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# The NASA Low Thrust Propulsion Program

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## THE NASA LOW THRUST PROPULSION PROGRAM

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### SUMMARY

The NASA OAST Propulsion, Power, and Energy Division supports a low thrust propulsion program aimed at providing high performance options to a broad range of near-term and far-term missions and vehicles. Low thrust propulsion has a major impact on the mission performance of essentially all spacecraft and vehicles. On-orbit lifetimes, payloads, and trip times are significantly impacted by low thrust propulsion performance and integration features for Earth-to-orbit (ETO) vehicles, Earth-orbit and planetary spacecraft, and large platforms in Earth orbit. Major emphases of the program are on low thrust chemical, both storables and hydrogen/oxygen; low-power (auxiliary) electric arcjets and resistojets; and high-power (primary) electric propulsion, including ion, magnetoplasmadynamic (MPD), and electrodeless concepts. This paper will present the major recent accomplishments of the program and discuss their impacts.

### INTRODUCTION

The identification and development of new propulsion systems is fundamental to the continued effectiveness of NASA. Accordingly, NASA continues to maintain a technology program which spans focused development on Earth-to-orbit (ETO) and chemical transfer propulsion, basic propulsion research and technology (R&T), and advanced propulsion concepts (ref. 1). Essentially all spacecraft and vehicles incorporate low thrust propulsion systems which contribute a significant fraction of the vehicle/spacecraft mass, and the cost of delivering mass to low Earth orbit (LEO) is a major fraction of the cost of any mission. For example, on the Galileo mission, propellant accounts for 43 percent of the total mass and for the more challenging (larger velocity increment) comet rendezvous asteroid flyby (CRAF) mission, the propellant fraction is 76 percent (ref. 2). The objective of the low-thrust primary and auxiliary propulsion program, which is the focus of this paper, is to provide the basis for increased performance and operational life of small thruster systems by developing fundamental understanding in the areas of high-temperature materials, fluid physics, electrical and magnetic fields, and ion physics.

The emphasis of this program is on high-risk, high-payoff technologies such as chemical thrusters, arcjets, resistojets, ion engines, and magnetoplasmadynamic (MPD) thrusters. The work is principally being carried out through two NASA centers: the Jet Propulsion Laboratory (JPL) and Lewis Research Center (NASA Lewis). Both centers are working on a complementary program to advance the technology for low-thrust propulsion systems for a broad range of future space platforms, spacecraft, launch, and orbit transfer vehicles. JPL has been focusing principally on (1) improving ion engine life; (2) simplifying multiengine system architecture; and (3) developing the technology necessary to build and test an MPD thruster capable of steady-state operation at power levels and several megawatts. NASA Lewis has been focusing

principally on (1) improving chemical systems; (2) research on high-performance arcjets; and (3) basic physics and scalability of inert-gas ion thrusters and high-power MPD thrusters. There is a close interchange between JPL and NASA Lewis, and the two programs both complement and supplement each other.

This paper focuses primarily on progress beyond that reported at the 24th Joint Propulsion Conference (ref. 3), but some references to earlier work are included for clarity and continuity.

## CHEMICAL

Low thrust chemical propulsion finds application to a broad class of vehicles including ETO vehicles, Earth-orbital and planetary spacecraft and large platforms, including Space Station Freedom, in Earth orbit. On ETO vehicles storable propellants are currently used for on-orbit propulsion, but future systems may take advantage of using residual primary propellants (hydrogen or hydrocarbons and oxygen) for these functions. Advantages may accrue from both the use of scavenged residuals and from the increase in specific impulse ( $I_{sp}$ ). New architectures including propellant resupply create emphasis on extended life as well as increased performance. A high-temperature thruster effort is being conducted using advanced materials and diagnostic techniques and is aimed at increased performance and life with state-of-the-art and advanced storable propellants. Both storable (monopropellant hydrazine) and gaseous hydrogen/oxygen (H/O) systems have been investigated for Freedom.

Due in part to problems with Galileo's 10-N thrusters, interest has been heightened in understanding the fundamentals of small rockets. Support for small rocket fundamental research is provided by the Advanced Transportation Propulsion and Low Thrust Propulsion programs. This effort, which was initiated under the Low Thrust Propulsion program, is discussed briefly herein.

### Storable Propellants

Low-thrust, storable, monopropellant and bipropellant propulsion is widely used on a variety of spacecraft, ranging from small satellites to the space shuttle.

Bipropellant. - Storable bipropellants are used for many Earth-orbital and planetary spacecraft. A major accomplishment at NASA Lewis and Aerojet this past year has been the achievement of an extraordinary increase (~900 K) in chamber operating temperature for a 22-N advanced storable rocket using nitrogen tetroxide ( $N_2O_4$  or NTO) and monomethyl hydrazine (MMH) (ref. 4). The thruster, shown operating in figure 1, consists of iridium-coated rhenium fabricated by a chemical vapor deposition (CVD) process wherein the iridium coating is first deposited on an expendable mandrel of molybdenum and then the thicker rhenium layer deposited on top of it (ref. 5). The rhenium offers the desired high-temperature strength while iridium provides the needed oxidation resistance. Because of the high melting point of iridium temperatures approaching 2700 K are possible. This can also translate into a greater than tenfold increase in the operating life. This technology clearly overcomes the problems that existing thrusters have presented to NASA and other spacecraft users.

Aerojet first tested two iridium-coated rhenium thrust chambers with 8:1 expansion ratio, and over 15 hr of operation was demonstrated at temperatures from 2250 to 2530 K without failure or any evidence of corrosive attack or measurable change in throat dimensions. Similar results were recently demonstrated on a 22-N thruster with 150:1 expansion ratio fabricated from the same materials by the same technique (ref. 6). The significance of this can be seen in figure 2 which compares the state-of-the art (SOA) disilicide-coated niobium alloy chamber with the Ir/Re chamber. Not only does the Ir/Re chamber offer extended lifetimes but it offers higher  $I_{sp}$ . Because of this performance advance, NASA initially selected this technology for the Mariner Mark II comet rendezvous asteroid flyby (CRAF) mission. The reduced propellant mass (~600 kg) enabled the addition of three experiments to the spacecraft. To assure the readiness of this technology for the mission, the NASA Office of Space Science and Applications supported Aerojet, under the management of the JPL, for fabrication and test of a 440-N iridium-coated rhenium thruster, with the CVD processing performed by Ultramet. A 4-hr test indicated feasibility of the desired performance and life (ref. 7). While some spalling of the iridium has occurred because of contaminants that affected the iridium deposition both engines are serviceable and await additional testing. NASA Lewis is continuing to work with Ultramet to improve the fabrication process. A delay in the CRAF mission has recently occurred, resulting in less challenging propulsion requirements and a decision to use a propulsion system provided by NASA's European partner in this mission.

Monopropellant. - Although multipropellant resistojets and hydrogen/oxygen were baselined for Freedom propulsion, the technology of monopropellant hydrazine propulsion was advanced to provide a design option which might have lower development cost (although operating costs would be higher due to the requirement for propellant resupply). A life test of a 22-N thruster was conducted under a cooperation program between the Rocket Research Co. and NASA Lewis. The total impulse capability of  $4.8 \times 10^6$  N-s demonstrated is the greatest reported for small hydrazine rockets (ref. 8).

#### Hydrogen/Oxygen

Space Station Freedom. - Based on earlier studies manned space stations, two propulsion systems were investigated for different functions (refs. 9 to 11). As discussed elsewhere in this paper, low power multipropellant resistojets were selected for drag make-up. For the high-thrust (110 to 220 N) function, gaseous H/O thrusters were investigated (refs. 9 to 15). Initial thruster designs had an  $I_{sp}$  goal of greater than 400 sec<sup>1</sup> and, thus, had mixture ratios from 3:1 to 5:1. Subsequent studies of propellant availability for various station fluid management strategies led to the requirement to operate at an 8:1 mixture ratio with the products of water electrolysis and thus reduced the  $I_{sp}$  goal to 346 sec. This led to the need for thruster design modifications and an experimental demonstration of the  $8.9 \times 10^6$  N-s design goal. These successful results have led to H/O propulsion being baselined for the high thrust function on the Freedom.

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<sup>1</sup>In this paper  $I_{sp}$  is defined as thrust per unit weightflow and has units of seconds (sec) in both U.S. customary and S.I. units.

Future launch vehicles. - Studies at NASA Lewis have shown that advanced auxiliary propulsion subsystems (APS) can significantly increase the payload of advanced ETO vehicles (refs. 16 and 17). The concept is based on using an integrated H/O propulsion system in which propellants scavenged from the launch vehicle would be used for a liquid hydrogen/liquid oxygen (LH<sub>2</sub>/LO<sub>2</sub>) primary reaction control system (RCS) and a gaseous hydrogen/gaseous oxygen (GH<sub>2</sub>/GO<sub>2</sub>) vernier RCS. Unlike previous studies conducted during the space transportation system (STS) technology program, these studies have avoided any propellant conditioning requirements and have suggested the development of hardware to use the propellants in the as-delivered state. Figure 3 shows the proposed integrated H/O system in comparison with a 1960's vintage RCS concept.

The advantages of the currently proposed system include: (1) greater (by about 100 sec) system specific impulse; (2) simplified system through elimination of heat exchanger/gasifier, gas accumulator, and compressor; and (3) a decoupled thruster/propellant management system. Figure 4 shows the payoff in terms of increased payload mass for the integrated H/O system. A potential payload benefit of 4800 kg is projected for Space Shuttle 11. Such a payload benefit may also offer significant operational benefits in terms of the types of orbits that can be used. An industry study to verify these potential benefits is being initiated by NASA Lewis.

### Rocket Fundamentals

Present computer codes do not correctly predict the performance of small rockets. As a result, NASA Lewis has significantly modified a National Aerospace Plane (NASP) combustor code and extended it to space station H/O rockets. This code offers the advantage of including the effects of a shear layer to cool the nozzle (for a small rocket the boundary layer is not so thin in relative terms) and of not having numerical "patches" at the throat. The initial outputs of the code include combustion shear layer mixing, transonic flow, and viscous nozzle flows. The results are encouraging and NASA Lewis will work with NASA's Ames Research Center (ARC) to use the computational capability of the National Aerospace Simulator (NAS) for the continuing work (ref. 18).

### ELECTROTHERMAL

Electrothermal propulsion using subkilowatt resistojets is currently performing north-south stationkeeping (NSSK) on geosynchronous (GEO) satellites (refs. 19 and 20) and will be used for Freedom and man-tended platforms (ref. 2). The NASA electrothermal auxiliary propulsion program includes the development of resistojet options for high specific impulse and for long duration compatibility with multiple propellants. Kilowatt-class arcjets are being developed to provide increased life for higher power geosynchronous satellites. Power electronics technology is required for all these concepts and is, therefore, an integral part of the program, as it is also in the high power, primary propulsion area.

### Arcjets

1-kW preflight development. - NASA Lewis has been studying arc physics; electrode materials and their compatibility with propellants; heat transfer,

plasma arc, and fluid mechanics models. The NASA Lewis experiments have included plume fundamentals, arc/flow stability, and thruster/power processor integration fundamentals. NASA is working toward a flight test of an arcjet in the early 1990's. NASA Lewis successfully completed a 1000-hr, 500-cycle life test of a 1.3-kW arcjet this past year, thereby demonstrating system autonomy (ref. 21). No life-limiting mechanisms were identified as a result of the test. The arcjet met the pretest goals of 35 percent thrust efficiency and 450 sec specific impulse. Figure 5 shows the specific impulse as a function of the ratio of the power. NASA Lewis also demonstrated reliable, consistent startups of the arcjet (refs. 22 and 23). Figure 6 shows the arc voltage as a function of cycle number; note that most of the voltage rise occurred in the initial 10-hr burn-in period (ref. 21).

Before arcjet technology can be applied to spacecraft, however, a number of spacecraft issues must be addressed. One of these is the thermal impact the arcjet will have on both the spacecraft and on the power connection to the power processing unit. NASA Lewis has been analyzing the thermal/mechanical interface between the arcjet and the spacecraft. The goal is to keep the back temperature low enough ( $<400$  K) and the conducted heat low enough ( $<10$  W) to minimize problems in designing the rear electrical connector. An obvious approach would be to extend the lengths of the arcjet; however, the structural analysis showed that the arcjet cannot be made long enough to get the desired temperature because of excessive heat losses. This pointed out the need for high emissivity surfaces. Further analyses showed that a coating such as zirconium diboride with an emissivity of about 0.6 would meet the thermal interfaces (ref. 24).

Under NASA Lewis contract, supported by the OAST Flight Projects Office, Rocket Research Co. has fabricated a flight-type, kilowatt-class arcjet, which is shown in figure 7. The electrode configuration has been verified by performance demonstrations and life testing. The arcjet has been successfully integrated with the power processing unit (PPU) and has met the thermal interfaces. The dynamic structural analysis of the arcjet has been completed and the electromagnetic effects on satellite communications have been assessed.

While many major issues such as electrode life and reliability have been addressed (e.g., ref. 21) little has been done to improve the efficiency of the device. Preliminary experimental work has shown that the operating characteristics of arcjets depend on the location of the arc attachment zone in the anode/nozzle. An anode/nozzle consisting of five pieces of tungsten insulated from each other by boron nitride spacers was used in a radiation-cooled, segmented anode to study the physics of the arc attachment zone (ref. 25). Large potential gradients were shown to exist in the nozzle during operation with all the segments attached to a common point. Preferred modes of operation were identified by forcing the point of attachment to different positions. Further, use of the segments as floating probes gave rough estimates of plasma potential, and, by subtraction, the anode drop. A knowledge of the fundamental of arcjets is important because of the existing low efficiency (33 to 38 percent) of arcjets caused, in part, because it expends so much energy in breaking bonds leading to a "frozen flow" with about 80 to 85 percent of the energy in the gas itself. Computationally, nozzle studies are difficult because of the extremely small dimensions, low Reynolds number operation, and complicated arc physics in the constrictor/nozzle region. The plasma and flow fields are strongly coupled, which complicates the analyses. Figure 8 shows that the nozzle geometry affects the arcjet characteristics (ref. 26). While the performance is

insensitive to the upstream design of the thruster, it appears that a conical-diverging nozzle is the optimum design.

As part of its effort to understand the fundamentals of arcjet (and ion engine) performance, NASA Lewis has established a laser-induced fluorescence (LIF) facility and is now identifying the plume species as part of its investigation of the electrode losses and spacecraft interactions. Under internal funding, JPL will be measuring the velocity and temperature profiles in the exhaust stream of a 30-kW arcjet using LIF. Figure 9 shows the JPL LIF experimental setup (ref. 27).

Plume impacts. - As the arcjet thruster moves closer to being a flight-ready propulsion system onboard communications satellites for applications such as NSSK, spacecraft integration concerns become of significant interest. A key issue centers on the potential impact that the arcjet plume may have on the transmission of RF signals in common use for satellite communications. A first order study using Langmuir probe data indicated that the impacts appear negligible in the antenna far field, except for propagation paths near the arcjet source (ref. 28). More rigorous numerical techniques are necessary to fully define the interaction processes and effects which would occur if the plasma were to expand within the antenna near field. Recent results (ref. 29) extend the previous work and present a body of systematic experimental data which correlate measured plasma properties in the arcjet plume with a wide range of thruster operating conditions and constrictor geometries and confirm the earlier finding that communications impacts should be negligible.

5-kW System technology. - Recent interest in arcjets has led to the development and life tests of both 1 and 30 kW (e.g., ref. 30) arcjet systems for application to GEO satellite stationkeeping and prime propulsion. Orbit transfer missions from LEO to GEO might significantly benefit from arcjet systems in the 5-kW power range coupled with the use of high thrust chemical hydrazine propulsion to rapidly transverse the Van Allen radiation belts. The low thrust, high  $I_{sp}$  hydrazine arcjet system would then be used to complete the orbit raising and inclination changes required to achieve geosynchronous orbit. Such high-power arcjet systems will likely be energized from high-voltage buses since high bus voltage results in low spacecraft power system mass.

A 5-kW class arcjet system was recently developed at NASA Lewis (ref. 31) using a full-bridge, pulse-width-modulated power converter with electrical isolation. Previously, a 1-kW class pulse-width modulated parallel converter was used. For high bus voltages, a bridge converter has advantages over the parallel converter. The bridge converter power transistor switches operate at bus voltage whereas the parallel converter transistor switches must support twice the bus voltage. This advantage is partially offset by having two switches in series, but lower voltage transistors also have lower ON resistance. Another advantage of the bridge converter is that the power transformer is simpler and lower mass than the parallel converter power transformer. The laboratory 1-kW, radiation-cooled arcjet used for this work is an extension of a design described elsewhere (ref. 21). Specific differences from the 1-kW class arcjet include modifications in the cathode, constrictor, and thruster heat radiating area.

## Resistojets

High performance. - To advance storable propellant resistojets beyond the current state-of-the-art 300 sec mission-averaged  $I_{sp}$  (ref. 32), several efforts have been undertaken. NASA Lewis is working through a four-phase contracted industrial development program to design, fabricate, and characterize a hydrazine resistojet thruster capable of about 320 sec  $I_{sp}$ . The first phase, which was conceptual design and analysis, has been completed and Rocket Research was chosen for the design finalization and performance characterization. Currently the program is assessing the tradeoff on immersed heaters versus radiatively-coupled heaters. Improved heat exchanger performance, longer-life higher-temperature heaters (refs. 33 and 34), and improved nozzle performance (refs. 35 and 36) are being sought. The importance of high quality test facilities for evaluating performance has been demonstrated (ref. 37).

NASA Lewis has been modeling the supersonic viscous flow in a resistojet by solving the Navier-Stokes equation in conservative form with a finite-volume formulation. A typical solution is shown in figure 10, which gives a plot of Mach number contours for an axisymmetric converging-diverging nozzle with a  $20^\circ$  divergent half-angle, area ratio of 20, and 1086 Reynolds number (ref. 38).

Multipropellant. - As previously mentioned, multipropellant resistojets have been baselined for Freedom. The requirements are quite different from most other propulsion applications in that long life and integration features are much more important than performance (refs. 9 to 11). Utilizing Freedom wastes as propellant minimizes resupply requirement and may eliminate the need to return some wastes to Earth. Engineering model thrusters have been built and delivered to NASA for testing (ref. 39). Performance characteristics on a wide range of possible propellants have been obtained (ref. 40), and long-term life tests have been conducted for over 10 000 hr with over 140 deep thermal cycles (ref. 41). A power controller was developed to enable the Freedom resistojets to operate on high-frequency power (ref. 42). To facilitate the integration of this technology on Freedom, propellant and feed system options were assessed (ref. 43), and interfaces were defined (ref. 44). The exhaust plumes of the thrusters are of concern because of potential effects on sensors and experiments and potential attenuation of signals propagating through the plume. Analytical and experimental techniques have been developed (ref. 45), and a preliminary assessment made of the effect of nozzle geometry on plume characteristics (ref. 46). The exhaust of the engineering model resistojet was investigated using rotary pitot probes and a rotating quartz crystal microbalance. The effect of a plume shield has been evaluated and found to be small, but the absolute mass flux into the backflow region was very low ( $\sim 7 \times 10^{-7}$  g/cm<sup>2</sup>-sec) (ref. 47).

Water. - Interest in water resistojets developed as a spin-off to the Freedom multipropellant resistojet development. For man-tended, shuttle-serviced platforms such as the proposed industrial space facility (ISF), ease and safety of propellant resupply and ground handling operations are critical issues, which leads to interest in water as a propellant. With a resistojet capable of steam operation already under development for Freedom, water resistojets were baselined for the ISF (ref. 48). This necessitated the development of a zero-gravity steam generator when it was determined that available steam generators, such as that developed for the Manned Orbiting Research Laboratory, did not perform stably and reliably and that operation of the Freedom



resistojet with liquid water feed could be quite unstable (ref. 40). A cyclone steam generator was developed after evaluation of a number of boiler concepts studied for space power application. Further development of this concept led to an integrated vaporizer/resistojet utilizing a single heater (ref. 49). This concept is shown in figure 11; the liquid water is swirled to the outer wall and boiled by heat radiated from the central heater which also heats the resistojet by conduction. Thermal losses are reduced since the outer water operates at temperatures below saturation. This design also overcomes the many stability and phase separation problems typically encountered in zero-gravity boilers. Laboratory models have been built and performance mapped over a range of conditions, and flow visualization results obtained (ref. 50). Initial studies of the plume characteristics of steam resistojets has recently been conducted (ref. 51). Mass flux and flow angle measurements were obtained in the plume using a variable temperature quartz crystal microbalance.

### PRIMARY (HIGH POWER) ELECTRIC

Prospects now appear good for broad acceptance and application of electric propulsion systems (ref. 20). However, current applications are limited to low-power stationkeeping and orbit control (refs. 19, and 52 to 54). The successful operation of these systems has led to increased user confidence, and further operational use of electric propulsion appears imminent.

The National Commission on Space advocated a number of challenging missions, such as a return to the Moon, unmanned and manned exploration of Mars and its moons, and unmanned scientific exploration of the rest of the solar system (ref. 55). Many of these missions would be enhanced, and some would be enabled by high  $I_{sp}$  electric propulsion. To perform the challenging future missions, high power and high  $I_{sp}$  systems will be required. Candidate systems include electrostatic (ion) and MPD engines, with electrodeless approaches representing a longer term possibility. Electric propulsion is also applicable to advanced robotic exploration missions, offering both decreased propellant mass (and/or increased payload) and reduced trip times for deep space missions. Furthermore, this technology may be applicable to manned spacecraft using artificial gravity, where speed is not so critical. There are also many potential applications for high  $I_{sp}$  electric propulsion at the tens of kilowatts available from large solar power systems. High specific impulse clearly offers propellant mass savings, but in order to practically exploit that benefit for space missions, it is necessary for the overall vehicle to exhibit acceleration levels sufficient to meet mission time lines. This requirement necessitates low specific mass and high efficiency propulsion and power systems in order to keep power system mass low.

### Mission and System Analyses

Mission analyses covering spacecraft from small, solar electric propulsion (SEP) (refs. 2 and 56) to large, megawatt nuclear electric propulsion (NEP) spacecraft (refs. 57 to 61) show enhancing to enabling levels of capability for exploration of the planets and beyond (ref. 52). JPL considered a small, 180-kg demonstration spacecraft with a 1.5-kWe solar array using ion propulsion for transportation from LEO to lunar orbit (ref. 56). This Lunar Get Away Special (LGAS) mission would carry one instrument, nominally an Apollo gamma ray spectrometer, and would carry 36 kg of xenon propellant. JPL has

also studied a deep space mission concept, wherein the Thousand Astronomical Unit (TAU) Explorer, propelled by NEP, would reach 1000 A.U. in 50 years with a significant science payload (ref. 57). Both of these spacecraft accomplish the mission propulsion with a minimum of propellant mass and, thereby, maximize the science payloads at their destinations. A study was conducted by JPL (ref. 60) of the benefits ion propulsion might provide to the CRAF mission if it were used instead of storable bipropellants. Even though the mission and its science package were not optimized for use of electric propulsion, a 3-year trip time reduction could be achieved along with a 600-kg increase in launch mass margin.

A range of solar- and nuclear-powered Earth orbital missions have been studied and show cost/mass reductions benefits (refs. 62 to 64). In a USAF study it was shown that a solar electric orbit transfer vehicle for delivery of satellite constellations to their operational orbits could offer significant savings (ref. 62).

Nuclear electric propulsion system technology for an electric cargo vehicle has been identified as a major requirement for evolutionary solar system exploration (ref. 65). The use of an electric, low-thrust cargo vehicle in the Lunar Outpost to Early Mars Evolution case could potentially reduce the initial mass in LEO (IMLEO) by one-third over a standard chemically propelled vehicle. Studies have shown that for a Mars cargo vehicle large enough to support a manned mission, high performance electric propulsion with specific impulses over 4000 sec at multimewatt power levels can offer major reductions in propellant requirements at acceptable transit times. Compared with an aerobraking cargo vehicle using hydrogen/oxygen propulsion, and nonaerobraking high performance electric vehicle can result in elimination of at least three Heavy Lift Launch Vehicle (HLLV) launches, and savings are even greater when compared with nonaerobraking chemical propulsion (ref. 66). This reduction in the number of launches might permit the mission to be completed more rapidly if HLLV launch rates are low. To perform these challenging future missions, the total impulse capability must be advanced from the currently demonstrated  $10^6$  N-s for both ion and MPD engines to at least  $10^8$  N-s (ref. 67).

## Ion

Ion engines have demonstrated specific impulses from less than 2000 sec to more than 10 000 sec, thrust efficiencies to over 0.75 (e.g., ref. 68), and total thrust impulse as high as  $10^6$  N-s for 10-kW class thrusters. Propellants have included mercury, argon, krypton, and xenon. Ion thruster technology has been demonstrated with flight testing of the Space Electric Rocket Test (SERT 1, 11) ion thrusters (ref. 69). Electrostatic (ion) propulsion is planned for its first operational use on the Japanese Engineering Test Satellite VI (ref. 54). Thruster size is adjustable to fit mission requirements (fig. 12). But to advance to significantly higher power, the key issues are scale-up of ion acceleration subsystems for high power operation and thruster life. High power operation is limited by ion optics, and extraction area must be increased with increasing power. Ion propulsion systems in the range of 10 to 50-kW offer significant payload and trip time advantages over chemical systems for lunar ferry and outer planet explorer missions (including multiple target missions), as well as for missions outside the solar system. A near-term target for planetary and Earth orbital missions is about 5 to 30 kW per ion thruster using xenon as a propellant. The planned Pathfinder Cargo Vehicle

Propulsion program will focus on a target of 0.1 to 1 MW per ion thruster using argon or krypton. While the performance engineering of ion engines is well in hand the scalability, system, lifetime, and power processing issues remain open and are being addressed in the NASA program.

Lifetime. - Interest in near-term application of ion thrusters for solar electric propulsion in near-Earth missions (refs. 62 and 70) has led to a program to identify and extend the operating limits of ion thrusters originally developed for 3-kW operation with mercury to the 10-kW range using inert gas propellants (refs. 71 and 72). The original mercury ion engines presented life limiting phenomena like spalling of metal flakes eroded from the discharge chamber surfaces which could short out the screen accelerating grids as well as the problem of using a toxic, condensible propellant. As a result the program is now focused on inert gas propellants, primarily xenon. JPL modified the 0.3-m J-series mercury ion engines, which had previously been highly developed and were ready for space qualification testing at 2.2 kW, to use xenon. Recognizing that the xenon sputter rates are greater than for mercury and the xenon engine operating powers are 2 to 5 times those for mercury engines, lifetime is a critical issue. An ion thruster system for planetary applications has been designed and is undergoing engineering tests (ref. 73). The focus of these activities is on reducing system complexity, with the goal of reducing the cost of development and flight qualification. Proposed design changes are evaluated in terms of mission impact (ref. 74).

To overcome cathode erosion problems, JPL has been testing a 1.27-cm diameter hollow cathode shown in figure 13 (ref. 75). The cathode was successfully operated for 1200 hr on argon, of which 1050 hr were at 100 A and 24 hr were at 150 A. The cathode was still operating when the testing was terminated; cracking along the tantalum barrel at the upstream end and tungsten deposits which were in the process of closing off the insert bore were two of the problems observed in operating these cathodes at high power. There may have been a cathode jet formed which could have been responsible for the high erosion rate at the upstream side of the baffle. Preliminary data obtained by JPL in 1987 showed that the introduction of nitrogen into the xenon propellant reduced the baffle erosion. Nitrogen may also reduce the erosion of the accelerating grids. The work will be continued (ref. 76). In addition, alternatives to tantalum will be tested as baffle materials and erosion and deposition rates on critical surfaces and deposit structure will be characterized.

Scaling issues. - Results have previously been reported for 30- to 50-cm diameter xenon thrusters at power levels up to 20 kW. However, the only extended operation of a high power xenon ion thruster has been with a 30-cm J-series divergent-field thruster at 10 kW. This thruster had insufficient life for high total impulse mission because of cathode baffle erosion. NASA Lewis has addressed its cathode-baffle erosion, which was running about 0.5 mm/hr, by eliminating the baffle/pole piece and going to the ring cusp design shown in figure 14. This design will require the application of a magnetic field on the order of 500 to 1000 G at the boundary (refs. 77 and 78). Measurements made by NASA Lewis have shown that the erosion rates are very sensitive to the discharge current. In order to understand erosion better, NASA Lewis is developing a model of sputtering.

NASA Lewis has demonstrated the technology to fabricate 0.5-m ion thrusters (ref. 79) as shown in figure 15. Figure 16 shows the ion beam extraction capability of a 0.3-m ion engine and the new 0.5-m ion engine. The 0.5-m

engine doubles the power capability and extends the specific impulse capability. (The thruster efficiency is nearly independent of size and magnetic circuit.)

Early ion thruster grids (ion optics) were flat and were subject to severe, uncontrolled buckling when radial temperature gradients became too steep. The grids were then dished to control buckling. Increasing the thrust of ion engines will require increasing grid size. With the larger grids and potentially noncircular geometries now under consideration, computer modeling has been developed and experimentally verified for understanding structural response (ref. 80).

Engine system technology. - JPL has been working on a simplified system design and improving throttling strategy for planetary mission. As shown in figure 17, JPL has determined that ion engines operating at fixed propellant flow rates can be throttled over an input power range of 3.2:1 with only a small increase in propellant consumed and small decrease in payload delivered as compared to operating the engines at a variable flow rate. The improved throttling strategy seeks to select the most optimum number of operating engines, operational duty cycles, and specific impulse. The systems studies have shown that this optimized throttling strategy results in a reduction in the required engine lifetime of 25 percent to perform the base mission, which was a rendezvous with Comet Kopff, when compared to the conventional throttling technique. Obviously the constant flow strategy eliminates the flow controllers and permits operation at fixed discharge conditions - and it may also simplify the PPU. JPL plans to configure the laboratory PPU for a two-engine operation and operate the two-engine module on a single screen supply. Computer control algorithms for multiengine control will be developed.

NASA Lewis has examined in parametric fashion the impact of various thruster technologies and operating conditions and system technologies on the design and performance of a 10-kW class ion propulsion system for orbital maneuvering and transfer. A methodology is described wherein a computer code is used for precise determination of ion thruster performance (based upon pre-defined and user-defined thruster technologies), a determination of overall propulsion system design including a complete mass breakdown, and first-order calculations of total impulse and delta-V capabilities. Results are obtained from this code to examine the impact of thruster technologies (engine diameter, 2- and 3-grid ion optics, thruster operating conditions (power level propellant, and specific impulse)), and subsystem technologies on system design and mission capability. These analyses were also conducted for arcjet systems (ref. 81).

#### MAGNETOPLASMA DYNAMIC

Multimegawatt MPD thruster technology offers payload gains of a factor of 10 over chemical propulsion for typical orbit raising, lunar base ferry, and manned interplanetary missions. The MPD thruster combines high specific impulse (1000 to 5000 sec) with moderate thrust levels (10 to 100 N). Historically, steady state MPD thrusters have typically been run at or below the few hundred kilowatt range (e.g., refs. 82 to 84), while MPD thrusters operated in a pulsed mode have been run at power levels exceeding 1 MW (e.g., refs. 84 to 87). Propellants have included H<sub>2</sub>, Ar, He, Li, Ne, NH<sub>3</sub>, and N<sub>2</sub>H<sub>4</sub>.

High specific impulse (3000 to 6000 sec) at efficiencies of 30 to 50 percent have been demonstrated using hydrogen (e.g., refs. 88 and 89), though in general thrust efficiencies have not exceeded 30 percent for most propellants (fig. 18). Presently, MPD thruster lifetimes are limited by severe cathode erosion (refs. 90). MPD propulsion is less advanced than ion. Higher efficiencies and specific impulses are generally obtained with applied magnetic fields, but the fundamental theoretical understanding of this mode of operation is lacking. The highest total impulse demonstrated is  $10^6$  N-s, at about 25 kW (ref. 91).

Currently, NASA Lewis is involved in both experimental and theoretical research in the areas of cathode erosion and MPD thruster performance. NASA Lewis has made operational its high-power MPD thruster test stand and successfully demonstrated an operating power of 130 kW in a hybrid MPD/electrothermal mode and 60 kW in the MPD mode (ref. 91) NASA Lewis has extended the magnetic field to 0.3 T and shown reduced anode damage on start-up and an extended stable steady-state operating range as shown in figure 19 (refs. 92 and 93). A high-power MPD thrust stand (ref. 94) and nonintrusive plasma diagnostics (ref. 95) are being developed, and hardware is being assembled for a series of cathode erosion experiments. Codework is being written to simulate the magnetoplasma dynamic interactions (ref. 96), and will be used to evaluate thruster design changes (ref. 90). JPL is also working on such a test stand and both Centers are developing nonintrusive diagnostics for MPD testing. The potential impact of high-temperature superconductivity developments on MPD technology was assessed in a joint effort by Argonne National Laboratory and NASA Lewis (ref. 97).

Work at JPL has demonstrated that cathode evaporation can be minimized by designing for thermionic emission during steady-state operation. This may lead to practical thruster lifetimes ( $>1000$  hr). In the near term, JPL is focusing on demonstrating MPD thruster operation for periods greater than 100 hr. Previously, using the thruster concept shown in figure 20, JPL had completed 85 runs totaling 13 hr of operation, reaching a maximum power of over 72 kW at 2245 A. The engine operated for an 83-min period with no excessive arc spot damage on the cathode. The cathode tip temperature was less than 2240 K for operation up to 1700 A with an argon propellant mass flow rate of 0.16 g/sec.

JPL has observed two distinct steady-state operating modes as shown in figure 21. The first mode is characterized by a luminous cathode jet while the second mode has a cooler cathode tip and no cathode jet. At the same current the terminal voltage is as much as 40 percent lower in the second mode. It may turn out that the second mode could provide the more efficient acceleration of the plasma, due to a more contained radial current pattern. Work is continuing to understand the conditions under which the two modes occur.

As part of its effort to improve MPD thruster performance JPL has designed a radiation-cooled, applied field hybrid engine as shown in figure 22. This will be fabricated and tested. JPL has also been reviewing the literature to determine if designs previously discarded because of the previously existing technology limitations may now be useful. Recognizing that electrode sheaths have a significant effect on MPD thruster operational limitations and performance, JPL is sponsoring the development of a more sophisticated two-dimensional model that will provide a better understanding of MPD thruster operations than current one-dimensional models.

## ELECTRODELESS DISCHARGE THRUSTERS

Electrode thruster concepts offer the potential for very high energy coupling efficiency, and are therefore, of great interest for high power applications (refs. 98 to 104). Limitations imposed by electrode erosion in other electric propulsion thruster designs are potentially avoided.

Currently NASA Lewis is evaluating the microwave electrothermal (MET) thruster (ref. 90), which absorbs applied microwave power in a plasma discharge and heats the gas propellant by high pressure thermalization (fig. 23). Advantages include a high efficiency for power absorption; a high power density; and external control of the discharge location, shape, and volume. The MET thruster should readily scale with power (although performance may not) and it is compatible with high power operation. The current technology MET, being developed in conjunction with Michigan State University, operates at a microwave frequency of 2450 MHz and a power coupling efficiency in excess of 95 percent. Power levels up to 2.5 kW have been tested, and specific impulse values up to 600 sec have been obtained with helium. Recent results show energy efficiencies approaching 65 percent at low power (ref. 105). Gases tested include He, N<sub>2</sub>, H<sub>2</sub>, and O<sub>2</sub>, and a variety of microwave applicator designs have been demonstrated. Present limitations in performance are due to material constraints and MET throat/nozzle design (ref. 90).

Future work at NASA Lewis will focus on power levels of 30 kW at a frequency of 915 MHz, using H<sub>2</sub>, He, and N<sub>2</sub> at discharge pressure up to 10 atm. A magnetic nozzle, produced by a low temperature superconducting magnetic magnet (peak field 5.7 T), will be used to increase the plasma density in the discharge chamber walls. Computer modeling will be used to predict performance with magnetic nozzles, and to compare performance limits with other electrodeless concepts including electron cyclotron resonance (ECR) and ion cyclotron resonance (ICR) heated thrusters (refs. 90, 99, 105, and 106)). JPL is also conducting research on the ECR concept (ref. 104).

## ELECTROMAGNETIC LAUNCHERS

The possible application of electromagnetic launchers to deliver non-fragile materials from the Moon to LEO or other nodes was suggested by the National Commission on Space (ref. 55). Several years ago NASA Lewis completed an investigation of electromagnetic launchers for delivery of nonfragile payloads from Earth into various orbits (refs. 107 to 109). These studies indicated significant cost benefits for Earth-based launchers if the launch rate of appropriate cargo is high enough. Studies have also considered the utility of EML's for the launch of raw materials and nonfragile cargo from the lunar surface (refs. 110 to 113). However the launch rates would have to be higher than those envisioned in any of the exploration case studies before an Earth-to-space rail launcher would be competitive with chemical launchers. Advances in high temperature superconductivity could contribute significantly to the utility of such launchers (ref. 114) and make them the most economical means of bulk transport from the surface of the Moon for such materials as propellants (ref. 115).

## CONCLUDING REMARKS

The ongoing NASA program in low-thrust primary and auxiliary propulsion continues to make great strides. The development of high-temperature chemical thrusters offers order-of-magnitude lifetime improvements and higher specific impulses. The technology for resistojets has been taken to the point where they are baselined for space station. Work is advancing on arcjets leading to a planned flight demonstration. Both ion and MPD thruster technology issues are being addressed with the goal to flight demonstrate these systems so they will be ready for future missions such as a cargo vehicle flight to Mars.

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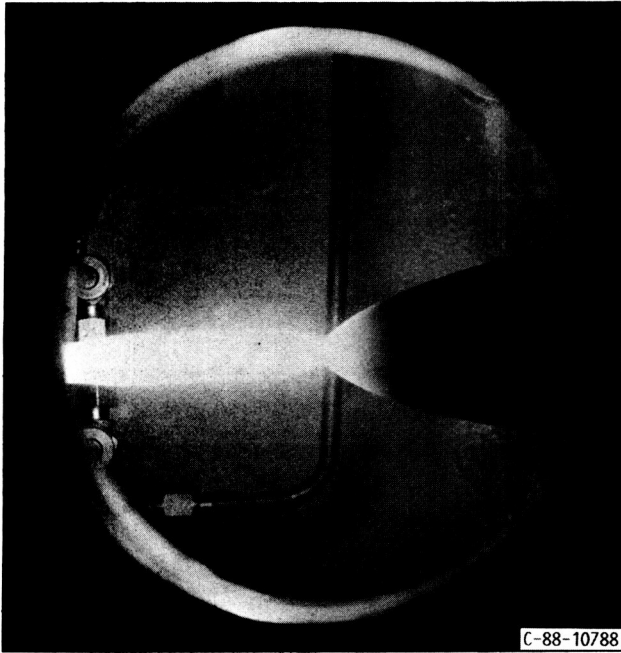


FIGURE 1. - ADVANCED 22-N Bi-PROPELLANT THRUSTER.

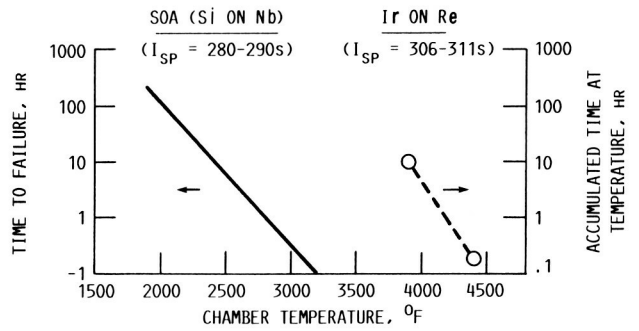


FIGURE 2. - PERFORMANCE AND LIFE EXTENSION PROVIDED BY IRIIDIUM-COATED RHENIUM THRUST CHAMBER.

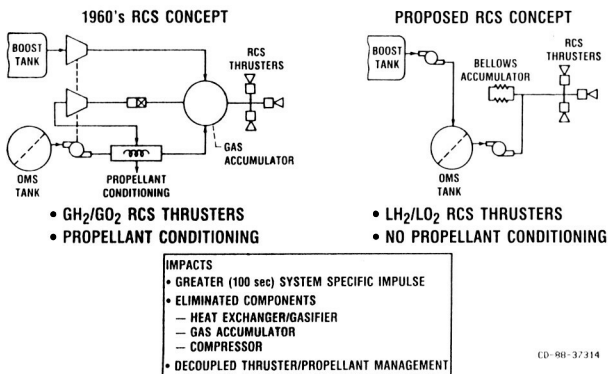
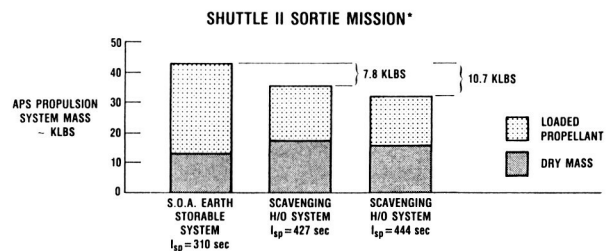


FIGURE 3. - INTEGRATED H/O AUXILIARY PROPULSION SYSTEM.



• A PAYLOAD BENEFIT OF 10,700 lb PROJECTED FOR SHUTTLE II  
• MAY OFFER SIGNIFICANT OPERATIONAL BENEFITS

\*VEHICLE MASS AND MISSION REQUIREMENTS SUPPLIED BY LaRC

FIGURE 4. - PAYOFF OF INTEGRATED H/O PROPULSION.

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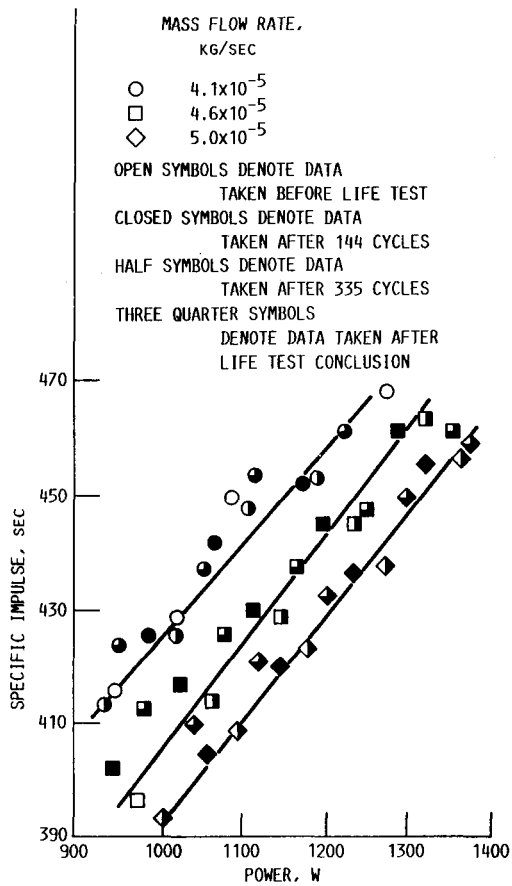


FIGURE 5. - LIFE-TEST 1-kW ARCJET SPECIFIC IMPULSE AS FUNCTION OF POWER.

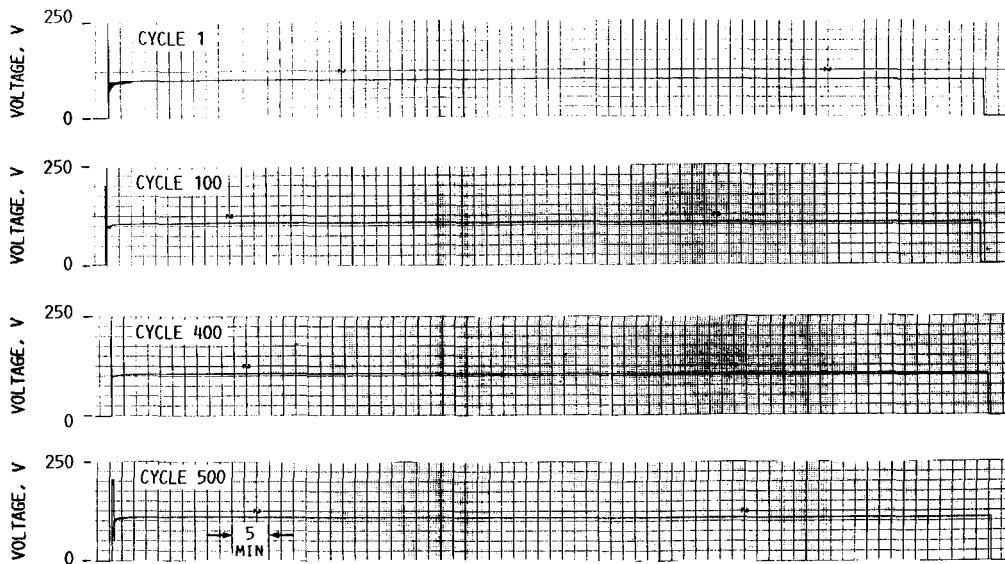


FIGURE 6. - LONG-TERM VOLTAGE TRENDS FOR LIFE-TEST 1-kW ARCJET.



- ELECTRODE CONFIGURATION VERIFIED BY:
  - PERFORMANCE DEMONSTRATIONS
  - LIFE TESTING
- PPU INTEGRATION DEMONSTRATED
- THERMAL INTERFACES MET
- DYNAMIC STRUCTURAL ANALYSIS COMPLETED
- COMMUNICATIONS IMPACTS ASSESSED

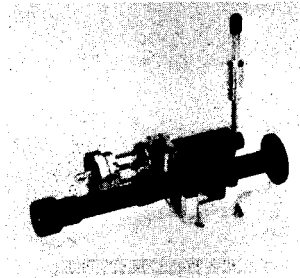


FIGURE 7. - FLIGHT-TYPE, KILOWATT-CLASS ARCJET.

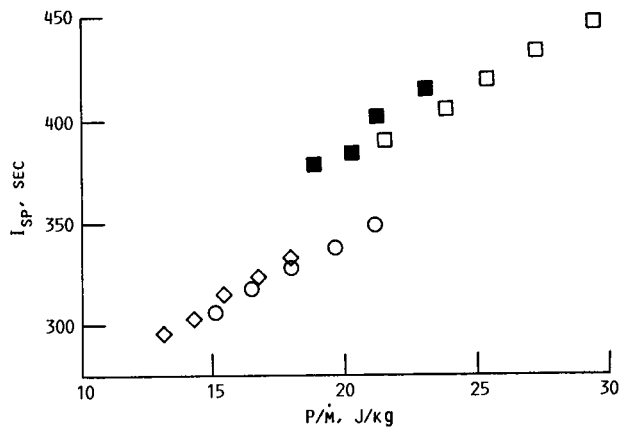
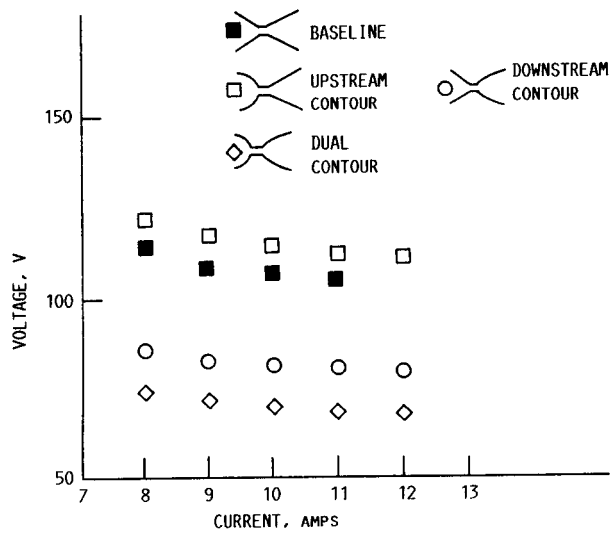


FIGURE 8. - NOZZLE GEOMETRY EFFECTS ON ARCJET CHARACTERISTICS.

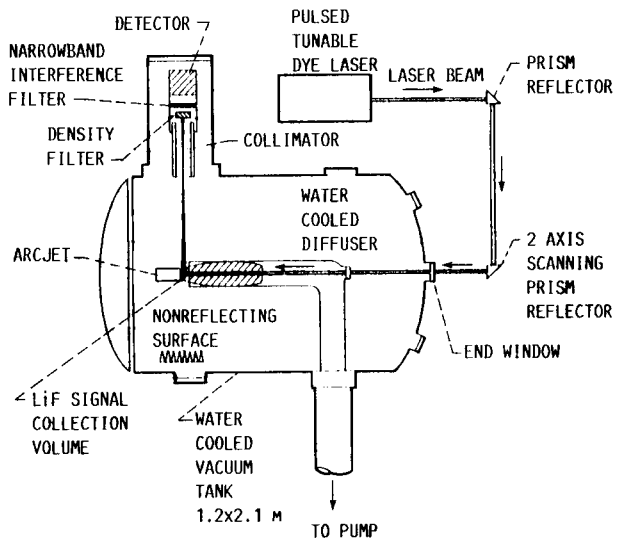


FIGURE 9. - JPL LASER INDUCED FLUORESCENCE (LIF) EXPERIMENTAL SETUP.

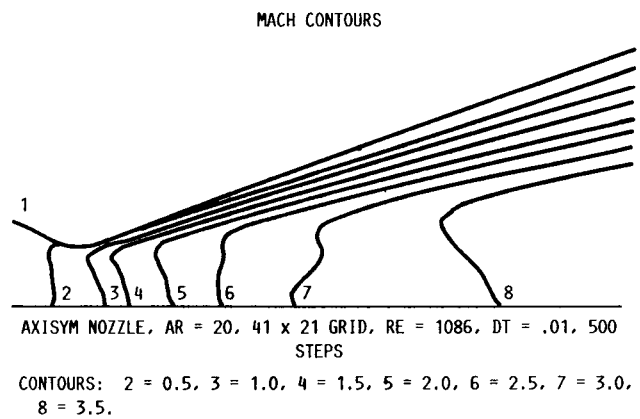


FIGURE 10. - COMPUTED MACH NUMBER CONTOURS FOR LOW REYNOLDS NUMBER CONVERGING-DIVERGING NOZZLE.

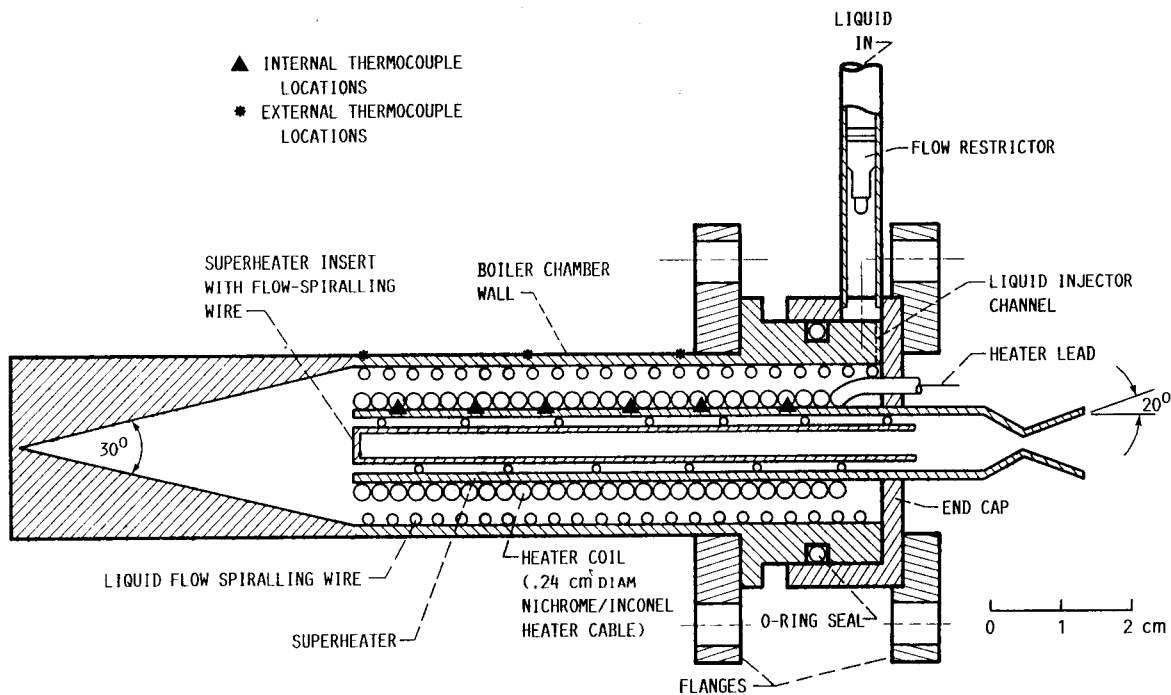


FIGURE 11. - INTEGRATED WATER VAPORIZER/RESISTOJET.

- NEAR TERM PLANETARY/EARTH ORBIT MISSION TARGET  $\approx$  5 TO 30 kW/THRUSTER, Xe
- PATHFINDER CARGO MISSION TARGET  $\approx$  0.1 TO 1 MW/THRUSTER, Ar OR Kr

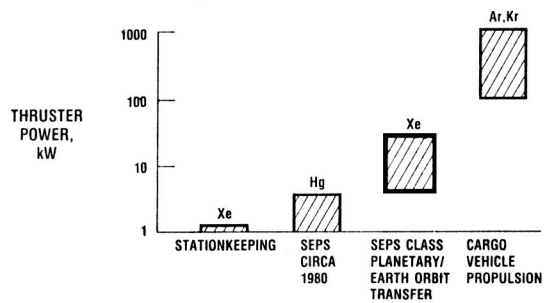
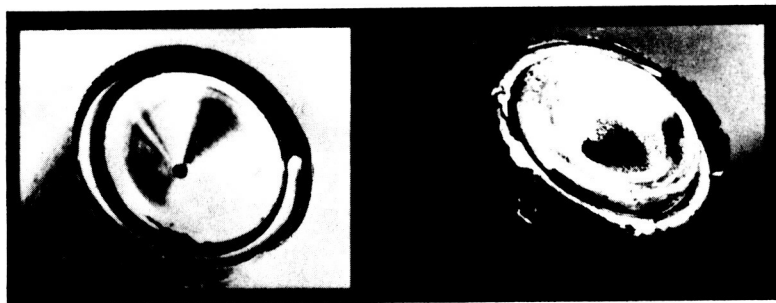


FIGURE 12. - ION ENGINE POWER RANGE FOR VARIOUS MISSIONS.

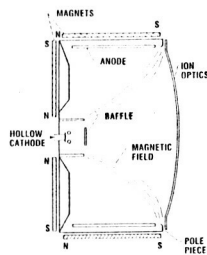


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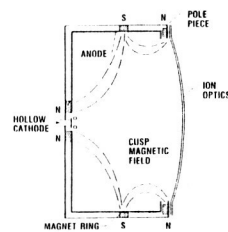
AFTER 170 HR AT 100 A

FIGURE 13. - JPL 1.27 CM DIAMETER HOLLOW CATHODE.

**DIVERGENT FIELD**



**RING CUSP**



- VOLUME B
- BAFFLE FOR DISCHARGE IMPEDANCE CONTROL
- USED FOR SERT I, SERT II, IAPS, AND SEPS

- BOUNDARY MAGNETIC FIELD
- NO BAFFLE

FIGURE 14. - DIVERGENT-FIELD AND RING-CUSP ION THRUSTERS.

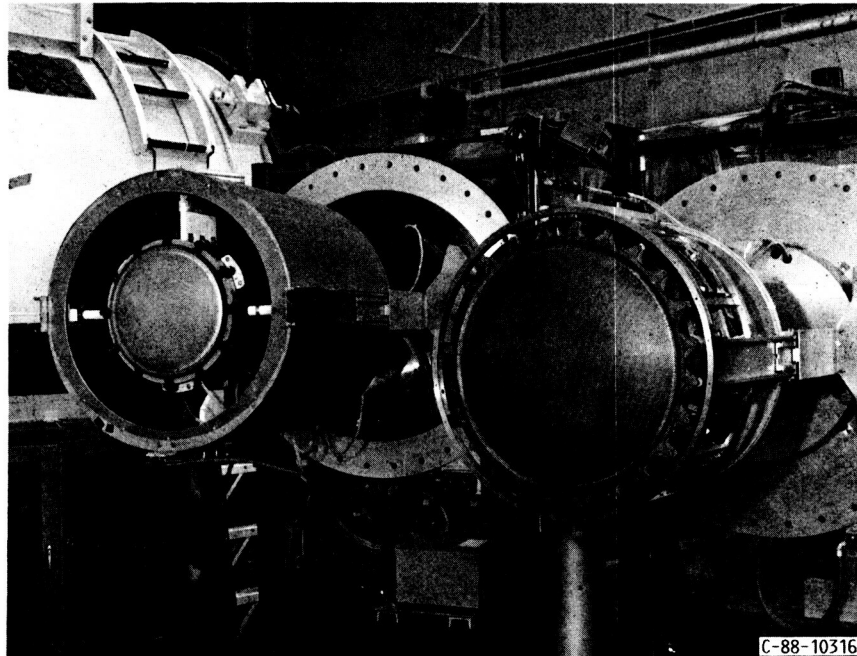


FIGURE 15. - 50 cm ION THRUSTER.

**LARGE AREA ION OPTICS PROVIDE:**

- 2X INCREASE IN POWER OVER 30 cm DIA. THRUSTER
- EXTENDED SPECIFIC IMPULSE CAPABILITY

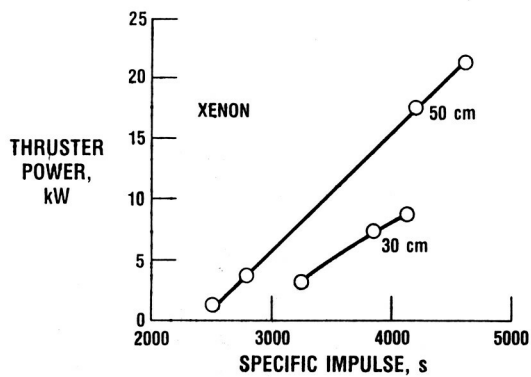


FIGURE 16. - ION EXTRACTION CAPABILITIES OF 30 cm AND 50 cm THRUSTERS.

- CONSTANT FLOW STRATEGY ELIMINATES FLOW CONTROLLERS AND PERMITS OPERATION AT FIXED DISCHARGE CONDITIONS - MAY SIMPLIFY PPU
- ENGINE LIFE REQUIREMENT REDUCED 24% BUT REQUIRES 9% MORE PROPELLANT

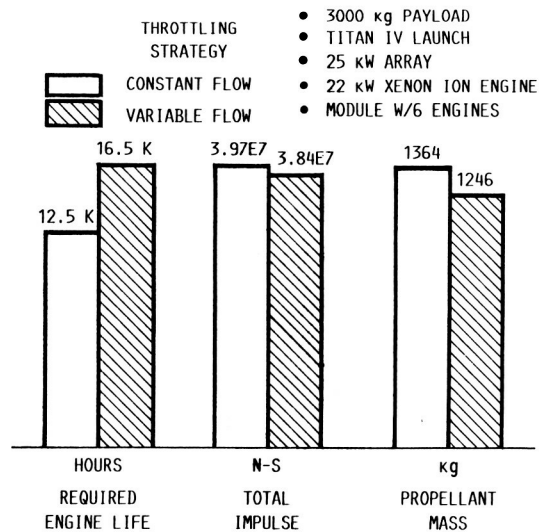


FIGURE 17. - ION ENGINE THROTTLING STRATEGY.

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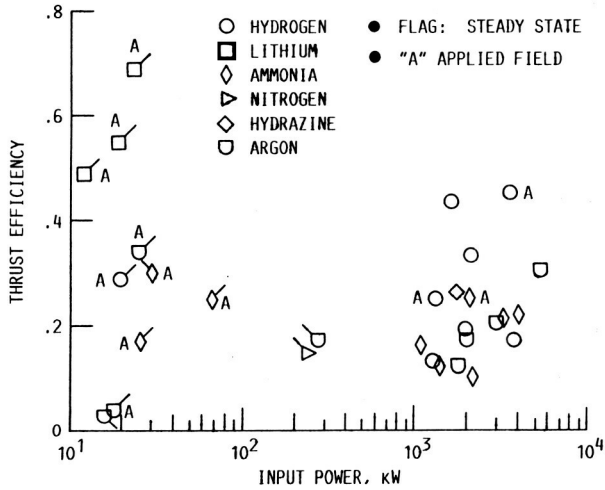
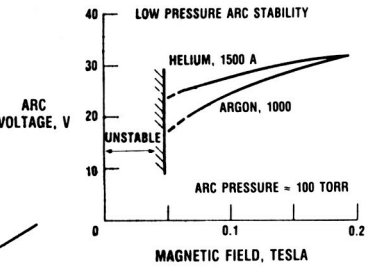
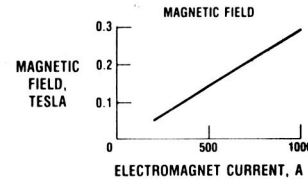
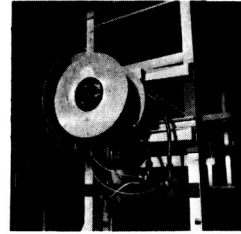


FIGURE 18. - DEMONSTRATED EFFICIENCY FOR MPD THRUSTERS.



- APPLIED MAGNETIC FIELD:
- EXTENDED TO 0.3 TESLA
  - REDUCED ANODE DAMAGE ON START-UP
  - EXTENDED STABLE STEADY-STATE OPERATING RANGE

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FIGURE 19. - OPERATING RANGE FOR LeRC APPLIED-FIELD MPD THRUSTER.

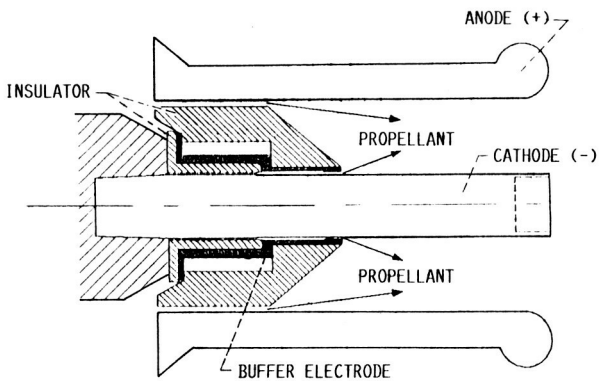


FIGURE 20. - JPL STEADY-STATE, SELF-FIELD MPD THRUSTER.

- DISTINCT STEADY-STATE OPERATING MODES OBSERVED BY D. KING IN 1987
- ONE MODE IS CHARACTERIZED BY A LUMINOUS CATHODE JET. A SECOND MODE HAS A COOLER CATHODE TIP AND NO CATHODE JET; THE TERMINAL VOLTAGE IS LOWER.
- SECOND MODE MAY PROVIDE MORE EFFICIENT PLASMA ACCELERATION

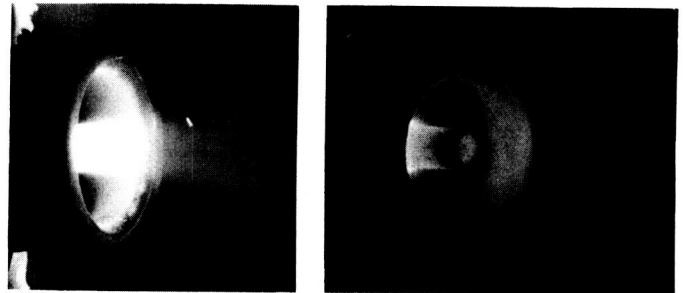


FIGURE 21. - STEADY-STATE, SELF-FIELD MPD MODE CHANGES.

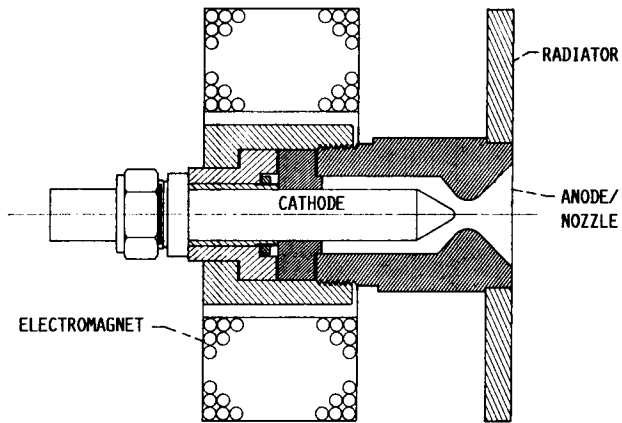


FIGURE 22. - JPL RADIATION-COOLED, APPLIED-FIELD MPD THRUSTER DESIGN.

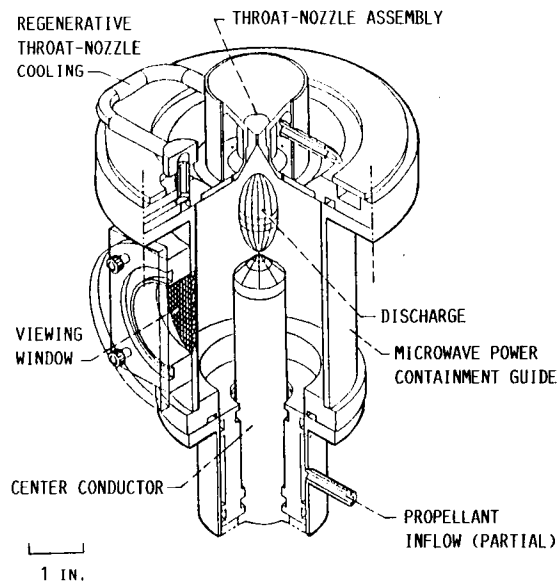


FIGURE 23. - ELECTRODELESS, MICROWAVE PLASMA TORCH THRUSTER DESIGN.



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16. Abstract The NASA OAST Propulsion, Power, and Energy Division supports a low thrust propulsion program aimed at providing high performance options for a broad range of near-term and far-term missions and vehicles. Low thrust propulsion has a major impact on the mission performance of essentially all spacecraft and vehicles. On-orbit lifetimes, payloads, and trip times are significantly impacted by low thrust propulsion performance and integration features for Earth-to-orbit (ETO) vehicles, Earth-orbit and planetary spacecraft, and large platforms in Earth orbit. Major emphases of the program are on low thrust chemical propulsion, both storables and hydrogen/oxygen; low-power (auxiliary) electric arcjets and resistojets; and high-power (primary) electric propulsion, including ion, magnetoplasmadynamic (MPD), and electrodeless concepts. This paper will present the major recent accomplishments of the program and discuss their impacts.					
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