A Survey of Propulsion Options for Cargo and Piloted Missions to Mars*

K. Sankaran, **L. Cassady**, **A. D. Kodys** and **E. Y. Choueiri**
Electric Propulsion & Plasma Dynamics Laboratory
Mechanical & Aerospace Engineering Department
Princeton University, Princeton, NJ 08544†

January 22, 2003

In this paper, high-power electric propulsion options are surveyed in the context of cargo and piloted missions to Mars. A low-thrust trajectory optimization program (RAPTOR) is utilized to analyze this mission. Candidate thrusters are chosen based upon demonstrated performance in the laboratory. Hall, self-field magnetoplasmadynamic (MPDT), self-field lithium Lorentz force accelerator (LiLFA), arcjet, and applied-field LiLFA systems are considered for this mission. In this first phase of the study, all thrusters are assumed to operate at a single power level (regardless of the efficiency-power curve), and the thruster specific mass and powerplant specific mass are taken to be the same for all systems. Under these assumptions, for a 7.5 MW, 60 mT payload, piloted mission, the self-field LiLFA results in the shortest trip time (340 days) with a reasonable propellant mass fraction of 57% (129 mT). For a 150 kW, 9 mT payload, cargo mission, both the applied-field LiLFA and the Hall thruster seem reasonable choices with propellant mass fractions of 42 to 45 % (7 to 8 mT) . The Hall thrusters provide better trip times (530-570 days) compared to the applied-field LiLFA (710 days) for the relatively less demanding mission.

Nomenclature

α_p	Specific mass of power supply (kg/kW)
α_t	Specific mass of thruster (kg/kW)
ΔV	Spacecraft velocity change (km/s)
η_{th}	Thrust efficiency (%)
B	Magnetic induction (T)
E	Electric field strength (V/m)
g_o	Acceleration due to gravity on earth (m/s^2)
I_{sp}	Specific impulse (s)
	= u_e/g_o
J	Total current (A)
$\mathbf{j}$	Current per unit area (A/m^2)
P	Input power to the thruster (kW or MW)
P_{th}	Thrust power (kW or MW)
T	Thrust (N)
$\dot{m}$	Propellant mass flow rate (kg/s)
u_e	Propellant exhaust velocity (km/s)

1 Introduction

For the first time in over a decade NASA has been given the green light to pursue nuclear options for spacecraft propulsion. The Nuclear Space Initiative (NSI), approved under NASA's FY2003 budget, is a multi-year program expected to total $2 billion with one goal being the development of space nuclear systems capable of 10-100 kW of power in space over the next ten years[1]. This initiative promises to open up the outer solar system to exploration by reducing spacecraft weight (propellant mass savings) and transit times (5 years versus 10 years to Pluto, with respect to chemical thrusters, at high power levels), and by providing a power supply to do science once the spacecraft arrives at the destination. As a result, the surface of Mars may now be accessible for long-term robotic and human exploration. This paper describes the first-phase of our study comparing near-term propulsion options for a two-stage (cargo & piloted) mission to Mars.

Because of their high exhaust velocities, electric

*The authors thank the NASA Johnson Space Center for providing the RAPTOR code.
†Presented at the *International Conference on New Trends in Astrodynamics*, Jan.20-22, 2003. Copyright by authors.

propulsion (EP) systems can provide significant propellant savings over chemical thrusters for high ΔV missions[2] like this one, and have been popular in such mission studies. With the success of the ion thruster as the primary propulsion system on the Deep Space 1 mission[3], and other recent EP enabled missions, the field of electric propulsion has now come of age as a reliable and efficient way of accomplishing relatively low energy missions. Currently, various types of EP devices, resistojets, arcjets, ion thrusters, Hall thrusters, and to a lesser extent, pulsed plasma thrusters, are routinely used for station keeping and maneuvering satellites. However, research on high-power electric propulsion has stagnated over the last two decades. As a consequence of the revival of interest in nuclear space systems, high-power propulsion options, first investigated in the 1960s to early 1980s (cf. ref.[4]) and then abandoned or continued at lower power levels due to lack of power in space, are receiving renewed attention.

1.1 Review of Previous Studies

Since the dawn of the space age, many mission studies have been performed on expeditions to Mars. Stuhlinger *et al.*[5] were among the first to propose the use of electric propulsion for missions of this kind. In the last two decades, several noteworthy studies have examined the advantages and disadvantages of various propulsion systems for a mission to Mars. We will briefly survey some piloted and cargo mission studies.

Coomes *et al.*[6] propose the use of a magnetoplasmadynamic thruster (MPDT - to be described in §3.2.1) operating at a power level of 6 MW for a piloted mission, and calculate a trip time of 600 days for Earth to Mars at this power level. King *et al.*[7] also examine the use of a MPDT for a similar mission and propose systems with input power up to 200 MW that can accomplish Earth-Mars round trip in less than a year. It is also suggested that MPDTs can offer trip time savings over chemical thrusters at power levels of 10 MW or higher. Gilland *et al.*[8] compare the use of MPDTs versus an array of ion thrusters for a similar mission, using a curve-fit for η_{th} vs. I_{sp} for the performance of these thrusters. Clark *et al.*[9] examine a 8 MW piloted (35 mT) fast trajectory mission for trip time, safety and reliability, abort options, and other costs. Pelaccio *et al.*[10] provide a technology readiness assessment of various thrusters for such a mission.

Clark *et al.*[9] also consider 4 MW minimum energy trajectory for a Mars cargo mission. They estimate that an array of ion thrusters offer significant mass savings over nuclear thermal systems, while maintaining comparable trip times. Frisbee *et al.*[11, 12] assess the technology readiness and development requirements for dynamic power conversion, power processing, and thrusters for Mars cargo mission. Polk *et al.*[13] examine the lithium Lorentz force accelerator technology (LiLFA - to be described in §3.2.2) for reusable orbit transfer vehicle with a parametric study of required power level, specific mass of power plant and performance to focus technology development. Noca *et al.*[14] consider robotic missions to outer planets with power levels ranging from 100 kW to 1 MW, using ion engines. Woodcock *et al.*[15] consider three outer planetary missions with small payloads, and consider the use of various propulsion systems.

As described above, a lot of work has been done on investigating propulsion options for missions to Mars. However, the abovementioned studies either perform the analysis with *extrapolated data*, and/or look at the problem from the perspective of research guidelines for a *specific thruster*. Therefore, there is a need for a comparison of multiple propulsion options for this mission using measured performance data only, and that is the goal of this paper.

1.2 Outline

The two-stage human Mars mission, chosen for this study, is described in detail in §2. This mission will benefit from nuclear power by yielding mass savings and reduced trip time, due to the ability to operate at higher power than would be otherwise available. In §3, we will examine some of the existing EP devices capable of taking advantage of near-future nuclear power systems, and we will discuss the current status of the technology and current trends in research. As will be described in §3.1, we will limit this study to thrusters that have been successfully operated (thrust measured) in the laboratory to keep with the near-term (10-20 years) spirit of the study and to perhaps provide some insight into technology drivers. The results of the mission analysis will be presented in §4. Following that, in §5, we will briefly discuss the propulsion options that were not considered in this analysis.

2 Mission Description

In this study we consider a two-stage mission to Mars. The power available for each stage of the mission was chosen in accordance with the near-term technology assessment of the thrusters, though many of the studies mentioned in §1.1 consider missions with larger power supplies.

In the first stage, 90 metric tons of cargo would be transported from Earth orbit to Mars orbit. The propulsion systems for this mission would have a total power supply of $\mathcal{O}$(100 kW) available, and the trip time is expected to be approximately two years. Because of the relatively low power available in this mission, the total payload of 90 mT would be transported in ten vehicles (each with a payload of 9 mT) to accomplish the trip within two years. The propulsion system for this stage of the mission should accomplish this with minimum propellant mass.

The second stage would carry the crew and supplies, totaling 60 metric tons of payload, and would be launched approximately two years after the launch of the cargo stage. The propulsion systems for this mission would have a total power supply of $\mathcal{O}$(1 MW) available, and the trip time must be less than one year, due to human health factors.

2.1 Trajectory Calculations

The mission simulation is accomplished with RAPTOR (RAPid Trajectory Optimization Resource), an optimization program developed at NASA-Johnson Space Center for low-thrust, interplanetary missions

The position and velocity of the departure and arrival planets are the boundary conditions. Given these, the code minimizes the total acceleration of the interplanetary trajectory. RAPTOR contains a genetic algorithm to converge on the proper Lagrange multipliers, trip length and departure date for the heliocentric code.

More details of how the code was used for this mission analysis will be given in §4.2.

3 Propulsion Options

In this section we will briefly describe the propulsion systems that may be suitable for this mission. Because of their high I_{sp}, EP systems are naturally attractive candidates for this type of mission. Within the family of electric propulsion devices, several types of thrusters, conceptually, have the ability to process 100s of kilowatts to megawatts of power at reasonably high efficiencies. We list them in table (3), and in §3.1, we narrow down the field to a few thrusters which will be considered in the mission analysis. Further information on these devices can be obtained from recent surveys, such as refs.[16, 17]. First, we will describe the criteria we used to select the thrusters.

3.1 Selection Criteria

While a variety of propulsion systems have been proposed for interplanetary missions, we restrict our analysis only to those that have:

1. been successfully characterized in a laboratory as a thruster (i.e., thrust and efficiency have been measured directly),

2. demonstrated a *potential* for attaining a significant lifetime ($\mathcal{O}$(100 hours)),

3. the ability process at least:

    - 25 kW per thruster for the cargo mission,

    - 500 kW per thruster for the piloted mission,

    so that the number of thrusters per spacecraft is reasonable.

Since, at present, there do not exist conclusive lifetime assessment tests of *any* of the devices, we consider those that have operated with tolerable erosion for 100 hours.

At high power levels, measured thrust and efficiency data is available for only three main classes of thrusters, Hall thrusters, thermal arcjets, and magnetoplasmadynamic thrusters (MPDT). Hall thrusters and thermal arcjets have been operated at power levels up to 150 kW, and are therefore well suited for the cargo phase of the mission. Some Hall thrusters examined in this study have operated only at lower power levels (25-75 kW) and therefore a cluster of 2-6 thrusters will be required to accomplish the mission. The only EP device to date to have demonstrated the ability to operate at megawatt power levels with a single (or reasonably small number of) thruster is the MPDT. For the MPDT two distinct variations exists differing in propellant and electrode design, both of which have been operated in the laboratory and will be discussed here.

Other promising thruster concepts, such as the ion thruster, Pulsed Inductive Thruster (PIT), and the VAriable Specific Impulse Magnetoplasma Rocket (VASIMR), that did not meet our selection criteria are discussed in §5.

3.2 Piloted Mission

Few thrusters have demonstrated performance at power levels of $\mathcal{O}$ (MW), and have survived many hours of laboratory testing. Consequently, the field narrowed down to gas-fed magnetoplasmadynamic thrusters (MPDT) and lithium Lorentz force accelerators (LiLFA). They will be briefly described in the subsequent sections.

3.2.1 Magnetoplasmadynamic Thrusters

In the magnetoplasmadynamic thruster (MPDT), a voltage applied between concentric electrodes breaks down a propellant gas, creating a quasi-neutral plasma within the thruster chamber. A high current ($\mathcal{O}(10^2 - 10^4)$A) carried by the plasma to the electrodes induces an azimuthal magnetic field, causing a Lorentz force ($\mathbf{j} \times \mathbf{B}$ per unit volume) to accelerate the plasma out of the thruster at velocities of $\mathcal{O}$(10 km/s)[4]. As shown in fig.(1), this body force accelerates the fluid in the direction perpendicular to both the electric and the magnetic fields.

The MPDT has a unique place among electric thrusters in its ability to process megawatts of electrical power in a small, simple, compact device and produce thrust densities (thrust per unit exhaust area) of $\mathcal{O}(10^5) N/m^2$[17]. However, this major advantage of the MPDT has also been the disadvantage to its development. Since high efficiencies ($> 30\%$) are only reached at high power levels ($>$ 200 kW), MPDTs require power levels that are an order of magnitude higher than what is currently available on spacecraft in order to be competitive with other propulsion options. Therefore, research on MPDTs was largely sidelined, in favor of thrusters that have higher efficiencies at lower power levels. Renewed interest in high-power MPDTs led to a flight-test of a quasi-steady MPDT in 1996 aboard the Japanese Space Flyer Unit, which was operated successfully at 1 kW [18].

Still, steady-state testing at the megawatt level is difficult, and to date all data in the 1 - 6 MW range has been taken in quasi-steady mode. In this mode, the thruster is operated for current pulse length of $\mathcal{O}$(1 ms), and data from this mode is expected to be a good indication of its steady-state performance[19]. Databases of measured quasi-steady thruster performance have been compiled in Japan [20] and at Princeton University [21]. A MW-class pulsed facility at the NASA-Glenn Research Center began operation in 2001, with plans to develop it to a steady-state facility[22]. So far, steady-state data is limited to less than 1 MW, and has been obtained mostly at the NASA Glenn (formerly Lewis) Research Center[23], and at the University of Stuttgart[24]. The NASA-Lewis test facility had the capability to operate at steady-state power level of up to 600 kW, but research was discontinued by the early 1990s. The focus of the research at University of Stuttgart is on investigating arc and plasma instabilities at power levels from 0.1 to 1 MW. At 0.5 MW, efficiencies of 28% at 1099 s were achieved during steady-state operation.

To date, the best gas-fed MPDT data has been achieved with hydrogen in quasi-steady mode, 43% efficiency at 5000s has been reported [25]. In general, however, results have been in the 10 - 35% in range at 1000 - 4500 s specific impulse using Ar, NH_3, and N_2 as propellants [26].

A major concern about the MPDT technology is the erosion of its cathode (cf. ref.([27]), which has often limited the lifetime of laboratory studies. However, there are some indications from recent research[24, 28] that this problem may be manageable.

To summarize, the three major technological issues exist in the development of the MPDT are: accurate performance measurements at steady state, characterization and optimization of the thruster's stable operating range, and demonstration of lifetimes of the order of mission requirements (8,000-10,000 hours).

3.2.2 Lithium Lorentz Force Accelerators

The lithium Lorentz Force Accelerator (LiLFA) can be considered the next-generation MPDT [17]. Its operating principle is essentially identical to that of the MPDT. The name is a result of largely historical reasons. However, two major differences between the LiLFA and the MPDT are to be noted. First is the choice of propellant. Whereas the MPDT traditionally uses inert gas propellants, such as argon, helium, and hydrogen, the LiLFA, as its name indicates, uses lithium vapor. Furthermore, the central electrode of the LiLFA differs from the single rod design common to most gas-fed MPDTs. Instead, the LiLFA employs multiple rods, tightly packed within a hollow tube. Propellant flow (see fig.(2)) is through the channels in the cathode that are created in between these smaller rods, rather than from the electrode base, as in the MPDT (fig.(1)).

These two major differences address some of the fundamental limitations of the MPDT uncovered during extensive testing in the 1960s and 70s. First, the choice of a low-ionization energy propellant (lithium) reduces the non-recoverable energy lost in ionizing the propellant, thus improving efficiency, especially at low power levels ($<$ 200 kW) where the ionization sink can approach 50% of the total input power. The use of lithium also precludes the need for high-voltage ignition capacitors and hardware required to achieve breakdown in inert gas systems. Moreover, lithium can be stored in solid form onboard the spacecraft, leading to potential mass savings. However, no space-qualified system exists for feeding lithium propellant. The multi-channel design for the central electrode, combined with the lithium propellant, has been shown to improve efficiency and increase thruster lifetime by reducing electrode erosion[29]. The role of the electrode design in increasing efficiency and lifetime is, however, poorly understood.

The possibility of the LiLFA as a high-power propulsion option was shown nearly a decade ago with the demonstration of 500 hours of erosion-free operation, at

Table 1: Summary of Propulsion Options for Two-Phase Mars Mission.

Thruster	Laboratory Models	Measured Thrust at P >25 kW	Lifetime > 100 hrs
Arcjet	√	√	√
Hall	√	√	√
Ion	√		√
LFA	√	√	√
MPDT	√	√	√
PIT	√	√	
VASIMR	√		

a power level of 500 kW. At this power level, an exhaust velocity of 40 km/s, thrust of 12 N, and thrust efficiency of 60 % were reported [29]. Due to its high efficiency and high thrust-to-power ratio, the LiLFA has emerged as a promising candidate for the piloted mission. However, since very little data is available on this thruster aside from what has been cited in ref.[29], there is a need for further verification of this performance. To this end, NASA-JPL is in the process of developing a high-power steady state test facility for LiLFA at MW power level.

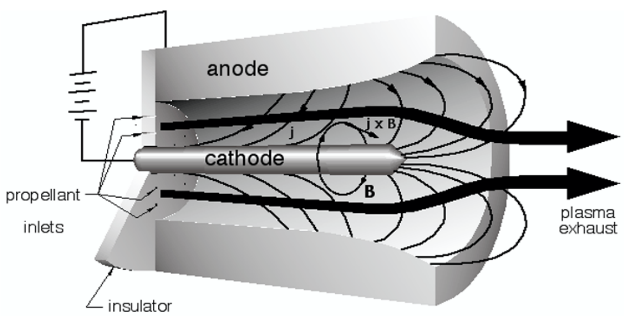

Figure 1: Schematic of the operation of the MPDT.

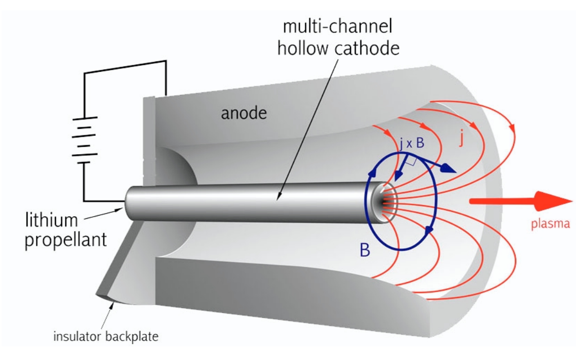

Figure 2: Schematic of the operation of the self-field LiLFA.

3.3 Cargo Mission

As mentioned earlier, there has been more research done on relatively low power propulsion than on MW-class propulsion required for the piloted mission. Hence, for the cargo mission, more thrusters are available for consideration. The thrusters that have met our selection criteria will be briefly described in the subsequent sections.

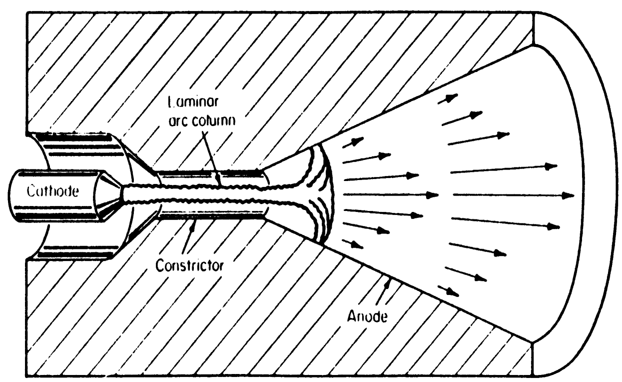

Figure 3: Schematic of the operation of the arcjet.

3.3.1 Arcjets

In this class of thrusters, an electric arc is used to add enthalpy to the propellant. Akin to a chemical thruster, part of the enthalpy in the flow is converted to directed kinetic energy using a nozzle.

As shown in fig.(3), a tightly constricted electric arc, carrying currents up to $\mathcal{O}(100)$ A, heats the core of the propellant stream to temperatures up to 10000 K, while the walls of the thruster are maintained at much lower temperatures ($<$ 3000 K) to prevent melting. Because of the higher temperatures in the core, and consequently, higher specific enthalpy, the exhaust velocity of an arcjet can reach, or even exceed, 10 km/s, as opposed to only 4 km/s for a chemical thruster. Its simple design and its high thrust density are some of the attractive features of the arcjet.

We will see in §4.3, however, that arcjets tend not to be competitive as our other option at the power levels considered.

3.3.2 Hall Thrusters

Hall thrusters derive their thrust by accelerating heavy ions, using an electric field, to high exhaust velocities.

Under high magnetic fields, and low densities, the current in the direction perpendicular (specifically, in the $\mathbf{E} \times \mathbf{B}$ direction) to the electric field (commonly called the Hall current) can exceed the current along the electric field. As implied by their name, Hall thrusters utilize this Hall current to lock electrons into a nearly collisionless cross-stream drift, leaving the positive ions free to be accelerated by the applied electric field (cf. fig.(4)). In a sense, these devices are hybrid electrostatic-electromagnetic accelerators with space-charge neutralization automatically provided by the background of drifting electrons. Because of this, the Hall thrusters are not affected by space-charge limitations. Therefore, Hall thrusters produce higher thrust densities than space-charge limited devices, such as ion thrusters.

Because the magnetic fields in these devices are externally supplied, and because the mass flow densities are intrinsically low, these thrusters optimize their performance at considerably lower powers than those of the self-field MPDTs. In fact, the requirement of low mass density (to maintain significant Hall effect) precludes the Hall thruster from producing thrust densities comparable to the MPDT and the LiLFA at a given power level.

The first Hall thrusters were developed in the US in the 1960s, but the research was discontinued in favor of ion engines[17]. However, research in the former Soviet Union picked up in the late 1960s, and by 1990s they had demonstrated efficiencies greater than 50%.

Hall accelerators of the closed-drift type are at present the most commonly used plasma thrusters. Since 1972, more than 100 Hall thrusters have been flown on Russian spacecraft. They are developed and used by the commercial sector for orbit insertion, attitude control, and drag compensation of satellites.

The Hall thrusters chosen for this study were the NASA-457M high power model[30] using xenon propellant, and the thruster with anode layer (Bi-TAL) from TsNIIMASH[31] using bismuth propellant.

The NASA-457M has been tested for power levels up to 75 kW. At this power level, its measured efficiency was 58%, specific impulse 2900 s, and thrust 2.95 N. A variant of this thruster[32] had an efficiency of 62%, specific impulse of 3250 s, and thrust 0.95 N at 25 kW of input power.

The Bi-TAL had efficiency greater than 70%, and a specific impulse of 8000 s at 140 kW input power. A variant of this thruster (designated TAL-200) had an efficiency of 67%, specific impulse of 3000 s, thrust of 1.13 N, at an input power level of 25 kW[31]. As a result of its high I_{sp} and high efficiency, this thruster has emerged as a promising candidate for the cargo mission. However, very little data is available on this device, and therefore, there is a need for further verification of its performance and lifetime.

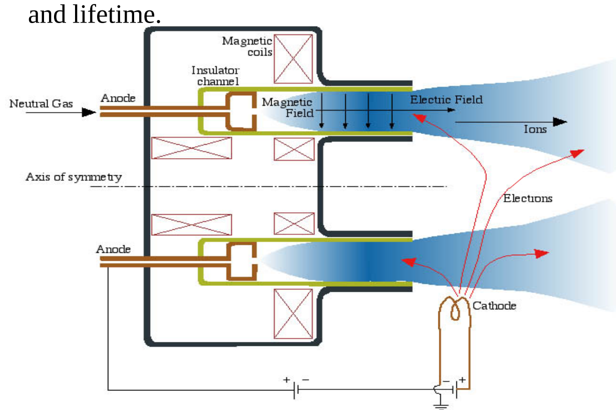

Figure 4: Schematic of the operation of the Hall thruster.

3.3.3 Applied Field LiLFA

Stated simply, the applied field LiLFA (AF-LFA) is a design to increase MPDT/LiLFA efficiency at power levels less than 200 kW, by adding an additional source of magnetic field. By using an external solenoid to enhance the magnetic field, efficient electromagnetic acceleration can occur at power levels where the current is too low to induce a substantial magnetic field. The AF-LFA offers the advantage higher efficiencies ($\geq$ 40 %) at lower power ($<$ 200 kW) compared to MPDT and LiLFA, while maintaining exhaust velocities (10-35 km/s) that are comparable. Apart from high-energy planetary missions, potential applications of the AF-LiLFA include missions requiring high thrust-to-power ratios, such as orbit transfer, N-S stationkeeping, and drag compensation.

Table 2: Summary of performance of thrusters for cargo phase of mission

Thruster	Power (kW)	Thrust (N)	I_{sp}	Efficiency (%)	Reference
H_2 Arcjet	75	2.5	1715	29	[34]
Bi Hall	25	1.13	3000	67	[31]
	150	2.5	8000	70	[31]
Xe Hall-1	25	0.95	3250	60	[32]
Xe Hall-2	75	2.9	2900	56	[30]
AF-LiLFA	150	4.0	3290	40	[33]
Argon MPDT	150	6.7	849	19	[24]

Table 3: Summary of performance of thrusters for piloted phase of mission (* denotes quasi-steady data.)

Thruster	Power (MW)	Thrust (N)	I_{sp}	Efficiency (%)	Reference
LiLFA	0.5	12	4077	60	[29]
H-MPDT-1*	1.5	26.3	4900	43	[25]
H-MPDT-2*	3.75	88.5	3500	43	[25]
H-MPDT-3*	7.5	60.0	6000	25	[21]
Argon MPDT	0.5	25.5	1099	28	[24]

At the power levels considered for cargo missions, the applied field lithium Lorentz force accelerator, jointly developed by the Moscow Aviation Institute (MAI) and NASA-JPL was a suitable candidate for consideration. The conclusion of a 5-year AF-LiLFA research program at the MAI [33] was the design of a 48% efficient thruster operating at 185 kW with 4200s and 4.5 N. For 130 kW of input power, this thruster has demonstrated 40% efficiency, 3290 s specific impulse, and 4.0 N of thrust in their study[33]. However, like in the case of its self-field variant, very little data is available on this thruster, and hence there is a need for further verification of its performance. Therefore, the collaborative effort by NASA-JPL and Princeton University is aimed at testing the applied-field LFA at power levels $\mathcal{O}$(10 - 100 kW).

4 Mission Analysis

4.1 Assumptions

In order to simplify our analysis, we have made the following assumptions for the present study, some of which will be relaxed in future studies.

1. The specific mass of the power supply, α_p, is assumed to be 4.0 kg/kW for all the cases. This is within the range of previous studies such as ref.[12] and ref.[8].

2. The specific mass of the thrusters for the piloted mission were all assumed to be 0.35 kg/kW [13]. Since many of the thruster considered in this study are still laboratory models, it is not easy to arrive at an accurate estimate for α_t. The influence of this assumption on the result is yet to be determined, since the mass of the thruster is expected to be only a small fraction of the total mass.

3. For the cargo mission, the mass of the thruster system was assumed to be negligible compared to the total mass, and was hence neglected.

4. The arrival date on Mars orbit was fixed to be the same for all cases of each of the two (cargo and piloted) missions.

5. The cargo mission was analyzed at a power level of 150 kW, irrespective of the optimum power level of the thruster. The piloted mission was analyzed at a power level of 7.5 MW, irrespective of the optimum power level of the thruster.

6. For the cargo mission, the total payload of 90 mT was evenly distributed among ten spacecraft. This

was done in order to ensure a reasonable trip time, given the limited (150 kW) power supply.

4.2 Calculations

For the thrusters that met our selection criteria (§3.1), we selected the highest measured performance data that was available and used it as input into the RAPTOR code. A summary of the thruster data is presented in tables (2 & 3). As noted in table (3), the data for the three types of hydrogen MPDT were obtained in a quasi-steady mode of operation, and it is expected to be a good indication of the steady-state performance as well[4].

For this study, we did not use the genetic algorithm, described in §2.1 to optimize the departure date due to the large amount time required to calculate the shortest trip. The dates of 12/1/2016 for the cargo, and 12/1/2018 for the piloted missions, were chosen as the arrival date at Mars (before the spiral) because those dates are expected to be near the minimum for those missions. The genetic algorithm was used to find the Lagrange multipliers and trip length that best satisfied the mission.

Given an initial mass in Earth orbit, the RAPTOR code can determine the final mass, or given a payload to Mars, the code can find the initial mass required. Since we have chosen a payload, RAPTOR will be run in latter mode. First, RAPTOR executes the spiral in to Mars orbit to determine that portion's duration and the propellant the maneuver requires. The heliocentric code then uses the mass of the payload and propellant to begin its optimizations. In this mode the date of arrival in Mars' sphere of influence is the only controllable date, all else is referenced to that date. The heliocentric code optimizes "backward" in time to find the minimum acceleration (i.e., minimum propellant) trajectory. Finally, the spiral to escape from Earth is executed and the mass at escape is matched with that at the beginning of the heliocentric transfer. The genetic algorithm is used only with the heliocentric portion of the code.

4.3 Results

The results of the RAPTOR code under the assumptions of this study are given in figures (5 & 6). The results for the trip time can be considered accurate to within $\pm$ 10 days, and for the propellant mass within $\pm$ 1 mT for both stages of the mission. The accuracy of the trip time is based on the sum of the round offs in the convergence calculation of various phases of the trip, and the accuracy of the mass estimate is based on the sum of the uncertainty associated with estimating the mass of the components such as the tank mass, and other structural mass.

The RAPTOR code does not explicitly optimize for trip time, rather it finds the trajectory that minimizes acceleration. This amounts to minimizing the required initial mass in nuclear-safe earth orbit. Since the thruster mass, payload mass, and power supply masses were assumed to be constant for each stage of the mission, the initial mass is a function of propellant mass alone. The minimum acceleration trajectory will result in the minimum propellant used and hence the minimum initial mass.

4.4 Cargo Mission

The cargo mission was restricted to a power level of 150 kW in order to utilize measured thruster data in this study rather than projections of performance. Our power restriction was found to place two major constraints on the analysis. First, as was discussed in the assumptions (§4.1), we found it necessary to divide the 90 mT of cargo into 10 smaller payload missions to reduce the trip times to something reasonable (in the neighborhood of 2 years). Second, even with the lower payload mass, the heliocentric portion of the mission was being optimized for trip times much longer than two years. To limit the mission length, 300 days was chosen for the heliocentric portion. Essentially, we picked the trip time and power available and from that allowed the code to find which thruster would use the least propellant to accomplish the mission. The differences in trip time come largely from the different times requires to spiral out of earth orbit. The assumption of restricting the heliocentric trip time did have one significant drawback. The code did not converge for the high-power bismuth Hall thruster in table(2), since its thrust-to-power ratio was too low to complete the heliocentric trip in 300 days.

Within the accuracy limits of our calculation, the AF-LiLFA and the Xe-Hall thrusters required the lowest launch mass fraction (42 to 45%, i.e., 7 to 8 mT). However, the time to spiral out of earth orbit for the AF-LiLFA is 140 to 178 days longer for a savings of only 1 to 2 mT over the Hall thrusters. Since this is close to our estimated error ($\pm$1 mT) it is not clear that the AF-LiLFA would provide significant mass savings over any of the Hall thrusters. From this analysis, the arcjet would not be the preferred choice for this mission. With a propellant mass fraction of 65% (18 mT), it required significantly more propellant mass and took longer time to arrive at Mars (712 days) than the AF-LiLFA or the Xe-Hall thrusters. The argon MPDT required the highest propellant fraction, 85% (55 mT), and took the longest(874 days) of the options considered.

4.5 Piloted Mission

For the piloted portion of the mission, the desired propulsion option would be the one which accomplishes the mission in the least amount of time. Trip times ranged from 490 days for the H-MPDT-3 ([21]) to just under 340 days for the LiLFA. The 3.75 MW H-MPDT-2 ([25]) had a trip time of only about one month longer (380 days) than the LiLFA. However, the initial mass required was also higher than the LiLFA. Due to its high specific impulse, the MPDT-3 required the least propellant mass fraction, 33% (45 mT), for the mission. However, its low efficiency (25%), and its lower thrust-to-power (compared to other choices at the same power level), prevented it from being competitive because of long trip time.

The range of trip times (340-490 days) depends upon the power level chosen (7.5 MW) for this stage of the mission. At that power level, the LiLFA is the best option, with the minimum trip time and a moderate propellant requirement (57% = 129 mT lithium) compared to other choices.

5 Other Viable Candidates

As mentioned in §3.1, we restricted our analysis to thrusters that have measured performance data, and have demonstrated significant lifetime. This eliminated many thruster concepts that may be promising for this mission.

Ion propulsion has demonstrated the in-space performance and lifetime necessary to be incorporated into future mission design and planning [35]. Good power throttling over a rather broad power range make ion propulsion ideal for solar electric propulsion missions where the electric power available varies with distance from the sun. In fact, the next-generation ion thrusters are being developed for such missions. NEXT, NASA's Evolutionary Xenon Ion Thruster, will provide higher power capabilities and lower specific mass with slightly increased exhaust velocities over Deep Space 1 technology[36]. These advances will meet the requirements of several near-term planetary missions including a Neptune orbiter and a Titan explorer[36].

Due to the electrostatic nature of ion propulsion, increased power (exhaust velocities) and propellant throughput (thrust) require corresponding increases in thruster size. The 30-cm DS1 thruster was capable of operation at up to 2.5 kW. The NEXT thruster will increase the effective area by 2, by moving to 40-cm diameter optics, and power capabilities near 10 kW. NASA's long-range goal for the development of ion engine technology is the demonstration of operation at 30 kW and above [35]. Work in 1968 investigated the feasibility of much higher power ($>$ 100 kW) ion thrusters. Preliminary tests on a 150-cm engineering model showed that operation at 177 kW was possible with exhaust velocities in excess of 7000s and calculated efficiencies of 76%. Thruster conditioning and grid stability issues arose at this size and power, as well as a need for higher power electron sources [37]. Lack of potential missions at that time caused the research program to end before these issues were solved or thrust measurements could be obtained. However, there appears to be no fundamental limit on thrusters of this size and power[35].

Another thruster concept that could be promising is the pulsed inductive thruster (PIT) [38]. Using ammonia as propellant, this thruster demonstrated 48% efficiency, with Isp of 4000 s at discharge energy of 2 kJ per pulse. If this thruster can be operated at a pulsing frequency of ($\mathcal{O}$(100-1000 Hz)), it would be competitive with the LiLFA for the piloted mission. However, the PIT has yet to show potential for lifetime of the order of the mission duration for it to be a serious candidate.

In addition, there are other thruster concepts, such as the VAriable Specific Impulse Magnetoplasma Rocket (VASIMR)[39], that may be suited for this mission. The VASIMR is a two-stage plasma propulsion device: the production of the plasma is accomplished in the first stage, and the heating and acceleration in the second. It is hoped that the separation of these two processes would allow for better control of the exhaust velocity, while utilizing maximum available power. This device is intended to operate at power levels ranging from 10 kW to 100 MW. If proven, its ability to vary specific impulse independent of power (which will likely require varying the propellant), can reduce both trip time and propellant utilization. However, this device has not yet been successfully operated in the laboratory as a thruster, and propulsive characteristics and performance have not been directly measured.

6 Concluding Remarks

The goal of this study was to examine electric propulsion options for near-term (10-20 years) cargo and piloted missions to Mars. Thrusters for the study were chosen from the highest performance data available, subject to the following constraints: that they had demonstrated operation at power levels of 25 kW (cargo) or 500 kW (piloted) in a *single* laboratory thruster, that thrust measurements at this power level had been published, and has demonstrated a *potential* for lifetimes on the order of at least 100 hours. Power levels chosen for this study were 150 kW for the cargo mission and 7.5 MW for the piloted mission. Trajectory analysis was performed by the NASA-JSC RAP-

TOR code which optimized acceleration for the heliocentric portion of the mission.

The cargo mission results showed that several of the thrusters we considered are promising candidates. For a chosen power level of 150 kW, the AF-LiLFA and all three of Hall thrusters considered could deliver the 9 mT payload with nearly the same mass in earth orbit.

For the piloted mission at 7.5 MW, the lithium Lorentz force accelerator (LiLFA) provided trip times savings of at least one month over any of the MPDTs in the study. The initial mass required to accomplish this was in the middle of the range of the thrusters considered. Overall, the LiLFA seems to be a promising technology for high-power, high ΔV missions of this type. Because the power available for this mission is fixed at 7.5 MW, the range of trip times (340-490 days) is longer than the estimates in other studies that consider much higher power levels.

This study provides a survey of electric propulsion options for cargo and piloted Mars missions. In order to more completely determine the relative strengths and weaknesses of each system considered, several of our assumptions need to be addressed in future work. So far, we have completed only the first phase of this study, with strong assumptions on specific mass of components, and thruster operation at a single power level only. The next step would be to perform trajectory analysis for each mission stage at a range of power levels. This would allow for the determination on of the optimum thruster and power level for a given mission. In addition, the assumptions of constant thruster and power supply specific mass for all thruster options might have influenced our results, given the large variations in power requirements of each thruster. Obtaining better estimates of these values would increase the relevance of our results. Finally, in the next phase of this work, a parametric study of thruster efficiency and specific impulse will be undertaken, which could provide guidelines for future research in thruster design and optimization.

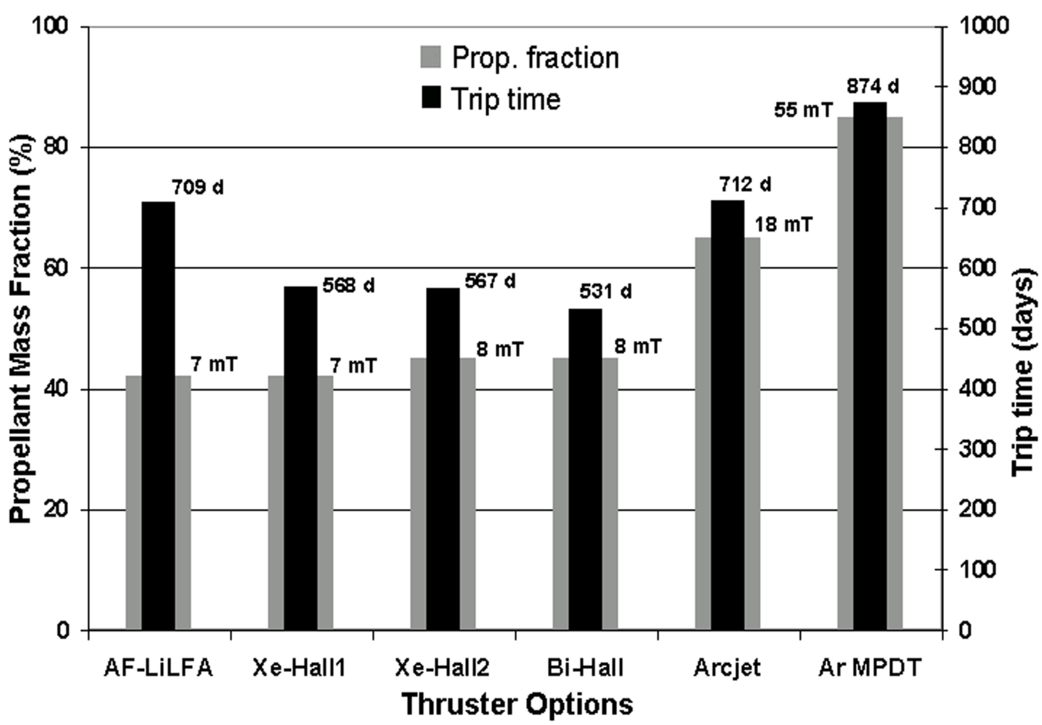

Figure 5: Results of the cargo mission analysis (all set to arrive on 12/1/2016), with increasing propellant consumption from left to right.

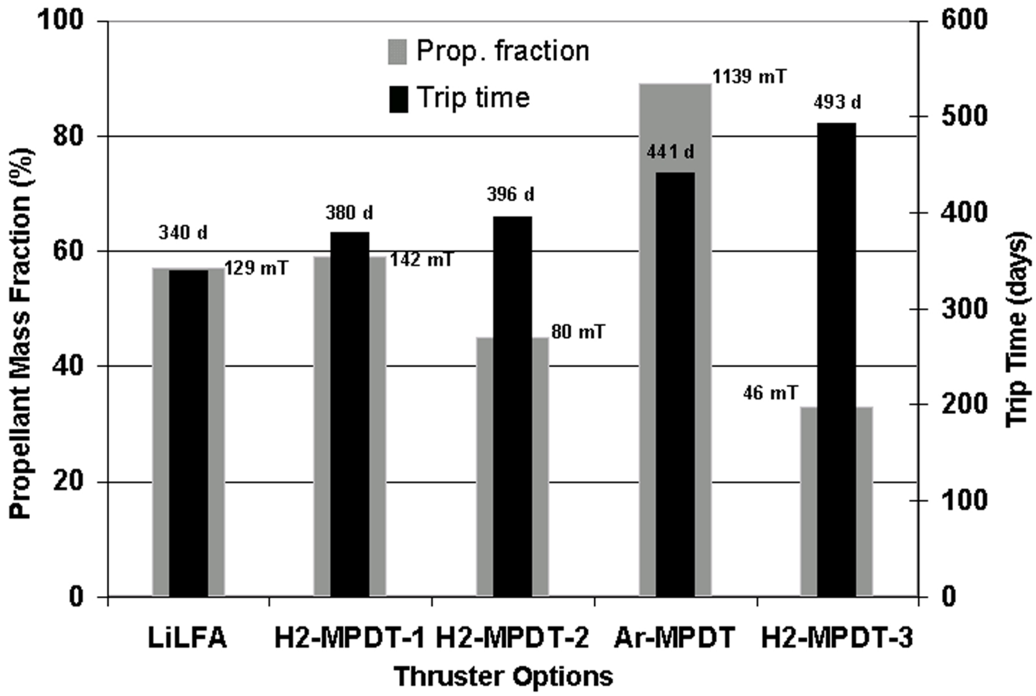

Figure 6: Results of the piloted mission analysis (all set to arrive on 12/1/2018), with increasing trip time from left to right.

References

[1] The Planetary Society. Nuclear Space Initiative: A Planetary Society White Paper. *The Planetary Report*, 2002.

[2] K.E. Tsiolkovsky. Exploration of the Universe with Reaction Machines. *The Science Review*, 5, 1903. English translation from the website of *Tsiolkovsky State Museum of the History of Cosmonautics*.

[3] J. E. Polk, R. Y. Kakuda, J. R. Anderson, J. R. Brophy, V. K. Rawlin, M. J. Patterson, J. Sovey, and J. Hamley. Validation of the NSTAR Ion Propulsion System on the Deep Space 1 Mission. *AIAA-99-2274*, 1999.

[4] R.G. Jahn. *Physics of Electric Propulsion*. McGraw-Hill, 1968.

[5] E. Stuhlinger and J.C. King. Concept for a Manned Mars Expediction with Electrically Propelled Vehicles. *Electric Propulsion Development: Progress in Astronautics and Aeronautics*, 9:647–664, 1963.

[6] E.P. Coomes, D.Q. King, and M.J. Patterson. PEGASUS: A Multi-Megawatt Nuclear Electric Propulsion System. *Third Annual Symposium on Space Nuclear Power*, 1986.

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

[8] J.H. Gilland and R.M. Myers. Multimegawatt Electric Propulsion System Design Considerations. *AIAA-90-2552*, 1990.

[9] John S. Clark et. al. Nuclear electric propulsion: A "better, safer, cheaper" transportation system for human exploration of mars. In *Space Technology and Application International Forum*, Albequrque, NM, USA, 1994. January 9-13.

[10] D.G. Pelaccio, G.A. Rauwolf, G. Maggio, S. Patel, and K. Sorenson. An Examination of Emerging In-Space Propulsion Concepts for One-Year Crewed Mars Misisons. *Annual Symposium on Space Nuclear Power*, 2002.

[11] Robert H. Frisbee, Nathan J. Hoffman, and Kathy H. Murray. Sp-100 dynamic power and lithium propellant MPD nuclear electric propulsion technology requirements. In *Space Technology and Application International Forum*, Albequrque, NM, USA, 1994. January 9-13.

[12] R.H. Frisbee and N.J. Hoffman. Electric Propulsion Options for Mars Cargo Missions. *AIAA-96-3173*, 1996.

[13] James E. Polk et. al. Technology requirements for high-power lithium lorentz force accelerators. In *Space Technology and Application International Forum*, Albequrque, NM, USA, 2001.

[14] M. Noca, J.E. Polk, and R. Lenard. An Evolutionary Strategy for the Use of Nuclear Electric Propulsion. *Annual Symposium on Space Nuclear Power*, 2000.

[15] G. Woodcock et al. Benefits of nuclear electric propulsion for outer planet exploration. *AIAA-02-3548*, 2002.

[16] M. Martnez-Sanchez and J.E. Pollard. Spacecraft electric propulsion- an overview. *Journal of Propulsion and Power*, 14:688–699, 1998. September-October.

[17] R.G. Jahn and E.Y. Choueiri. Electric Propulsion. *Academic Press Encyclopedia of Science & Technology*, 3rd edition(5), 2002.

[18] K. Toki, Y. Shimuzu, and K. Kuriki. Electric Propulsion Experiment (EPEX) of a Repetitively Pulsed MPD Thruster System Onboard Space Flyer Unit (SFU). In *International Electric Propulsion Conference*, 1997. IEPC 97-120.

[19] K.E. Clark and R.G. Jahn. Quasi-Steady Plasma Acceleration. *AIAA Journal*, 8:216–220, 1970.

[20] Akihiro Sasoh and Y. Arakawa. Electromagnetic Effects in an Applied-Field Magnetoplasmadynamic Thruster. *Journal of Propulsion and Power*, 8:98–102, 1992. January-February.

[21] E.Y. Choueiri and J.K. Ziemer. Quasi-Steady Magnetoplasmadynamic Thruster Performance Database. *Journal of Propulsion and Power*, 17:967–976, 2001. September-October.

[22] LaPointe Michael R. and Pavlos G. Mikellides. Design and operation of mw-class mpd thrusters at the nasa glenn research center. In *Joint Propulsion Conference*, Indianapolis, IN, USA, July, 2002. AIAA 2002-4113.

[23] R. Myers, M. Mantentiek, and M. LaPointe. MPD Thruster Technology. *Conference on Advanced SEI Technologies, AIAA-91-3568*, September, 1991.

[24] T. Wegmann et. al. Experimental comparison of steady-state nozzle type and cylindrical MPD thrusters at high current levels. In *International Electric Propulsion Conference*, Seattle, WA, USA, 1993. IEPC 93-122.

[25] H. Tahara, H. Yasui, Y. Kagaya, and T. Yoshikawa. Development of a Quasi-Steady MPD Arcjet Thruster for Near-Earth Missions. *AIAA-87-1001*, 1987.

[26] James S. Sovey and Maris A. Mantenieks. Performance and Lifetime Assessment of Magnetoplasmadynamic Arc Thruster Technology. *Journal of Propulsion and Power*, 7:71–83, 1991. January-February.

[27] J. E. Polk. *Mechanisms of Cathode Erosion in Plasma Thrusters*. PhD thesis, Princeton U., 1995.

[28] H. L. Kurtz, M. Auweter-Kurtz, W. Merke, and H.O. Schrade. Experimental MPD Thruster Investigations. *Journal of the British Interplanetary Society*, 41:223–232, 1998.

[29] V.P. Ageyev and V.G. Ostrovsky. High-current stationary plasma accelerator of high power. In *Proceedings of the 23rd International Electric Propulsion Conference*, Seattle, WA, USA, 1993. IEPC-93-117.

[30] D.H. Manzella, R.S. Jankovsky, and R.R. Hofer. Laboratory Model 50 kW Hall Thruster. *AIAA-2002-3676*, 2002.

[31] S. Tverdokhlebov, A. Semenkin, and J. Polk. Bismuth Propellant Option for Very High Power TAL Thruster. *AIAA-2002-0348*, 2002.

[32] D.T. Jacobson and R.S. Jankovsky. Performance Evaluation of a 50 kW Hall Thruster. *NASA-TM-1999-209447*, 1999.

[33] G. Popov, V. Kim, V. Tikhonov, S. Semenikhin, and M. Tibrina. The Fourth (Final) Quarterly Report on the Milestones (a)(4) and (a)(5)(D) of SoW of Contract No 960938 Between RIAME-MAI and JPL to NASA-JPL (Items 8 and 9 of Delivery Schedule). *RIAME-MAI Technical Report, Moscow, Russia*, 1998.

[34] M. Auweter-Kurtz, T. Golz, H. Habiger, F. Hammer, H. Kurtz, M. Riehle, and C. Sleziona. High-Power Hydrogen Arcjet Thrusters. *Journal of Propulsion and Power*, 14:764–773, 1998.

[35] Matthew T. Domonkos, Michael J. Patterson, and Robert S. Jankovsky. Ion engine and hall thruster development at the NASA Glenn Research Center. In *International Mechanical Engineering Congress*, New Orleans, LA, USA, November 17-22 2002. IMECE 2002-34444.

[36] Steven Oleson et al. Mission Advantages of NEXT: NASA's Evolutionary Xenon Thruster. In *AIAA Joint Propulsion Conference*, Indianapolis, IN, July 7-10 2002. AIAA 2002-3969.

[37] S. Nakanishi and E.V. Pawlik. Experimental Investigation of a 1.5-m-diam Kaufman Thruster. *Journal of Spacecraft and Rockets*, 5:801–807, 1968. July.

[38] C.L. Dailey and R.H. Lovberg. Pulsed Inductive Thrusters Performance Database for Megawatt Class Engine Applications. *AIAA-93-0103*, 1993.

[39] F. R. Chang Diaz, J. P. Squire, R. D. Bengston, B. N. Breizman, F. W. Baity, and M. D. Carter. The Physics and Engineering of the VASIMR Engine. *AIAA-00-3756*, 2000.