ABSTRACT

The state-of-the-art performance of nine electric propulsion devices in terms of specific impulse and thruster and power processor efficiency and specific mass are discussed. Initial vehicle mass and trip time for four cases, ΔV of 1 and 6 km/s and payload mass of 3000 and 30,000 kg, was calculated and compared to a state-of-the-art storable chemical propulsion system. Situations where electric propulsion systems give a lower initial vehicle mass than a chemical propulsion system are shown. The advantages of the beamed energy thrusters, the microwave and laser thrusters, which don't carry their power source with them are also shown. The desirability of having a low power supply specific mass for those electric propulsion systems which must carry their power supply with them is demonstrated.

INTRODUCTION

Electric propulsion consists of the utilization of electrical energy to accelerate a low molecular weight gas to extremely high velocities to produce thrust. Because of the potential for significantly higher specific impulse than chemical propulsion, the corresponding reduction in the propellant supply which must be contained and transported in the spacecraft permits the inclusion of a greater portion of useful payload. Currently the highest specific impulse which can be expected from chemical propulsion system is approximately 420 to 480 seconds for hydrogen/oxygen reactants (Centaur G) in which the propellants are cryogenic and storability becomes a problem. The alternative fuel is nitrogen tetroxide and monomethylhydrazine, MMH, (N₂O₄,CH₃N₂H₃), which is storable at normal temperatures for long periods of time and provides 342 seconds specific impulse (XLR-132 engine). There exists three basic types of electric propulsion: electrostatic (ion thruster), electrothermal (resistojet, arcjet, laser, microwave, pulsed electrothermal), and electromagnetic, (magnetoplasmadynamic (MPD), pulsed plasma, pulsed inductive). Electrostatic thrusters use electric body forces established between an ion source and a negative grid electrode to accelerate a collisionless beam of positive atomic ions which subsequently joins a stream of electrons producing a beam of zero net charge, which in turn can provide specific impulses up to 10,000 seconds. Electrothermal propulsion comprises all techniques, including electromagnetic energy beaming from a remotely located source, whereby a propellant gas is heated and then expanded through a nozzle to convert its thermal energy to a jet of directed kinetic energy which can achieve specific impulses up to 2000 seconds. Electromagnetic propulsion utilizes a magnetic field acting on an electric current to accelerate an ionized gas by means of the Lorentz body force which is the vector cross product of the magnetic field and the discharge current. A specific impulse of 2000 to 4000 seconds can thus be obtained.

In addition to the specific impulse, three other performance parameters are of interest: thruster efficiency, input power and thrust. The relationship among these four quantities can be written as

\[ P = \frac{T I_{sp}}{2\eta} \]  

where \( g \) is the gravitational constant, \( T \) is the thrust, \( P \) is the power input to the thruster, \( I_{sp} \) is the specific impulse and \( \eta \) is the thruster efficiency. It should be noted that the thrust obtained for a given power is inversely proportional to the specific impulse, therefore electric propulsion thrusters are inherently low thrust devices due to power limitations and require relatively long transit times compared to chemical propulsion. For example, whereas a chemical rocket can orbit raise to geosynchronous Earth orbit from low Earth orbit in a matter of hours, electrothermal propulsion requires days and electrostatic propulsion requires months for the same transfer, but can accomplish it with up to 75% less propellant mass including the power supply mass and consequently less cost.

The spacecraft would normally be composed of the power supply (solar array or nuclear), payload, propellant and tankage and propulsion module which consists of the electric thrusters, power processor, and associated hardware. The available power options are solar electric or nuclear electric systems. The technology to fabricate lightweight, radiation hardened, large solar arrays has progressed significantly in recent years. For example, small solar arrays utilizing gallium arsenide technology have been fabricated with a specific mass of 1 kg/kW. The nuclear electric program is under development in which SP-100 will be available in the 1990's to provide power in the range from 10 to 1000 kW with a specific mass of 30 kg/kW including the power processing unit. The critical components of the nuclear power system are the reactor where the heat is generated, the thermionic converter which converts heat directly to electricity, and the secondary heat transfer loop where the waste heat is rejected to space.

Power processing units (PPU's) are required to convert the electrical power generated by the power supply to the voltage-current characteristics required by the specific thruster. For example, ion thrusters require high voltages (1000 volts) at low currents (1 Amp) while electrothermal thrusters require lower voltages (100 volts) at higher currents (20 Amps). Magnetoplasmadynamic thrusters require current on the order of kiloamps to operate in the steady state mode and pulsed devices such as the pulsed electrothermal and pulsed inductive thrusters require a means to rapidly switch a large amount of stored electrical power.

The purpose of this study was to compare the performance of state-of-the-art electric thrusters with a storable chemical propulsion system for near Earth missions. Only experimentally measured performance values for the various electric thrusters were utilized in this study. Flight ready hardware is available for some of the electric propulsion devices (i.e. ion thrusters, pulsed plasma thrusters, etc.) and in these cases the most recent test results were used.
The electric propulsion systems are compared with the XLR-132 liquid rocket motor technology using nitrogen tetroxide (N₂O₄) and monomethyl-hydrazine (CH₃N₂H₃). The engine specifications are 342 seconds specific impulse and the specific weight is 3.25 gm per Newton thrust. The design thrust of the XLR-132 is 3,750 lb at a chamber pressure of 1,500 psi. and a 400:1 nozzle expansion ratio. One unique feature of the XLR-132 engine is that it uses the oxidizer (N₂O₄) instead of the fuel (MMH) to regeneratively cool the nozzle. The design weight of the engine is 120 lbm.

Electrostatic Propulsion

The xenon ion thruster, as shown in Figure 1, is the latest version of a mature technology. Ions are generated in the thruster by a low-pressure discharge. The available ions are then extracted at one end of the discharge chamber by negative grids. These grids are thin, closely spaced sheets, usually of molybdenum, with many holes in each sheet. The grid closest to the discharge chamber is at a sufficiently negative potential to form the ions into many small beams. The second grid aligned with the first one provides further ion acceleration. The third grid is used to reduce the ion beam velocity to the desired exit velocity. Electrons are added by a neutralizer located at the exit plane to provide both space charge and current neutralization. The efficiency of an ion thruster can be increased with increasing atomic mass of the propellant (atomic mass of Xenon is 131) and by increasing the probability of propellant ionization which can be achieved by a deeper discharge chamber resulting in a greater neutral residence time. Using Xenon as the propellant, the ion thrusters can achieve specific impulses up to 3000 sec with an efficiency up to 0.59. The major concern about electrostatic thrusters is the power processing which entails both losses and the heat rejection capability and because operation at high pressures reduces frozen flow losses by lowering the dissociation level and increasing the recombination/reionization rates, improves the heat transfer from the heat source to the gas, reduces the radiation by increasing the optical depth of the gas, and permits a smaller chamber and nozzle for a given mass flow. The limiting factor is increased stress, both mechanical and thermal, on the chamber walls and increased nozzle throat erosion. The latter process has frequently been found to be a serious factor on the lifetime of an electrostatic thruster. The optimum operation pressure is estimated to be 1-5 atm. The specific mass of the resistojet system is 19 kg/kW and specific impulse varies from 385 sec to 117 sec for hydrogen and argon as the propellants respectively. The thruster efficiency with hydrogen propellant has been measured at 0.81.

Electrothermal Propulsion

Five electrothermal propulsion systems including a state-of-the-art resistojet system, a near-term arcjet system, an advanced pulsed electrothermal system and two beamed energy systems, microwave and laser propulsion, are considered in this analysis.

The simplest of all electric propulsion devices is the resistojet, wherein the propellant gas is heated by passing it over an electrically heated solid surface. As shown in Figure 2, many types of heaters have been considered including reentrant flow passages and parallel flow passages. The heat transfer to the propellant gas is primarily by conduction; convection and radiation have secondary effects. The choice of chamber pressure for a given electrothermal propulsion thruster may be an important factor in its overall efficiency because operation at high pressures reduces frozen flow losses by lowering the dissociation level and increasing the recombination/reionization rates, improves the heat transfer from the heat source to the gas, reduces the radiation by increasing the optical depth of the gas, and permits a smaller chamber and nozzle for a given mass flow. The limiting factor is increased stress, both mechanical and thermal, on the chamber walls and increased nozzle throat erosion. The latter process has frequently been found to be a serious factor on the lifetime of an electrostatic thruster. The optimum operation pressure is estimated to be 1-5 atm. The specific mass of the resistojet system is 19 kg/kW and specific impulse varies from 385 sec to 117 sec for hydrogen and argon as the propellants respectively. The thruster efficiency with hydrogen propellant has been measured at 0.81.

The arcjet uses an electric arc to heat a propellant, which then expands through a nozzle to generate thrust. As shown in Figure 3, it consists of a cathode, an anode also used as the nozzle and a constrictor where the energy exchange from the arc to the gas occurs and which also stabilizes the arc column. Most of the energy input occurs in the arc column, which fills only a small fraction of the constrictor cross section. The arc transfers its energy to the gas by means of radiation, convection and conduction and establishes the desired temperature profile across the gas stream where an intensely hot central core is surrounded by a relatively cold gas stream. For example, at a pressure of 1 atm, an arc current of 150 amp and a voltage gradient of 25 volts/cm, the central temperature of a hydrogen arc reaches 20,000 K and the wall temperature is approximately 2000 K. The recent test of 30 kW-class arcjets provides a specific impulse of 950 seconds with an efficiency of 0.40 using ammonia propellant. The overall specific mass of an arcjet is 1.8 kg/kW which includes power processor, the thermal rejection system, the arc chamber and nozzle. The life limiting problems are erosion of the cathode tip, the constrictor wall, and of the propellant injection nozzle. Cathode erosion is critical when it loses material from the tip and grows whiskers. After a 573 hour test, 1.95 gm of tungsten cathode was lost. Both the specific impulse, efficiency and lifetime are expected to improve as a result of current research.
One way to recover the ionization and some of the dissociation energy in an electrothermal thruster is to raise the gas pressure because the recombination rate varies as the square of the density. To avoid raising the power level due to the high pressure, the thruster is operated in a pulsed mode with a duty cycle of $10^{-3}$, which allows the thruster to operate at a kilowatt level CW while the peak power is in the megawatt range, sufficient to create the required plasma energy. Using this principle, liquid-plasma discharges are of growing interest for the generation of high density, high enthalpy flows for rocket propulsion. The pulsed electrothermal thruster (PET), shown schematically in Figure 4, uses a high pressure (approximately 100 atm), modest temperature (approximately 10,000 K) plasma generated in a capillary-confined electric discharge. A liquid propellant (water) is injected into the capillary chamber through a small orifice at the cathode. Plasma generation in these devices proceeds with the application of a high current pulse from a capacitor. Critical to the understanding of these discharges is the physics of two-phase heating of an injected liquid by a pulsed electric discharge in a capillary tube. Droplet-plasma mixing can occur in the fast flow field as a steady vaporization or via instabilities leading to an explosive event. Using water as the propellant, recent studies have shown that the pulsed electrothermal thruster can achieve a specific impulse of 1700 seconds with the efficiency of 0.66. The specific mass of a pulsed electrothermal thruster system is 14 kg/kW, including power processing and a heat rejection system.

Laser propulsion consists of using energy from a remotely located laser to heat a low molecular gas to extremely high temperatures followed by a gas dynamic expansion through a nozzle to provide thrust. Because of the potential for significantly higher specific impulse than chemical propulsion and adequate thrust to provide reasonable transit times, laser propulsion can be considered for a wide range of mission applications. Major components of a typical continuous laser propulsion are seen in Figure 5. The laser source can be either on the earth or in space. Continuous laser propulsion uses a steady-state (or quasi-state) plasma to absorb the energy of a laser through inverse bremsstrahlung to heat a propellant gas to extremely high temperatures (15,000 to 20,000 K). If the working gas is argon, a specific impulse of 1000 seconds can easily be obtained with the efficiencies of 0.35 to 0.40. The laser thruster is extremely lightweight with a specific mass which is approximately 0.0265 kg/kW including regenerative cooling plumbing, heat rejection system and laser energy collection optics. Efficiency is expected to improve as a result of current research.

Microwave propulsion is an alternative continuous beam propulsion system. The microwave energy is absorbed by a gas in a resonant cavity, a coaxial microwave plasmatron, or a plasma flame front region in a microwave waveguide, similar to a combustion wave. As shown in Figure 6, the resonant cavity mode appears to be the most promising method for propulsive purposes since the microwave-heated plasma is formed away from the walls of the cavity. This reduces heat losses to the walls and possible destruction of the cavity. The microwave energy can be transmitted into the cavity through a dielectric window which, along with the cavity, can form the gas containing rocket chamber. Microwave-heated plasmas are cooler (<10,000 K) than laser-heated plasmas and thus radiation losses are lower. The resonant cavity can be tuned such that up to 99% of the microwave energy can be absorbed in the gas. A specific impulse of 207 sec at an efficiency of 0.48 was measured for nitrogen gas and a specific impulse of 500 sec at an efficiency of 0.4 was measured for helium. For the cases where the microwave energy is beamed to the spacecraft from another source or where the spacecraft already has microwave generation...
equipment onboard for another purpose (i.e., communications, radar), the specific mass is assumed to be the same as for the laser thruster, 0.0265 kg/kW.

**Electromagnetic Propulsion**

Three electromagnetic propulsion systems, the magnetoplasmadynamic (MPD) thruster, the pulsed plasma thruster and the pulsed inductive thruster were considered in this analysis. The optimum operating conditions and performance characteristics are quite different from electrothermal propulsion. Electrothermal devices operate with relatively high chamber pressures (order of 100 torr or above) and low powers (order of 10 kW) and provides 800-1500 sec specific impulse at 40 to 50% efficiency. In contrast, electromagnetic devices operate at much lower pressures (order of 10 torr or below) and high powers (order of MW) with specific impulses up to 4000 seconds at 20 to 40% efficiency. In general, electromagnetic propulsion is based on the fact that an electric current drawn between two electrodes (cathode and anode) generates a self-induced azimuthal magnetic field. The actual acceleration force arises from the vector cross product of this current and its self-induced magnetic field, the "Lorentz force". This class of electromagnetic thrusters are called self-field thrusters. The class of electromagnetic thrusters which are called applied-field thrusters use an external magnetic field to produce the acceleration force.

The self-field steady magnetoplasmadynamic (MPD) thruster is a azimuthally symmetric device as shown in Figure 7. Only steady-state MPD thrusters were considered in this study since it is felt that the high erosion rates experienced by pulsed MPD thrusters will preclude their use in an operational system. The thruster consists of an cathode, an anode which forms the downstream portion of a cylindrical discharge chamber, and an insulator which covers the upstream end of the discharge chamber. Propellant is injected into the chamber through the insulator and current driven between the anode and cathode ionizes it and the discharge current generates the necessary magnetic field to accelerate the plasma through the Lorentz force. For acceptable performance characteristics, the discharge current should exceed several thousand amperes. This current level requires operating power levels in excess of 300 kW, therefore MPD thrusters are considered to be high power devices. With argon as the propellant, the state-of-the-art steady MPD system can provide a specific impulse of 1800 seconds at 10-20% efficiency with a specific mass of 0.15 kg/kW including power processor and thermal rejection system.

As shown in Figure 8, the pulsed plasma thruster (PPT) consist of two parallel flat-plate electrodes separated by solid teflon at the upstream end of the discharge volume. A high current, high power pulse is provided by a capacitor once the gap between electrodes becomes conducting by the discharge of a small ignitor plug mounted in the cathode. Once ignited, the current flows along the solid teflon surface and vaporizes a thin sheet of solid fuel which is subsequently accelerated downstream by the self-generated electromagnetic field. The overall pulse duration is some tens of microseconds. Typical specific impulses range from 1000 to 2000 seconds with an efficiency of 6%. Specific mass of a flight qualified thruster system is 264 kg/kW including heat rejection system, power processor and associate hardware.

As shown in Figure 9, the pulsed inductive thruster (PIT) consists of a flat spiral coil that is periodically pulsed from a capacitor. Prior to each pulse, a layer of propellant gas is transiently injected over the coil surface; the rapid rise of the radial magnetic field in the propellant induces electrical breakdown in the azimuthal direction and drives a large circulating plasma current that repels the gas from the coil at a useful specific impulse. Efficiency of the thruster is determined by a combination of factors that include matching of the electric circuit oscillation period with the time for decoupling of the plasma, and, particularly the compactness of the gas layer over the coil surface at the time of the discharge. The pulsed inductive thruster gives 1900 seconds of specific impulse at 29% efficiency with argon as the propellant and 2800 seconds of specific impulse at 32% efficiency with ammonia as the propellant. The specific mass is 0.44 kg/kW including capacitor bank, heat rejection system and hardware.

**MODEL**

Two missions ΔV's were examined in this study. For a low thrust mission to transport a Space-Based Radar from LEO (250 km) to a circular orbit of 1190 km altitude with an inclination change from 57° to 61° the ΔV is approximately 1 km/s using the Edelbaum Equation. The low thrust ΔV for a LEO to GEO transfer with a 28.5° inclination change was taken to be 6 km/s.

For each propulsion module (thruster, power processor and heat rejection system) examined in this study the supplied electrical power, $P$, and the type of power supply (solar versus nuclear) were specified. For nuclear power a specific mass of 30 kg/kW was assumed while for solar power gallium arsenide cells with a specific mass of 1 kg/kW were used. This allows one to calculate the propulsion module mass.
The initial vehicle mass can now be obtained from power levels examined. For a mission AV of 4 km/s propulsion in terms of initial vehicle mass and trip time. All electric propulsion devices except the pulsed plasma thruster and the resistojet give initial masses less than the chemical rocket when powered by solar energy although when powered by nuclear energy only the microwave and laser thrusters give a lower initial mass. One interesting result shown in Figure 11 is that the resistojet gives the lowest trip time over the range of power levels examined.

Figure 11 plots the initial vehicle mass and trip time for the same AV of 1 km/s but with a payload mass of 30,000 kg (the approximate maximum shuttle payload to LEO). The trends are the same as in Figure 10 although the initial vehicle masses and trip times are higher and the curves are more closely spaced together. All the electric propulsion devices except the pulsed plasma thruster and the resistojet give initial masses less than the chemical rocket when powered by solar energy although the pulsed plasma thruster still gives the longest trip time. All the electric propulsion devices considered except the resistojet and the pulsed plasma thruster give a lower initial mass than the chemical rocket for solar supplied power while for nuclear supplied power only the microwave and laser thrusters give a lower initial mass than a chemical rocket.

Figure 12 plots the initial vehicle mass and trip time for a mission AV of 6 km/s and a payload mass of 30,000 kg. For this case the resistojet gives the highest initial vehicle mass although the pulsed plasma thruster still gives the longest trip time. All the electric propulsion devices considered except the resistojet and the pulsed plasma thruster give a lower initial mass than the chemical rocket for solar supplied power while for nuclear supplied power only the microwave and laser thrusters give a lower initial mass than a chemical rocket.

Figure 13 plots the initial vehicle mass and trip time for a mission AV of 6 km/s and a payload mass of 30,000 kg. Once again the trends are the same as in Figure 12 with higher values for the initial vehicle mass and trip time but with the curves more closely spaced together. All the devices except the resistojet give an initial mass lower than the chemical rocket for solar supplied power and only the resistojet and the pulsed plasma thruster give a higher initial vehicle mass when nuclear power is utilized.

CONCLUSIONS

The state-of-the-art performance of nine electric propulsion devices was compared to storable chemical propulsion in terms of initial vehicle mass and trip time. For the four cases examined, AV of 1 and 6 km/s and payload mass of 3000 and 30,000 kg, most of the electric propulsion
thrusters give a lower initial vehicle mass than chemical propulsion when powered by solar energy although trip times are longer. For the high A/V, high payload mass case, most of the electric propulsion devices give a lower initial vehicle mass for both solar and nuclear power. Advantages of the beamed energy thrusters, the microwave and laser thrusters, were shown, particularly at high power levels since these devices do not have to carry their power supplies on board the spacecraft. The results show the importance of having a power supply specific mass as low as possible. The compiled electric propulsion database can be updated as new test results are obtained.

REFERENCES


### Table 1: Summary of Thruster Performance Data

<table>
<thead>
<tr>
<th>Thruster</th>
<th>Propellant</th>
<th>$T_p^{28}$</th>
<th>$I_{sp}$</th>
<th>$n$</th>
<th>$q_{pm}$</th>
<th>Ref.</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chemical</td>
<td>$N_2O_4/MMH$</td>
<td>0.05</td>
<td>342</td>
<td>-</td>
<td>-</td>
<td>2</td>
<td>Rocketdyne XLR-132</td>
</tr>
<tr>
<td>Ion</td>
<td>Xenon</td>
<td>0.09</td>
<td>2800</td>
<td>0.59</td>
<td>17.5</td>
<td>11</td>
<td>XIPS</td>
</tr>
<tr>
<td>Resistojet</td>
<td>H$_2$</td>
<td>0.15</td>
<td>385</td>
<td>0.81</td>
<td>19</td>
<td>13,14</td>
<td>Space Station Use</td>
</tr>
<tr>
<td>Arcjet</td>
<td>NH$_3$</td>
<td>0.05</td>
<td>950</td>
<td>0.40</td>
<td>1.81</td>
<td>15</td>
<td>JPL</td>
</tr>
<tr>
<td>PET</td>
<td>H$_2$O</td>
<td>0.05</td>
<td>1700</td>
<td>0.66</td>
<td>14</td>
<td>17,18</td>
<td>G-T Devices</td>
</tr>
<tr>
<td>Laser</td>
<td>Ar</td>
<td>0.05</td>
<td>1000</td>
<td>0.38</td>
<td>0.0265</td>
<td>19,20</td>
<td>Need tests with H$_2$</td>
</tr>
<tr>
<td>Microwave</td>
<td>He</td>
<td>0.10</td>
<td>600</td>
<td>0.40</td>
<td>0.0265</td>
<td>22</td>
<td>Need tests with H$_2$</td>
</tr>
<tr>
<td>Steady MPD</td>
<td>Ar</td>
<td>0.05</td>
<td>1800</td>
<td>0.10</td>
<td>0.15</td>
<td>23,24</td>
<td>JPL</td>
</tr>
<tr>
<td>PPT</td>
<td>Teflon</td>
<td>0.0</td>
<td>1000</td>
<td>0.06</td>
<td>264</td>
<td>25</td>
<td>Used on LES satellites</td>
</tr>
<tr>
<td>PIT</td>
<td>NH$_3$</td>
<td>0.05</td>
<td>2800</td>
<td>0.32</td>
<td>0.44</td>
<td>26</td>
<td>TRW</td>
</tr>
</tbody>
</table>
Initial mass, $M_i$, and trip time as a function of supplied power for $\Delta V=1$ km/s and a payload of 30,000 kg.
Fig. 12 Initial mass, $M_1$, and trip time as a function of supplied power for $\Delta V = 6$ km/s and a payload of 3000 kg.

Fig. 13 Initial mass, $M_1$, and trip time as a function of supplied power for $\Delta V = 6$ km/s and a payload of 30,000 kg.