TECHNOLOGY REQUIREMENTS FOR HIGH-POWER LITHIUM LORENTZ FORCE ACCELERATORS

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Abstract

Lithium Lorentz Force Accelerators (LFA's) are capable of processing very high power levels and are therefore applicable to a wide range of challenging missions. An analysis of a reusable orbit transfer vehicle with a solar or nuclear electric power source was performed to assess the applicability of high-power LFA's to this mission and to define engine performance and lifetime goals to help guide the technology development program. For this class of missions, the emphasis must be on achieving high efficiency at an Isp of 4000–5000 s at power levels of 200–250 kWe. The engines must demonstrate very reliable operation for a service life of about 3000 hours. These goals appear to be achievable with engine technologies currently under development.

INTRODUCTION

Lorentz force accelerators are the only type of electric thruster with a demonstrated capability to process steady state power levels up to several MWe in a relatively compact device. In these engines a very high current is driven between coaxial electrodes through an alkali metal vapor or gaseous propellant. The current interacts with a self-induced or externally-generated magnetic field to produce an electromagnetic body force on the gas. LFA's can operate efficiently at power levels from 150 kWe up to tens of MWe and are therefore ideally suited for a variety of future missions requiring high power levels. This paper outlines the current status of lithium LFA thruster technology and discusses the use of LFA's in reusable orbit transfer vehicles. A parametric study of the impact that propulsion system performance and life has on mission performance was used to define engine technology goals.

LORENTZ FORCE ACCELERATOR TECHNOLOGY STATUS

The current focus of LFA technology feasibility assessment is on applied-field, lithium-fuelled engines operating at 100–150 kWe. At these power levels the discharge current is not high enough to generate significant self magnetic fields, so an external field generated by a solenoid is used. This field induces azimuthal currents which interact with the radial and axial magnetic field components to accelerate the plasma. Lithium propellant yields very high engine efficiency because it has low frozen flow losses. Because it has a very low first ionization potential and a high second ionization potential, very little power is expended in creating the plasma. Figure 1 shows a schematic of an engine being developed by the Moscow Aviation Institute (MAI) under JPL sponsorship. This design is based on over 30 years of experience in testing and modeling high-power lithium thrusters at MAI.

The electrode geometry is designed to balance engine performance considerations with lifetime concerns. The tungsten anode is designed to be radiatively-cooled for this range of power levels, and operates at a temperature of 2000 K or less. It is profiled to follow the magnetic field lines, which results in a more uniform distribution of current density. If the anode is not properly contoured, excessive heating of the downstream edge can lead to melting. In general, thrust and efficiency increase with increasing current and magnetic field. However, when the current exceeds a critical level at a given mass flow rate, depletion of charge carriers in the anode region by the radial $\mathbf{j} \times \mathbf{B}$ force and increased magnetization of the electrons leads to an increasing anode fall voltage and instabilities which can cause anode spot formation and localized melting.

A ratio of anode-to-cathode radius of 3.55 was chosen in this initial design to permit stable operation for specific impulses up to 4500 s.

The cathode is composed of a bundle of tungsten rods enclosed in a tungsten tube. An integral heater consuming about 3.5 kWe vaporizes the lithium propellant which then flows through the channels between rods into the discharge. These channels act as small hollow cathodes which, for the proper choice of mass flow rate and current, will very efficiently ionize the lithium. Multichannel hollow cathodes such as these offer more emitting area than comparably sized rod or single channel hollow cathodes. In addition, the attachment is more stable, the operating voltage is lower than large single channel cathodes and there is a higher probability of recapturing material evaporated from the emitting surfaces. The cathode is sized so the current density does not exceed 200 A/cm^2 of cross-sectional area (a total current of 3200 A) and operates at a temperature of about 3000 K. When the cathode is preheated to about 1300 K, the discharge will ignite reliably when voltages as low as 40 V are applied. The main insulator between the cathode and anode in the

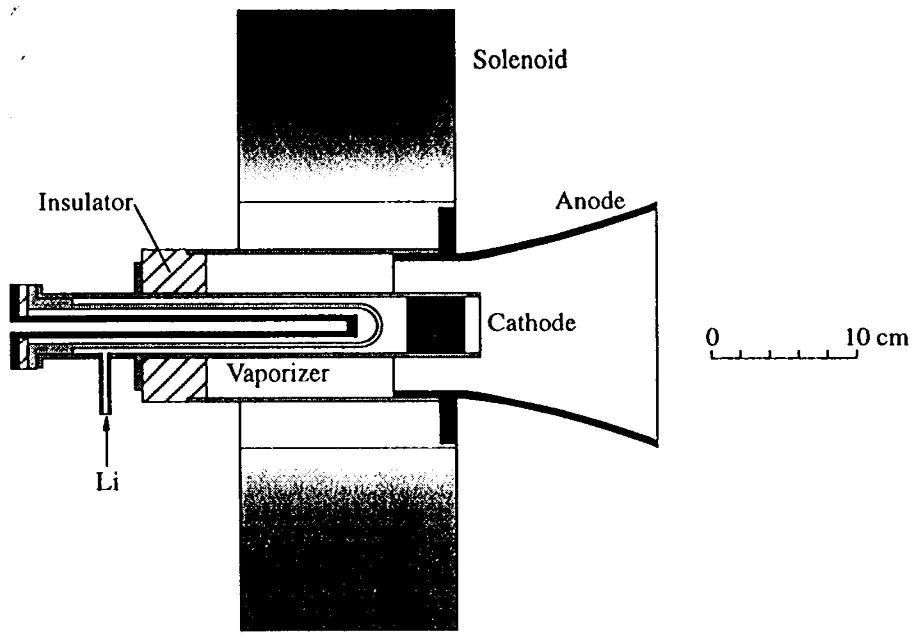

FIGURE 1: 150-200 kWe LFA Developed at the Moscow Aviation Institute.

laboratory-model thruster is aluminum-boron nitride, which provides sufficient life for short-term tests. Hot insulators can react with the lithium vapor, so the engine is designed to maintain a relatively low insulator temperature and isolate it from the lithium plasma. The solenoid in this engine is a water-cooled copper coil which consumes about 4.5 kWe. The electrodes, insulator and cathode heater/vaporizor have a mass of 17 kg.

This engine has been operated at up to 130 kWe and has demonstrated 44 percent efficiency at an Isp of 3500 s. In relatively short duration tests performed so far, the primary life-limiting component appears to be the cathode. With proper design and operating conditions the anode and insulator are not subject to significant wear. The cathode, however, must operate at high temperatures to emit electrons and erodes by evaporation. Erosion rates in terms of mass loss per unit charge transfer as low as 0.1–1 ng/C appear to be achievable if the current density is less than 200 A/cm^2 and the cathode is operating in the hollow cathode mode. At 3200 A, this yields a mass loss rate of 1–10 g/khr, so operation for up to several thousand hours appears feasible. Although this engine has not been run for more than a few hours at a time, a self-field thruster with a similar cathode design was run for 500 hours at a power level of 500 kWe. The current research program is focussed on increasing the engine power level to 150 kWe and demonstrating 45 percent efficiency at 4500 s Isp. Cathode erosion measurements in this engine and in longer duration tests (tens of hours) with a 30 kWe-class engine are being combined with modeling of erosion processes and cathode thermal behavior to assess the potential for long service life.

TECHNOLOGY REQUIREMENTS FOR ORBIT-RAISING MISSIONS

One relatively near-term potential application of high-power electric thrusters is raising large payloads from low Earth orbit (LEO) to geosynchronous Earth orbit (GEO). An analysis of a reusable orbit transfer vehicle with a solar or nuclear electric power source was performed to assess the applicability of high-power LFA's to this mission and to define engine performance and lifetime goals to help guide the technology development program. The Titan IV expendable launch vehicle with a Centaur upper stage is capable of placing a 4540 kg payload in GEO for a cost of $400–450 million. The concept studied in this analysis is to deliver a similar sized payload (less than 4950 kg), launched to LEO at a cost of $40–50 million with a Delta 7920, to GEO with a reusable transfer stage. This electric upper stage then returns to LEO and is resupplied with a propulsion module containing new engines, fuel and, in the case of the solar electric vehicle, new solar arrays. It is then ready to lift another Titan IV/Centaur-class payload to GEO. A starting orbit of 370 km is assumed for both the solar electric and the nuclear electric cases. The nuclear electric vehicle has a chemical propulsion system which serves as the attitude control system and also has sufficient fuel to raise the vehicle to a nuclear-safe orbit of 700 km in the event of a reactor problem.

The two conceptual vehicle designs are consistent with relatively near-term power and propulsion technologies. The baseline solar electric propulsion (SEP) vehicle has two solar arrays providing a total of 200 kWe. The solar arrays are assumed to have a specific mass of 10 kg/kWe (excluding the cabling, which is treated separately). Two modules are used; for a nominal width of 27.2 m and a sunlight-to-electricity conversion efficiency of 21%, each 100 kWe panel has an unfolded length of 13.6 m. No specific solar array technology is assumed, although the specific mass is typical of advanced APSA-type arrays. Several array technologies could be used to meet both the specific mass and packaging requirements, including APSA, inflatable or concentrator arrays.

The engines are mounted on a 41 m boom perpendicular to the solar array wings. This places the edge of the solar array 30 m from the engines to minimize the potential for contamination from the lithium propellant. The payload is also mounted on the end of the boom furthest from the engines. The propulsion module, consisting of the attitude control system, the lithium propellant tank, engines and a plume shield, is designed to dock with the vehicle because it must be replaced for each mission.

The lithium is launched as a solid in a thin stainless steel tank approximately 2 m in diameter. Once on orbit, the lithium is melted with electric heaters and fed to the vaporizors in the engines. The propellant tankage mass is assumed to be 2.8% of the propellant mass, based on a detailed analysis of a Mars cargo vehicle using lithium LFA's (Frisbee, 1993). Each engine is assumed to be capable of processing all of the available power at a nominal efficiency of 50% and an Isp of 4500 s. The engine lifetime is assumed to be 5000 hours, so a total of three engines is required to complete the baseline mission with one spare. An engine electrode mass equal to that of the MAI thruster is assumed, but the water-cooled solenoid is replaced with a copper Bitter solenoid which would be self-radiating at about 700 K, consume 4.5 kWe and weigh about 60 kg. This yields a thruster specific mass of 0.35 kg/kWe. A thin, deployable plume shield is included to limit the backflow of lithium vapor. Preliminary analyses suggest that plume shields are quite effective at mitigating the contamination potential of high power lithium thrusters (Frisbee 1993).

The solar arrays are configured to match the engine terminal characteristics (60 V and 3200 A discharge current for the baseline case), so power processing is required only for the vaporizor and the electromagnet. The switching scheme is shown in Fig. 2. Commercially available switches rated at 500 A continuous, 3800 A on opening and 2000 A on closing are mounted in parallel with series load-balancing resistors and used to switch the load. Non-load break switches are used to switch between engines with the control switches open and the individual lines are fused for over-current protection. The cathodes are grounded to avoid the need for isolators in the lithium propellant lines and the anode leads are switched. This direct-drive approach yields an efficiency of 99.5% with negligible mass. In the SEP vehicle, however, the cabling required for low

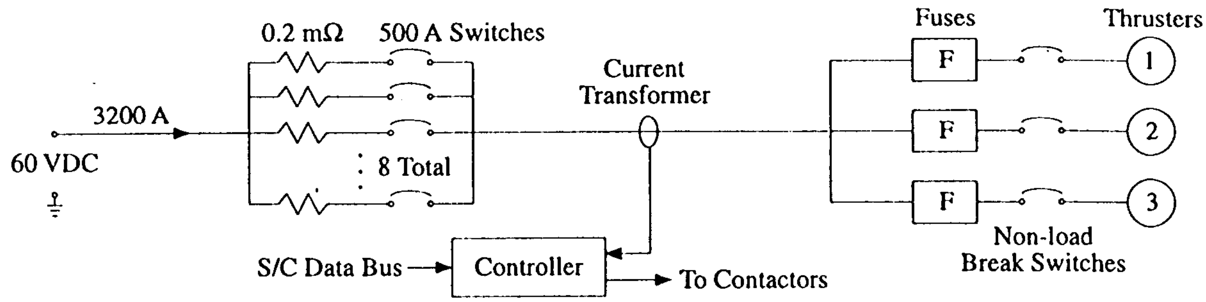

FIGURE 2: Schematic of the Direct-Drive Switching.

voltage DC power transmission is a significant mass. The cable mass for the arrays and boom was scaled from a detailed 1.5 MWe SEP Mars cargo vehicle study (Frisbee, 1996), yielding a specific mass of 6.46 kg/kWe. Copper, aluminum and lithium were evaluated as potential conductor materials. Aluminum was chosen because it gives the best combination of strength and low resistance per unit mass. Aluminum tubes used for the power transmission also form the main structural elements of the solar array and boom, partially offsetting the mass penalty. The voltage drop in the cables results in a transmission efficiency of 90.8%. The total specific mass, including the switches, cabling and small power processing units for the electromagnet and vaporizor, is 6.75 kg/kWe with a total efficiency of 89.6%. The losses associated with the solenoid and the vaporizor are included in the engine efficiency.

The attitude control system uses a chemical bipropellant engine with an Isp of 300 s and a tankage fraction of 16% for a mission ΔV of 10 m/s. The design also includes 1000 kg for structure and contingency.

The nuclear electric propulsion (NEP) vehicle uses SP100 reactor technology with thermoelectric conversion, which has a specific mass of 31 kg/kWe at the nominal 200 kWe power level. The payload is located at the end of a 30 m-long boom behind the power system shield to minimize the impact of heat and radiation from the reactor and radiator. The propulsion module is located on the unshielded side of the reactor to minimize cable lengths. The materials in the engine are compatible with this radiation environment and heat radiated from the reactor may be used to help melt the lithium fuel. The propulsion module in this vehicle also has a plume shield, but the high operating temperatures of most reactor and radiator surfaces will also help prevent bulk lithium condensation. The only concern is the low temperature radiators associated with the multiplexers, which are located near the reactor in the SP100 design. It may be necessary to relocate these radiators and couple them to the multiplexers with a pumped coolant loop to prevent lithium condensation. The thermoelectric elements are configured to match the engine terminal characteristics so the same direct drive scheme illustrated in Fig. 2 can be used. An estimate of the cabling requirements along with the switches and power processing units for the magnet and vaporizor yields an overall specific mass of 0.67 kg/kWe and an efficiency of 98.2%. The chemical propulsion system includes additional fuel for the 181 m/s ΔV required to raise the vehicle to a 700 km nuclear-safe orbit if the reactor or electric propulsion system fails.

These vehicle and system characteristics were used to calculate mission performance in terms of initial mass in LEO (IMLEO) and resupply mass (replacement propulsion module and, in the case of the SEP vehicle, new solar arrays) as a function of round trip time. Parametric studies of the effect of total power level, power system specific mass α, engine efficiency η, Isp and lifetime were performed to help focus technology development efforts. Figure 3 shows the IMLEO including the vehicle and a 4540 kg payload and the resupply mass versus trip time for power levels ranging from 100 to 250 kWe. All other propulsion system parameters were fixed at the nominal values with the exception of the nuclear power supply α, which decreases with increasing power (Mondt, 1996). At the nominal 200 kWe level the SEP vehicle delivers the payload to

GEO and returns for another payload in 305 days while the NEP vehicle takes just over one year. At this power level either vehicle could be launched with the first 4540 kg payload on a Titan IV for \$250–300 M. Resupplying the vehicles for subsequent trips could be accomplished with a single Delta 7920 launch at \$40–50 M. The SEP vehicle has a lower IMLEO and a shorter trip time because of the lower specific mass, but the mass savings are not enough to enable the use of a smaller launch vehicle. The NEP vehicle resupply mass increases with power because of the additional propellant required for the heavier vehicle. The SEP resupply mass includes propellant and new arrays, and for power levels above about 150 kWe the resupply mass of the SEP system exceeds that of the NEP vehicle. There is a large penalty in trip time for power levels lower than the nominal value, and a marginal increase in IMLEO for higher powered systems. At power levels above about 225 kWe the NEP vehicle can no longer be launched with a single Titan and the SEP vehicle resupply mass exceeds the capability of the Delta.

The impact of the power source technology, encapsulated in the parameter α, is shown in Fig. 4. At a comparable specific mass, the SEP system has a longer trip time because of the 10% losses in the cabling (which reduces the jet power and, therefore, thrust) and the 15% time penalty associated with periods in the Earth's shadow. The IMLEO increases with α because of increased propellant requirements as well as increased power system mass. The NEP vehicle resupply mass increases with α, again because more fuel is required to lift the heavier reactor to GEO and return it to LEO. The SEP resupply mass increases more rapidly with α because of the solar array mass. If the NEP system specific mass exceeds about 33 kg/kWe the vehicle plus payload cannot be launched on a Titan IV and if the SEP system α is over 11 kg/kWe resupply with a Delta becomes impossible. Advances in technology which reduce the power supply mass can have a significant effect on trip time.

The effect of engine performance on mission performance is summarized in Figures 5 and 6. A lower Isp results in higher thrust for fixed power, which has a relatively large impact on trip time. The propellant mass decreases with increasing Isp, but this has a relatively small impact on IMLEO and resupply mass because the propellant is only about 20% of the total mass. The minimum Isp which still allows a Titan launch for the nominal NEP system is 4000 s, and if the SEP system operates at less than 4000 s resupply with a Delta is not possible. The engine efficiency has essentially no effect on system mass for fixed power; higher efficiency just results in greater jet power which reduces trip time. The trip time required to make this approach economically feasible depends on the revenue lost during transit compared to the gains in launch vehicle cost. Round trip times less than one year, however, require an engine efficiency greater than 42% for the SEP vehicle and 52% for the nuclear-powered spacecraft. The engine lifetime was varied from 3000 to 6000 hours, but had very little effect on mission performance because the engines are so light. The shorter lifetimes required the addition of one engine with a mass increase of only a few hundred kg and a time penalty of a few days. System complexity and packaging may be more critical drivers of engine life, although the engines are quite compact, the liquid feed systems are relatively simple and the direct coupling to the arrays or thermoelectric elements considerably simplifies the power management system.

The parametric studies suggest the following directions for technology development. Minimization of the trip time requires the highest power level possible, but launch vehicle constraints limit the power supplies to less than about 225 kWe with the near-term baseline technologies. Improvements which reduce α would permit higher power levels. The nuclear system has the advantage of longer life than the solar arrays, which reduces the resupply mass and complexity, but has a higher development cost. The trip time is improved by operating at the highest efficiency achievable at moderate Isp, with 4000 s being the lower bound for the baseline case. For fixed power, efficiency affects only trip time. The engine lifetime is not a strong driver for mission performance, but must be at least several thousand hours. A complete optimization is required to assess the value of various technology trades; for instance, operation at a higher Isp may save sufficient propellant mass that a heavier power source with more power can be tolerated, yielding a net gain in trip time. The parametric study results, however, provide reasonable initial technology goals.

THE FOCUS OF ENGINE TECHNOLOGY DEVELOPMENT

For this class of missions, the emphasis must be on achieving high efficiency at an Isp of 4000–5000 s at power levels of 200–250 kWe. The engines must demonstrate very reliable operation for a service life of about 3000 hours if the flight thruster specific mass can be constrained to a value similar to that of the lab model

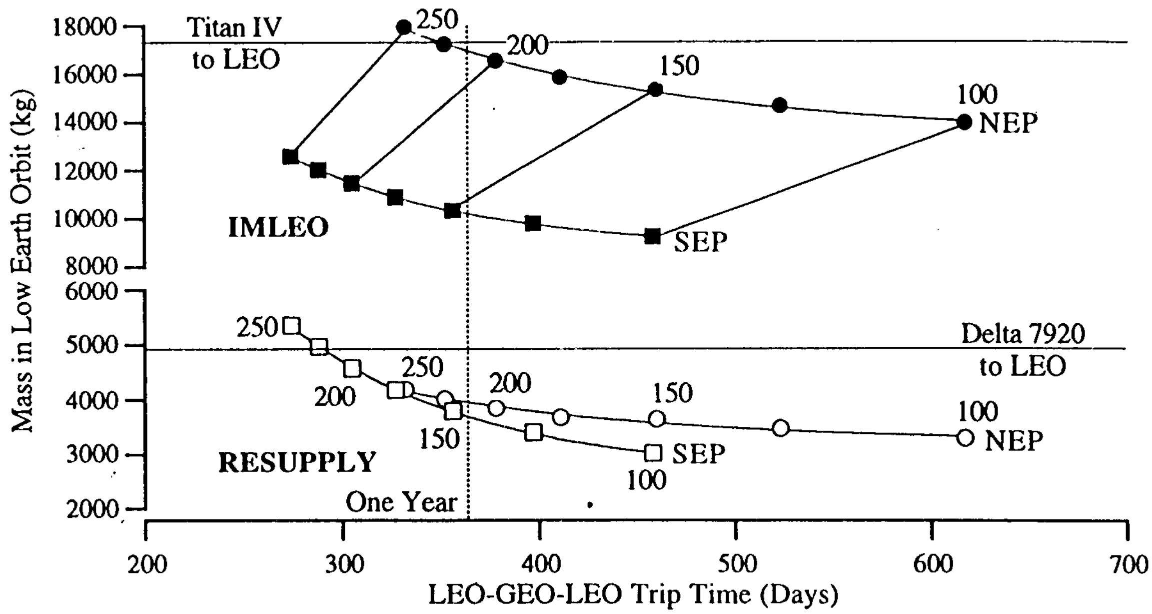

FIGURE 3: Mission Performance with Total Bus Power in kWe as a Parameter. α(SEP)=10 kg/kWe, α(NEP)=31 kg/kWe, Isp=4500 s, η=50% and Lifetime=5000 hours.

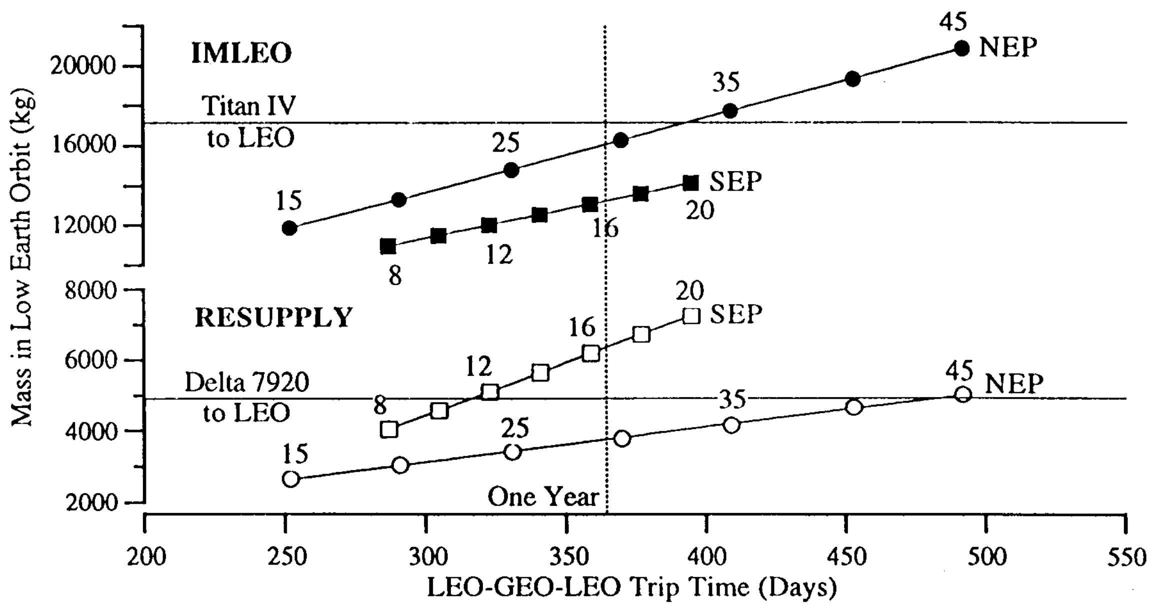

FIGURE 4: Mission Performance with Power Source Specific Mass in kg/kWe as a Parameter. Power=200 kWe, Isp=4500 s, η=50% and Lifetime=5000 hours.

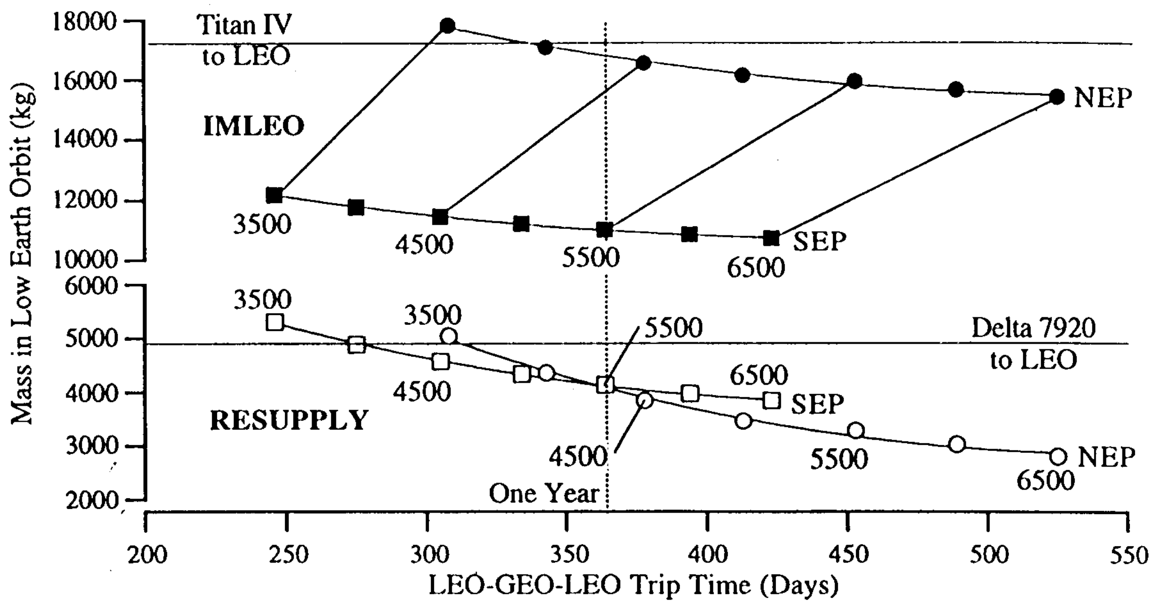

FIGURE 5: Mission Performance with Engine Specific Impulse in Seconds as a Parameter. Power=200 kWe, α(SEP)=10 kg/kWe, α(NEP)=31 kg/kWe, η=50% and Lifetime=5000 hours.

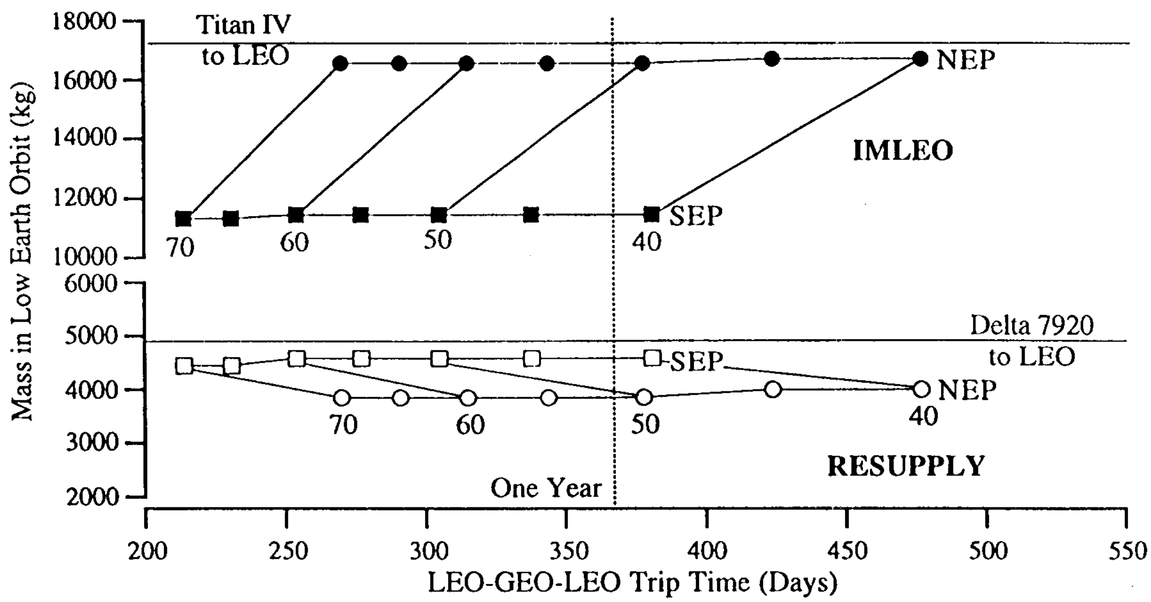

FIGURE 6: Mission Performance with Engine Efficiency in Percent as a Parameter. Power=200 kWe, α(SEP)=10 kg/kWe, α(NEP)=31 kg/kWe, Isp=4500 s and Lifetime=5000 hours.

engine. Engine efficiency and Isp increase with increasing current. Performance gains are limited by three lifetime concerns: the need to maintain stable plasma acceleration without starving the anode region, the current density capability of the cathode and the allowable anode heat flux. The stability criterion defines a maximum current capability for a given anode-to-cathode radius ratio. Performance models predict that the MAI thruster should be capable of operating at 200 kWe with an efficiency of 45% at a specific impulse of about 4000 s. With small modifications, primarily decreasing the electrode radius ratio, it should be possible to achieve an efficiency of greater than 50% at Isp's of 4500–5000 s and power levels of 200–250 kWe. To develop this engine for flight requires optimizing the mass flow rate and magnetic field strength to yield stable operation at the desired performance level, endurance testing to identify the dominant failure modes and characterize the operating environment (particularly for the cathode) and modeling these critical failure modes to assess engine reliability. This engine would enable round trip times of about one year for an NEP vehicle and 300 days for an SEP vehicle.

Improvements in power source technology which resulted in a lower specific mass, such as the use of dynamic power conversion instead of thermoelectrics in the nuclear system, would enable the use of higher power levels for orbit-raising missions within the launch vehicle constraints. This, in turn, would allow engine operation with higher efficiency and improvements in trip time. Maintaining stable operation at higher current levels requires decreasing the anode-to-cathode radius ratio further, which places greater loads on the electrodes. This may require an additional vaporizor in the cathode to introduce a small quantity of barium to the propellant flow to achieve the required life. Barium in small quantities will reduce the cathode work function, resulting in lower operating temperatures. The feasibility of this has been demonstrated in high power lithium thruster testing. In addition, at higher currents magnetic pinching can lead to a redistribution of propellant flow radially and higher temperatures on the periphery of the multichannel cathode. Cathode designs which force a more uniform mass flow and current density distribution will be required. With higher power densities it may no longer be possible to cool the anode radiatively. Active cooling, perhaps with a pumped liquid metal loop from the low temperature side of the reactor radiator in a nuclear system, or use of a larger radiator surface coupled to the anode with heat pipes may be required. These technologies should be explored for as applications at higher power levels arise.

AN EVOLUTIONARY PATH FOR THE DEVELOPMENT OF HIGH-POWER LFA's

More ambitious missions associated with the human exploration of the solar system will demand even higher power processing capability per engine. The next evolutionary step in the development of LFA's will be lithium-fuelled engines operating at 500 kWe to 1 MWe, an outgrowth of the technologies developed for the lower power levels. The technical feasibility of operation at these power levels has been demonstrated in Russia, where engines have been operated at over 1 MWe for several minutes at a time and a 500 kWe engine was run for 500 hours. The primary technical challenge is increasing the current capability of the cathode. At these power levels discharge currents of 6 to 12 kA will be required, an increase of two to four times over that required at the lower power levels. This can be achieved by scaling the cathode diameter up to maintain the same current density. The addition of barium will also likely be required to achieve useful life.

At these current levels the self-magnetic field is sufficiently high that applied fields are not necessary to achieve good performance. An applied field might still be used to increase the terminal voltage and control the current distribution on the anode. The same electromagnet technologies used in the lower power engines can be applied here, although the magnet may be actively cooled along with the anode. The same insulator material can also be used in the higher power engines as long as the design maintains a similar thermal environment.

Self-field engines operating at 400 to 500 kWe have demonstrated efficiencies over 60 percent at specific impulses of 4000 to 5000s. The main development issues once again are associated with defining regimes of stable operation and scaling up the cathode technology. Many of the dominant failure modes will be similar to those encountered at lower power levels though, so analysis tools developed for the first generation of engines will be applicable to the higher power thrusters. Detailed system and mission analyses have shown that engines operating at these power levels could be used to deliver 90 metric ton cargo payloads to Mars with an IMLEO of about 200 metric tons and trip times of about 2 years, which is a very attractive capability in supporting human exploration of Mars. Lithium LFA technology can be applied to a wide spectrum of missions which define an evolutionary growth in engine power processing capability, the unique domain of these engines.

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