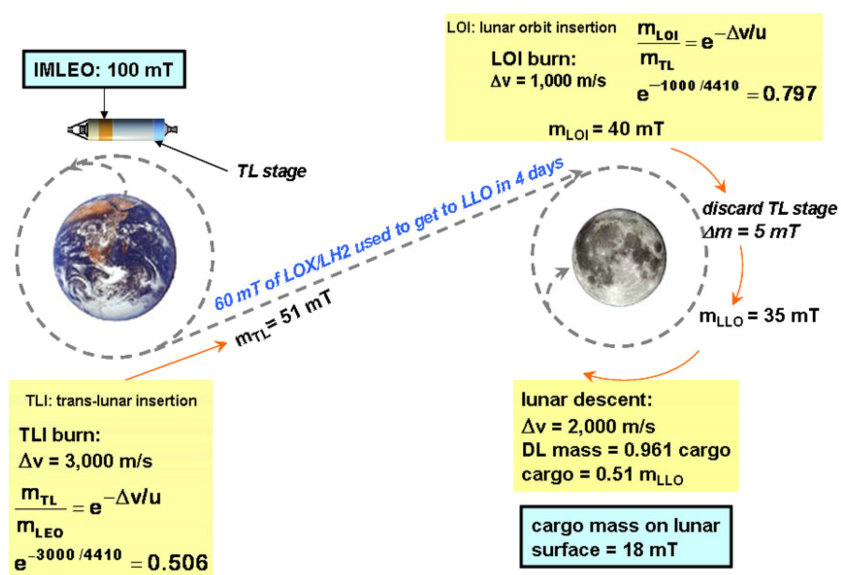

Figure 2: 450 s Isp all-chemical lunar cargo system.

IV. Performance Calculations

Efficiency of the thruster was set at 65% for all values of Isp. This was done so that the effects of transit time, specific mass and Isp on performance (cargo mass) could be determined independent of thruster technology.

Delta-v's for the spiral trajectories of the SEP system were calculated using Edelbaum's equation (approximately 8,000 m/s total from 550 km altitude circular LEO to 100 km altitude circular LLO). For a given Isp, initial mass in LEO was calculated from the rocket equation. Required power was found by calculating the total kinetic energy imparted to the propellant mass and dividing by the transit time. Successive approximation was used to find a consistent solution for a given value of α, subject to the constraints of the 100 mT IMLEO of the CDV. The mass available for cargo and the descent lander was found by subtracting the following quantities from the 100 mT IMLEO: the propellant mass required for a given Isp and α, the mass allocated for tankage, and 5 mT for CDV structure.

Robert Vondra, an aerospace consultant under contract to Ad Astra, performed the calculations supporting this study. Andrew Ilin (Ad Astra) performed a number of trajectory calculations by integrating the equations of motion for confirmation of Vondra's results.

V. Results

Figure 3 illustrates the effect of transit time on both the performance, which is the cargo mass delivered to the Moon's surface (solid lines), and what is essentially the cost for this performance, the mass of the OTV (dashed lines). For comparison, a horizontal line at 18 mT is drawn to show the performance of the all-chemical system illustrated in Fig. 2. The dashed line at 48.4 mT is the "ultimate limit", the mass available for cargo if zero propellant were used to effect the LEO-LLO transfer. This limit is a reflection of the mass ratio between the cargo and its lander (*1:0.961*), which in turn is limited by the Isp of the lander's chemical propulsion. Detailed results for the 180 day transit are presented in Table 1. At the lowest Isp considered – 3,000 seconds – the performance of the solar electric system, 32.8 mT, is nearly double that of the all-chemical system's delivery of 18 mT. At this Isp, we have 68% of the maximum cargo possible. Raising the Isp to 5,000 s brings us to 37.6 mT, or 78% of the maximum. We achieve this improvement at the expense of additional power. Referring to Table 1, the 3,000 s system requires 1090 kW while the 5,000 s case requires 2010 kW. This additional power levy could be relieved by increasing the transit time from six months to ten (lowest curve in Fig. 3). The main conclusions to be drawn from Fig. 3 are: 1) for an OTV with a specific mass of 6 kg/kW, there are significant cargo gains to be had at Isp's above 3,000 seconds but this comes at the expense of higher power, and 2) while increasing the transit time past four months does little to improve performance, it rapidly relieves the power requirement and OTV mass.

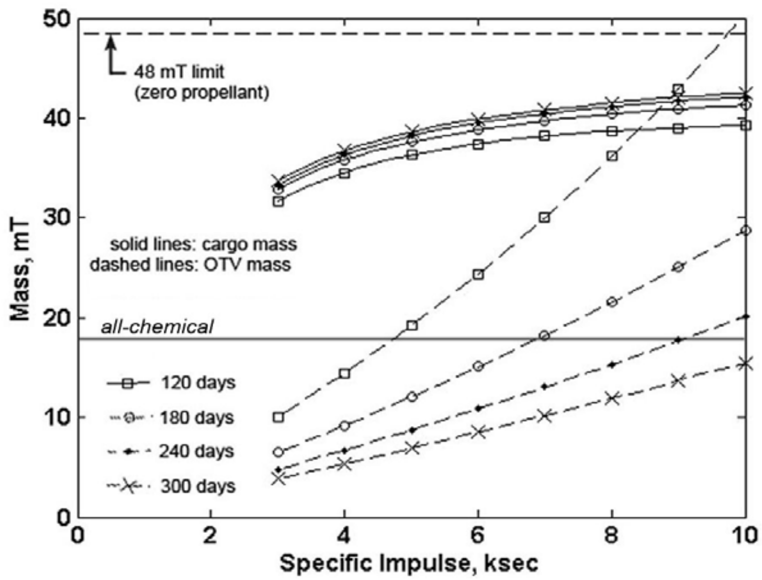

Figure 3: Effect of transit time on cargo mass and mass of the OTV (for α = 6 kg/kW).

Table 1. 180 day transit; OTV α = 6 kg/kW

180 day transit		Mcargo	IMLEO	Fuel Mass	Fuel Mass	Fuel Mass	Return
		(mT)	(mT)	(mT)	Out (mT)	Rtn (mT)	Time
Isp (s)	P (kW)						(days)
3000	1090	32.8	107	27.4	25.4	2.04	14.5
4000	1530	35.8	108	22.2	20.1	2.08	18.6
5000	2010	37.6	112	19	16.9	2.13	22.8
6000	2510	38.8	115	16.8	14.6	2.19	26.9
7000	3030	39.7	118	15.2	13	2.25	31.1
8000	3590	40.4	122	14.1	11.8	2.31	35.3
9000	4180	40.9	125	13.2	10.8	2.38	39.5
10000	4800	41.3	129	12.5	10.1	2.45	43.7

Figure 4 illustrates the effect of specific mass on performance for a fixed transit time of 180 days. As Isp increases, the mass of the OTV is driven up. Past a certain Isp, the propellant mass begins increasing rather than decreasing with Isp, forcing the cargo mass down. In addition, as specific mass increases, the maximum cargo value drops, and occurs at lower Isp. Higher specific mass not only raises the OTV mass, but it reduces the maximum performance of the system.

VI. Economics

While the benefits of a solar electric lunar cargo system relative to an all-chemical system are clear in terms of performance, a compelling case would have to be made on economic grounds for the implementation of such a system. Some insight can be obtained by considering the recurring cost of propellant, and amortization effects on the cost of space photovoltaic power; however myriad factors would influence the cost of such a system and at this time, it is by no means clear which technology would be the best choice.

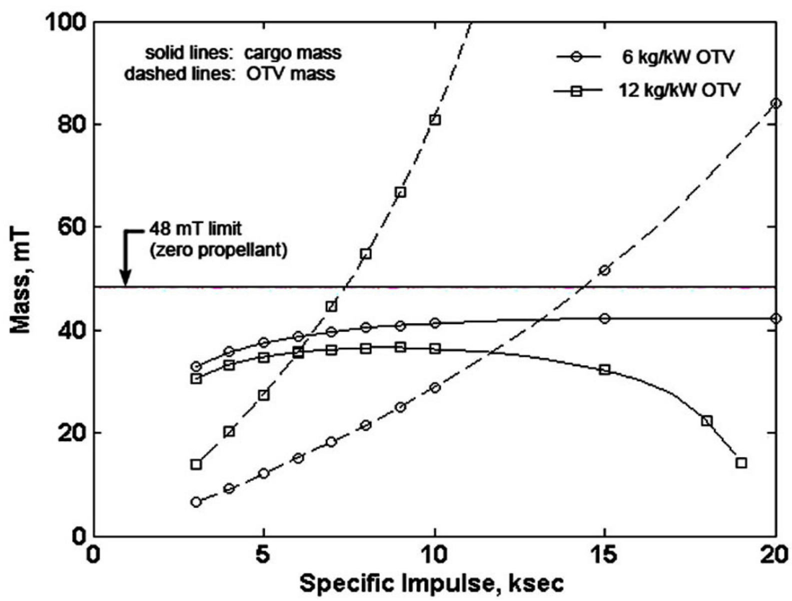

Figure 4: Effect of specific mass (α) on cargo system performance. Transit time is 180 days.

Two fundamental costs are propellant and power. To calculate the cost per roundtrip, we divide the cost of the photovoltaic power system by the number of roundtrips possible over the lifetime of the array, and add to it the cost of propellant for a single roundtrip. In Fig. 5, we have calculated this cost as a function of the number of roundtrips for two potential thruster technologies, based on the numbers in Table 1. The Xenon-based thruster is assumed to operate at an Isp of 3,000 s and a power level of 1.1 MW, and consumes 27.4 mT of Xenon per roundtrip. The cost per trip is plotted for two different Xenon prices, $1000/kg and $2000/kg; estimates of this cost span approximately this range. The cost per trip is plotted for a single Argon price, since it is a common commodity whose price fluctuates little. Furthermore, its price is so low that it is not a significant cost driver. Of much greater influence is the cost per watt of space photovoltaic power. Estimates for this price are typically in the range of $300 - $500 per watt for the relatively small-scale systems currently employed. The plot on the left in Fig.5 uses the midpoint of this range, $400/watt. The plot on the right assumes that economy of scale might result in a cost for the megawatt systems under consideration of $100/watt.

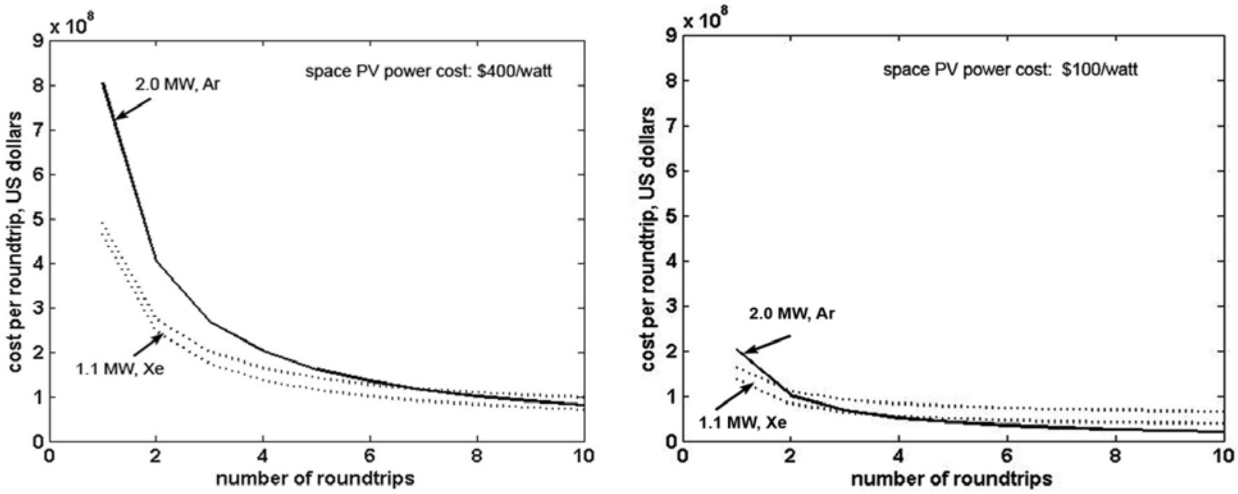

Figure 5: Cost dependence of lunar cargo technologies on propellant cost, power cost and system lifetime.

Far more detailed studies are needed to make meaningful cost comparisons. The plots in Fig. 5 are presented only to illustrate the most basic trends of two cost drivers, power and propellant.

VII. Conclusions

Based on the calculations presented here, it is clear that low OTV specific mass is crucial to obtaining the potentially high lunar cargo capability of the VASIMR^{TM} technology. Given that low specific mass can be achieved, the selection of a lunar cargo propulsion technology and the Isp at which it should operate will be strongly affected by the cost of space photovoltaic power. Provided that the development of megawatt space power systems results in economy of scale, higher specific impulse systems will be favored.

References

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