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James S. Sovey, Francis M. Curran, Thomas W. Haag,
Michael J. Patterson, Eric J. Pencil, Vincent K. Rawlin,
and John M. Sankovic
Lewis Research Center
Cleveland, Ohio

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DEVELOPMENT OF ARCJET AND ION PROPULSION FOR SPACECRAFT STATIONKEEPING

James S. Sovey, Francis M. Curran, Thomas W. Haag,
Michael J. Patterson, Eric J. Pencil, Vincent K. Rawlin,
and John M. Sankovic
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

Near term flight applications of arcjet and ion thruster satellite station-keeping systems as well as development activities in Europe, Japan, and the United States are reviewed. At least two arcjet and three ion propulsion flights are scheduled during the 1992-1995 period. Ground demonstration technology programs are focusing on the development of kW-class hydrazine and ammonia arcjets and xenon ion thrusters. Recent work at NASA Lewis Research Center on electric thruster and system integration technologies relating to satellite stationkeeping and repositioning will also be summarized.

INTRODUCTION

Most communication satellites developed in the United States use either monopropellant hydrazine, chemical bipropellants, or electrically augmented hydrazine thrusters for North-South stationkeeping (NSSK). Higher specific impulse electric propulsion systems, employing thermal arcjets or ion thrusters, can provide significant reductions in spacecraft mass; extended on-orbit lifetimes; and in some cases, choices of smaller and less expensive launch vehicles (refs. 1,2). Advanced development programs for arcjet and/or ion propulsion are now being pursued in Europe, Japan, and the United States (U.S.) (refs. 3-5). The National Aeronautics and Space Administration's (NASA's) goal is to develop and transfer the NSSK electric propulsion technology to U.S. government and industry users and also extend this technology to higher power applications such as maneuvering, repositioning, orbit transfer, and planetary propulsion.

NASA's arcjet program has focused on 0.5 to 2 kW hydrazine systems for the NSSK application. The arcjet and power processor system has undergone simulated flight qualification tests, including life tests, well beyond anticipated NSSK requirements (refs. 1,5). The technology has been transferred to industry for the development of 1.8 kW arcjet systems for a series of American Telephone and Telegraph (AT&T) comsats built by the Astro-Space Division of the General Electric (GE) Company. Present technology efforts at NASA involve analytical and experimental studies of arcjet-spacecraft integration issues such as electromagnetic compatibility and more generally, plume interactions with spacecraft. Additionally, storeable propellant arcjets are being evaluated at a few hundred watts for potential lightsat applications and higher power levels for larger near-Earth free-flyers or platforms.

A radio-frequency ion thruster experiment, developed in Germany, was launched in July 1992 using the U.S. Space Transportation System (ref. 6). In 1994, ion thrusters, developed by the Mitsubishi Electric Corporation (MELCO) of Japan, are sheduled to provide an operational demonstration of spacecraft NSSK (ref. 7). In the U.S.. ion thrusters operating in the 0.5 to 2 kW range are being developed at NASA's Lewis Research Center (LeRC) and also by Hughes Research Laboratories (HRL) (refs. 8,9). HRL is developing a 0.3 to 0.4 kW, 13 cm diameter xenon ion thruster for NSSK. The NASA LeRC device is 30 cm in diameter and is operated in a throttled or derated condition to mitigate known life-limiting phenomena.

This paper will summarize some of the near-term flight applications of arcjet and ion thruster stationkeeping systems as well as recent thruster development activities in Europe, Japan, and the United States. The major focus will be on recent work conducted by NASA LeRC on arcjet, ion thruster, and system integration technologies as they relate to satellite NSSK, maneuvering, and repositioning. Thruster physical characteristics, performance data, life projections, and results of plume and electromagnetic compatibility tests will be summarized.

NEAR-TERM FLIGHT APPLICATIONS AND GROUND DEMONSTRATIONS

The major flight qualified electric propulsion systems employ resistojets, ion thrusters, ablative pulsed-plasma thrusters, stationary plasma thrusters, or pulsed magnetoplasmadynamic thrusters (ref. 10). Hydrazine resistojets and the Russian stationary plasma thrusters are the only kW-class electric propulsion devices used operationally either for satellite stationkeeping or orbit correction (refs. 11,12). At least 96 hydrazine resistojets, providing a specific impulse of about 300 s, have been supplied by the Rocket Research Company (RRC) for satellite NSSK (refs. 10,11,13). TRW also developed hydrazine resistojets for NSSK aboard the INTELSAT-V series of spacecraft (ref. 14). The Russians have flown more than 50 stationary plasma thrusters since 1972 on various series of spacecraft such as Meteor, Gorizont, and Ekran (ref. 12). Stationary plasma thruster power levels were in the 0.5 kW to 1.5 kW range. In the very near future hydrazine arcjets and xenon ion thrusters are scheduled to perform NSSK for GE's Series 7000 spacecraft and Japan's Engineering Test Satellite VI (ETS VI) spacecraft, respectively (refs. 1,7) (See Table I). Operational flights, experimental flights, and ground demonstration results of arcjet and ion systems will now be briefly reviewed.

Arcjet Systems

Hydrazine arcjets can provide a 50% to 100% increase in specific impulse over conventional chemical and resistojet systems. In a typical mission, the increased specific impulse would translate into a mass savings of about 100 to 200 kg of propellant. This mass savings could be used to extend the life of the satellite, to increase the payload, or to reduce the launch vehicle class. The mass benefit comes not only from reduced NSSK mass, but also from savings in apogee motor propellant because of the lower spacecraft mass in geosynchronous transfer orbit. The savings in apogee motor propellant is about 60 kg for an INTELSAT VII growth version spacecraft (ref. 15).

Hydrazine arcjets systems have reliably demonstrated specific impulse levels up to 520 s, and such devices have been flight qualified for GE's Series 7000 communication satellites (ref. 1). Laboratory model, engineering model, and flight thrusters (Fig. 1) have been developed by NASA LeRC and RRC. Extensive ground tests were conducted including at least six life demonstration tests (ref. 5). Early testing indicated that the gas generators developed for resistojets would not meet anticipated arcjet system qualification life requirements. Post test component evaluation revealed that failures were attributable to excessive temperatures of the capillary injector tubes and concomitant deposition of non-volatile residuals. A thermal redesign of the injector region by RRC was successful in significantly reducing injector temperatures, and flight units have been developed and tested well beyond required qualification life for the GE Series 7000 spacecraft. An arcjet power conditioning unit (PCU) was developed to operate from a battery system with input voltages from 96 V to 65 V (ref. 1). The PCU incorporated a "soft start", pulse starting circuit which was based on early breadboard PCU's tested at NASA LeRC (ref. 16). The PCU, packaged for flight, had a mass of 4.2 kg, and the interconnecting power cable mass was 0.8 kg (ref. 1). The PCU efficiency was between 91% and 94% implying less than 180 W of thermal power had to be rejected. Heat rejection by the thruster to the spacecraft was estimated to be less than 10 W (ref. 1). Arcjet subsystem masses and performance parameters are shown in Table II.

The GE/RRC arcjet system has undergone thermal-mechanical qualification tests, cyclic-life tests, plume impact tests, as well as thermal loading and contamination experiments (refs. 1,5). The thruster qualification test program included acceptance vibration tests, functional tests, performance maps, life tests, flow interruption tests, and post-test inspections. A qualification life test successfully demonstrated 891 h of cyclic operation with a profile similar to that expected in on-orbit operation. The total impulse demonstrated in this test was 685,000 Ns (ref. 1).

As shown in Table III technology efforts using laboratory model hydrazine arcjets are underway at Japan's Institute of Space and Astronautical Science (ISAS) and at Osaka University. The ISAS arcjet, operating at power levels from 1.3 kW to 1.9 kW, has obtained a specific impulse in excess of 600 s using hydrazine decomposition products (ref. 4). Studies are also ongoing at ISAS to examine arc ignition reliability consistent with low electrode erosion. At Osaka University, a kW-class arcjet using hydrazine decompositon products was successfully operated in the 400 s to 550 s specific impulse range and tested for 50 h to assess electrode erosion rates. Projections from erosion and performance diagnostics indicate the 1-kW arcjet could operate at 500 s specific impulse for over 1000 h with 1000 restarts (ref. 17).

Stationkeeping class arcjet work is also also ongoing at Italy's BPD Difesa e Spazio and Centrospazio (CS) as well as Germany's University of Stuttgart and Messerschmitt Bolkow Blohm (MBB) (ref. 3). Typical performance of these thrusters is shown in Table III. At BPD, a low power laboratory arcjet, using hydrazine decomposition products, demonstrated a specific impulse of about 440 s. Activities at CS focus on arcjet modeling, power processor design, and thruster performance parametrics. At BPD, parametric performance and endurance testing will be performed with catalytically decomposed hydrazine and simulated hydrazine decomposition products (refs. 18,19).

At the University of Stuttgart and MBB, a kW-class hydrazine arcjet, gas generator, and power processor are under development (refs. 20,21). Pre-engineering models of the arcjet and gas generator have been developed and tested. A decomposed hydrazine gas mixture was preheated to simulate the output of the gas generator, and a specific impulse of about 520 s was obtained at 1.2 kW (ref. 20). Later, the thruster and an engineering model hydrazine gas generator were integrated and tested for periods up to 3 h (ref. 21). A flight demonstration of a 0.7 kW version of the arcjet is planned to be performed on an amateur radio satellite, AMSAT P3-D, using ammonia propellant (ref. 21) (See Table I). The arcjet will provide about 20% of the 500 kg spacecraft's delta-V requirement for orbit positioning and stationkeeping.

Ion Thruster Systems

As shown in Tables I and II there are one flight experiment and two operational NSSK demonstrations of xenon ion propulsion scheduled in the next four years (refs. 7,22,23). The European Space Agency has sponsored development of ion propulsion systems for the European Retrievable Carrier (EURECA) and the Advanced Relay and Technololgy Mission (ARTEMIS) communication satellite. EURECA is a 4000 kg platform that was placed in a 580 km circular orbit by the U.S. Space Shuttle in July 1992 (ref. 24). Germany's Radiofrequency Ion Thruster Assembly (RITA) is a platform experiment to demonstrate the operational use of ion propulsion, compare space and ground test performance, and obtain operational experience onboard a spacecraft. The RITA includes a 0.44 kW xenon ion thruster operating at about 3300 s specific impulse with a design life greater than 1700 h with 1082 cycles (ref. 24). Japan's National Space Development Agency (NASDA) chose to develop a xenon ion propulsion system for NSSK for the 2000 kg ETS VI satellite which is scheduled for launch in 1994 (refs. 4,7) (See Tables I and II). The 12 cm diameter ion thrusters each provide 23 mN thrust and about 2900 s specific impulse at an input power of ~0.6 kW. Thruster design life is about 6500 h and NSSK mission life is 10 years. In 1991 performance, thermal vacuum, electromagnetic compatibility, vibration, and acoustic tests were performed on protoflight models (ref. 7). Preliminary results indicate there were no serious obstacles to the development of flight systems (ref. 7). Also in 1991 six thruster life tests were in progress with demonstrated thrusting times up to 7160 h.

The ARTEMIS, an experimental communications satellite, is scheduled for launch in 1995. North-South stationkeeping ion propulsion systems will be provided by Germany's RITA and the United Kingdom's UK-10 ion thrusters (refs. 22,23) (See Tables I and II).

As indicated in Table IV, Hughes Research Laboratories, NASA LeRC, and the National Aerospace Laboratory (NAL) of Japan have been involved in the development and ground-based demonstration of kW-class ion propulsion for NSSK (refs. 8,9,25-28). In 1987 a xenon ion propulsion system (XIPS) was developed by HRL (with INTELSAT support) and ground tested (with NASA support) for 4350 h with 3850 on-off cycles (ref. 25). This test simulated over 10 years of NSSK for a 2500 kg class communications satellite. The XIPS thruster was 25 cm in diameter and produced about 62 mN of thrust with an input power of 1.3 kW. The XIPS power processor was designed for a 28 V to 35 V bus and had seven outputs for thruster startup and operation. The mass of a flight packaged power processor was estimated to be about 10 kg with an efficiency of 90% and a parts count of about 500 excluding telemetry (ref. 25). More recently HRL developed a 13 cm diameter XIPS for NSSK (ref. 9). This version produced about 18 mN of thrust and 2600 s specific impulse with an input power of 0.44 kW. The thruster power processor contained only 400 parts in the seven power modules which include screen, accel, discharge, two keeper supplies, and two heater supplies. Xenon tankage fraction was estimated to be 12% at a storage pressure of 7.6 MPa (1100 psi). HRL has built two qualification test model thrusters and breadboard model power processors. Thrusters will undergo performance and vibration testing prior to extended life testing scheduled for late 1992.

Japan's NAL is also developing a kW-class xenon ion thruster for NSSK applications with a view to improve thruster reliability and lifetime (ref. 28). A 0.6 kW, 14 cm diameter ring-cusp thruster was developed to provide about 25 mN thrust and reliable operation for periods from 6000 h to 8000 h. Wear tests of 1000 h and 1859 h were conducted to evaluate hollow cathodes and determine erosion rates of the positive and negative grids. Test results indicated that the erosion of the positive grid was negligible, hollow cathodes required further optimization to insure long life, and the negative grid erosion rates were unacceptably high in part due to the high negative grid voltage of 800 V. Improved thruster ion optics endurance will likely come from reduced magnitude of the negative grid voltage and improved propellant efficiency which would reduce the flux of charge exchange ions impinging on the negative grid.

At NASA LeRC, 30 cm diameter xenon ion thrusters are being developed for NSSK and primary propulsion applications (ref. 8) (See Fig. 2). For the NSSK application, the focus is on power levels of 0.5 kW to 2 KW. To optimize the expectations for implementation of ion systems for NSSK, the 30 cm thruster, initially developed for primary propulsion, is operated at a fraction of its design and demonstrated power level. The derated xenon thrusters have provided specific impulse levels of 1700 s to 2500 s at overall efficiencies from about 45% to 60% (ref. 29). Ion thrusters being developed for NSSK under other programs are generally small compared to the 30 cm design and operate near both thermal and ion current density limits (ref. 8). The advantages of using this derated approach include the elimination of known life-limiting issues, increased thrust-to-power ratio, and reduced flight qualifications times. Detailed results of thruster performance tests, thruster design optimization, and life projections are addressed in the following section.

THE ARCJET AND ION THRUSTER DEVELOPMENT PROGRAMS AT NASA LERC

In recent years the NASA arcjet and ion propulsion technology programs have been primarily directed toward the development and technology transfer of low power propulsion systems for satellites in geosynchronous and low-Earth orbits (ref. 30). The NASA program involves in-house, university, and industrial development of propulsion system components, system development and integration, and fundamental research to better understand plasma processes, electrode phenomena, and plumes.

Arcjet Systems

Over the last nine years, NASA LeRC has maintained a program to develop kW-class hydrazine arcjets for NSSK of geosynchronous spacecraft. The LeRC in-house effort is focused on improved understanding of fundamental physical phenomena associated with arcjet operating characteristics as well as developing reliable power processors and providing information necessary for the successful development and integration of flight-type systems. Testing at LeRC has primarily been conducted using hydrogen and nitrogen mixtures to simulate the decomposition products of hydrazine. Initially, arc ignition and transition to steady-state operating conditions, using ballasted DC power supplies, were not well controlled, and significant electrode erosion was observed. These difficulties were successfully overcome by changing the electrode geometry and providing stronger flow stabilization as well as incorporating a pulse-width modulated power processor with high voltage pulse starting and a circuit to limit the current transient during start-up (refs. 16,31). Since 1985 parallel programs were conducted at LeRC and RRC to demonstrate the reliability and flight-readiness of kW-class hydrazine arcjets. At LeRC a 1000 h/500 cycle lifetest of a modular laboratory arcjet subsystem demonstrated long-term, reliable, non-damaging arcjet operation (ref. 32). In 1988, the GE Astro-Space Division (ASD) sponsored a hydrazine arcjet development program with RRC resulting in the test of two engineering model arcjet thrusters and gas generators for 1258 h and 870 h with 183 and 900 arc startups, respectively (ref. 33). RRC also conducted an 891 h qualification lifetest of a 1.8 kW hydrazine arcjet with 918 restarts and a specific impulse of 520 s (ref. 1). Post-test examination of the thruster and hydrazine gas generator revealed no phenomena that would preclude a thruster total impulse capability in excess of 654,000 Ns which was the qualification test requirement.

Much of the LeRC effort has been directed toward evaluating the integration of arcjets with spacecraft. Work is being conducted to assess the impacts of the partially ionized arcjet plume on communication signals; to examine the impacts of conducted and radiated emissions from the thruster subsystem; and to address user concerns such as contamination, thermal and momentum exchange, and radiated energy. To understand the effects of a partially ionized (<1%) plume on communication signals, the electron number densities and temperatures in the plume were measured using electrostatic probes (refs. 34-37). These data were used in a source flow model to estimate the far-field plume characteristics (ref. 37). The plasma was modeled as a slab to estimate phase shift and attenuation of a 4 GHz communications signal running parallel to and intersecting the plume centerline. For realistic propagation paths, first order analyses have indicated negligible impacts on signal transmission (refs. 37,38).

An experimental study of the spacecraft compatibility of operational arcjet systems was performed by TRW, under contract to NASA, using a FLTSATCOM qualification model spacecraft in a large space simulation chamber (ref. 39) (See Figure 3). Measurement of radiated and conducted electromagnetic emissions revealed that radiated emissions from the arcjet and its power processor were within acceptable limits above 500 MHz which indicated conventional communication links at S-band and higher frequencies would not be affected by the kW-class arcjet system. Broadband noise exceeded the tailored limits for communication satellites below 40 MHz. FLTSATCOM telemetry was monitored during the arcjet firings, and no changes in signals were attributed to the thruster system. Six calorimeters were located between 1.8 m and 2.3 m from the thruster exit plane. The maximum heat flux was equivalent to 0.18 suns which was considered satisfactory for thermally integrating the arcjet with most spacecraft (ref. 39). As expected, witness plates, located in the vicinity of the arcjet and on the spacecraft solar array, revealed no evidence of material deposition.

A joint test program, under a NASA Space Act Agreement, was established between LeRC, GE/ASD, and RRC to assess arcjet-spacecraft integration issues such as the compatibility of arcjet plumes with spacecraft materials, spacecraft charging, and electromagnetic compatibility (ref. 40). Test samples included both indium-tin oxide coated and uncoated optical solar reflectors, a 4 X 4 solar cell array, a thermal blanket, and various paints. Sample mounting plates were placed in NASA LeRC's 4.6 m diameter vacuum chamber and located relative to the arcjet to simulate the actual position on a spacecraft. A schematic of the test set-up is shown in Figure 4. Uncharged samples were immersed in the arcjet plume for about 40 h. Results indicated the plume had little impact on surface electrical or optical properties. The solar cells and optical solar reflectors were charged to about -10 kV , and paint samples were charged to about -500 V using a 20 keV electron beam. After exposure to the arcjet plume, the magnitudes of the potentials decreased benignly to ground potential in less than one second implying the arcjet might be used as a spacecraft charge control device. Radiated emissions were examined in various frequency ranges including the UHF, S, C, Ku, and Ka bands. With a receiving system sensitivity within MIL-STD-461 specifications, no electromagnetic interference (EMI) signals were detected in any of these ranges. However, like the TRW spacecraft compatibility tests, low frequency (< 10 MHz) incoherent broadband noise exceeding MIL-STD-461 C specifications was observed.

Other LeRC in-house test efforts are focused on increasing the power and specific impulse of the arcjet to 2 kW and 650 s, respectively, using hydrogen/nitrogen mixtures to simulate hydrazine decomposition products. A 300 h test at 550 s specific impulse was completed with no degradation in thruster performance (ref. 41). At the end of the test the cathode tip recession was found to be about 0.8 mm, and a 1.4 mm diameter crater was formed at the end of the cathode. Although the anode sustained no significant damage, further development is required to optimize arc current, cathode design, and mass flow parameters to insure a long-lived cathode. Using an advanced arcjet design and simulated hydrazine decomposition products, a specific impulse of 690 s was obtained at 2 kW for over 30 minutes without nozzle degradation. A non-erosive startup technique at the low flowrates, required for very high specific impulse operation, needs to be developed before lifetesting can be initiated.

A single, flight-type 1.3 kW arcjet was tested at both LeRC and RRC (ref. 42). Test objectives were to compare the performance at both facilities, to compare performance obtained with hydrazine and gaseous $N_2 + 2H_2$, and to examine background pressure effects on performance. Results indicate that at comparable test facility background pressures, the specific impulses measured at both facilities using $N_2 + 2H_2$ gaseous propellant agreed to within 1% over the 1.6:1 range in flow rate tested. The measured specific impulse using hydrazine and $N_2 + 2H_2$ propellants agreed to within 1.5% when an enthalpy correction was used to account for the hydrazine gas product temperature of ~800 K at the arcjet inlet. Measured specific impulse showed a strong dependence on background pressure and was 3% to 4% higher below 0.1 Pa than for background pressures greater than 5 Pa. This effect is now under study and is believed to be related to convection effects and/or changes in arc anode attachment with variations in pressure.

There are a number of power limited spacecraft, including low-Earth orbit communications satellites (ref. 43), which might derive significant benefits by using low power (0.1 to 1 kW) arcjets for orbit maintenance. A program to develop these low power arcjets has been ongoing at LeRC since 1989 (refs. 44,45). A preliminary investigation was conducted to determine the low power limit of arcjets utilizing simulated, fully decomposed hydrazine as the propellant. Performance data were taken at powers as low as 0.24 kW. Specific impulses between 360 s and 440 s were obtained at conservative specific energy levels and power levels ranging from 0.4 kW to 0.7 kW (ref. 44) (See Figure 5). It was found that the arc constrictor diameter, when varied from 0.38 mm to 0.64 mm, had little effect on performance. Over the 0.4 kW to 0.8 kW power range, specific impulse varied linearly with input power at constant flow rate implying a decrease in thrust efficiency with increasing power. Work is ongoing to examine the sensitivity of performance to power; to extend the power operating envelope to ~ 0.1 kW; increase specific impulse; and to understand fundamental parameters required for stable, reliable operation. Pulse-width modulated power electronics for a 0.2 to 0.4 kW arcjet were developed and integrated with a thruster (ref. 45). The power processor employed a full-bridge circuit switching at 8 kHz to minimize switching and transformer core losses. The arc was started using a train of 2.8 kV-30 microsecond pulses. The power supply had an output filter that included a 27 mH inductor which resulted in an acceptable current ripple of about 20 percent (refs. 16,45). The breadboard power processor, operating from a power bus of nominally 28 V, had an efficiency greater than 92% over the power operating range using a resistive load. Non-damaging arcjet starts and transitions to steady-state operation were demonstrated at input powers as low as 0.24 kW.

The low power arcjet effort has an outreach program that provides hardware and technical assistance to other institutions. Kilowatt class arcjet systems have been loaned to Stanford University, the University of California, the Aerospace Corporation, the University of Tennessee Space Institute, and the University of Illinois.

Ion Thruster Systems

At LeRC, much of the recent efforts are focused on the development of 30 cm diameter xenon ion thruster system technology for both auxiliary and primary propulsion applications in the 0.5 to 5 kW power range (ref. 30). To optimize the expectations for implementation of ion propulsion systems for stationkeeping, a low-risk, derated 30 cm thruster option is being pursued (ref. 8). This approach differs from other smaller NSSK ion thrusters which include the 12 cm MELCO (ref. 7), 14 cm NAL (ref. 28), 10 cm UK-10 (ref. 23), 10 cm RIT-10 (ref. 6), 15 cm RIT-15 (ref. 6), and 13 cm HRL thrusters (ref. 9). By operating at relatively low thrust densities, the derated 30 cm thruster virtually eliminates life-limiting issues. Performance data have been obtained, using the xenon derated thruster, over a 33:1 power range and 4.5:1 range in specific impulse (ref. 8). Detailed performance mapping was undertaken for operation in the specific impulse range of 1000 s to 3000 s since there may be mission enhancing benefits to power limited spacecraft in this range of performance (refs. 8,26). It is well-known that xenon ion thrusters operate efficiently at specific impulses greater than 3000 s, but little reported data exist at the very low specific impulse levels. Figure 6 shows typical xenon thruster performance in the low specific impulse range. Thruster efficiencies at specific impulses of 1500 s and 3000 s were about 40% and 66%, respectively. Thrust-to-power levels in the 50 to 57 mN/kW range were obtained over a range of specific impulse from 1200 s to 2700 s (ref. 26). Because of present limitations on ion optics' performance, the thruster maximum input power using xenon varied from about 1 kW at 1500 s specific impulse to more than 3 kW at 3000 s specific impulse. At a given input power, the derated 30 cm thruster operates at thrust levels 25% to 80% higher than that obtained with smaller flight-type ion thrusters. The higher thrust capability implies reduced on-orbit firing times and reduced ground qualification test times.

Since the derated thruster operates at low ion current densities, low discharge voltages, and low accelerating voltages, the thruster life and reliability are enhanced because of lower internal and external component erosion rates. The derated ion thruster positive and negative grid erosion rates have been estimated to be at least 16 and 41 times lower than those of smaller NSSK thrusters operating at the same input power of 0.64 kW (ref. 8). Calculations using negative grid erosion rates, beam area, and required thrusting times predict about 10 to 20 times lower sputtered efflux from the the negative grid of the 30 cm thruster as compared to smaller 2-grid thrusters. For example, using the life-limit rationale developed in References 8, 27, and 29, the ion optics and hollow cathode projected lifetimes of the 30 cm thruster easily exceeded 10,000 h at power levels of 0.64 kW, 1.6 kW, and 5.5 kW when the specific impulses were > 1500 s, > 2200 s, and > 3800 s, respectively.

A potential disadvantage of the derated thruster approach for NSSK is thruster integration on mass and volume constrained spacecraft. The 30 cm thruster is larger and more massive than the small, present generation ion thrusters which range in mass from about 1 to 5 kg (ref. 29). A recent study of satellites using derated ion thrusters for NSSK indicated the satellite mass in geosynchronous transfer orbit decreased by approximately 17 kg for each kilogram reduction in thruster mass (ref. 15). This strong sensitivity occurs because there are four thrusters per NSSK system, each with a gimbal assembly whose mass was estimated to be 34% of the thruster mass. In addition, the reduced thruster and gimbal masses require less structure, contingency mass, and propellant for NSSK, attitude control, and orbit transfer. The need for gimballed NSSK thrusters will be spacecraft specific and will ultimately be based on tradeoffs between propulsion module mass and attitude control system complexity and/or propellant mass.

Design modifications were made to the baseline 30 cm laboratory thruster whose mass was 10.7 kg (ref. 29). In 1992, most of the mild-steel and stainless steel components were replaced with aluminum; the number and size of magnets were reduced, and the cylindrical design was replaced by a conic geometry constructed primarily from aluminum (Figure 2). The thruster will soon undergo diagnostic vibration tests along three axes at sinusoidal levels of 0.5 g and 1 g. The thruster mass estimate including internal wire harness, propellant isolators, neutralizer, and mounting pads is between 6 kg and 7 kg.

Additionally, the LeRC program includes the development of major thruster components such as ion optics, hollow cathodes, and neutralizers. In an ion optics investigation, nine ion accelerating systems were diagnosed to understand and extend the limits of ion extraction capability (ref. 46). Increased ion extraction will enable increased thrust density which is particularly important for very low specific impulse NSSK ion thrusters. Grid hole pair misalignment, due to electrode forming or intentional offsets for beam vectoring, was found to be the major limiting factor to enhanced ion extraction capability. Ion extraction capabilities improved by as much as 90% when the only change made was to insure alignment of the roll direction of the molybdenum sheets prior to forming the dished configuration. The grid system ion extraction capability increased with decreasing values of the ratio of discharge voltage to total accelerating voltage. This phenomenon is the likely reason that the impingement limited beam current from large area ion optics increased with total accelerating voltage faster than the three-halves power as predicted analytically. The dimensions of ion beamlets, exiting the negative grid of a 30 cm diameter system, were measured as a function of radius. At the ion extraction limit, only the central 20 percent of the negative grid area showed evidence of ion impingement. Thus, if all hole pairs were aligned, the ion extraction limit would simply be dictated by the ion density profile uniformity which has an impact on thrust density. In addition, operation with xenon, krypton, and argon propellants led to impingement limited ion extraction values which increased inversely as the square root of the propellant mass as expected from theoretical considerations (ref. 47).

At LeRC, very encouraging results have been obtained showing that hollow cathode degradation due to oxygen contamination can be mitigated by developing criteria and procedures to ensure long-life cathodes. In this effort, three hollow cathodes have been wear-tested for periods of about 500 h each (ref. 48). Operational parameters and post-test microanalyses were documented. It was found that by employing a feed-line bake at 75 °C, reducing the propellant feed system leak/outgas rate to ~4 X 10^{-6} Pa-l/s, and using a gas purifier, the internal surfaces of the hollow cathodes showed an insignificant amount of material deposition, and overall operational reliability improved. Very small, highly localized amounts of tungsten, barium and calcium compounds, and Ba_2CaWO_6 were found on internal cathode surfaces, but none of these deposits impacted performance over the 500 h period. The discharge voltage changed by less than 2% during the course of the 500 h test, and the cathode tube temperature decreased from a high of 1090 °C to a low of 1025 °C. Research to develop detailed criteria for long-life, inert-gas hollow cathodes is continuing.

A series of xenon neutralizer performance diagnostic tests were completed at LeRC (ref. 49). It was found that the plasma screen surrounding the ion thruster should be isolated from facility ground, in order to insure that neutralizer electrons couple directly to the ion beam and do not find a return path via the plasma screen. Tests also indicated that stray thruster magnetic fields in the region of the neutralizer cathode could significantly degrade coupling to the ion beam. Further, an optimized xenon neutralizer required a xenon flowrate of about 9% of the total flow rate for thrusters operating in the 0.55 to 3.2 kW input power range. State-of-the-art xenon neutralizers generally require about 15 W to 20 W of input power per ampere of electrons emitted, and the ratio of neutralizer electron to neutral atom flowrate ranged from 15 to 35.

Although ion thruster power processor breadboards (PPB's) are not presently being developed at LeRC, the PPBs developed for arcjets use switching topologies and circuit integration methods that are applicable to the next generation ion thruster PPBs. In the area of component development, the University of Wisconsin, under grant to LeRC, is developing lightweight coaxial power transformers for higher power PPBs (ref. 50). Since the mass of NSSK ion system power processors is driven by magnetics mass, this new transformer technology may have a significant impact on future systems.

Under an outreach program, the lightweight thrusters, power consoles, and propellant management systems are being assembled for delivery to user organizations to familiarize them with the technology. The ion propulsion technology has also been transferred to the Space Station Freedom program for the development of plasma contactors which control spacecraft potential and eliminate arcing to structural components.

CONCLUDING REMARKS

Arcjet and ion propulsion development and flight programs for spacecraft stationkeeping are now being pursued in Europe, Japan, and the United States. The first operational arcjet and ion thruster NSSK systems are planned to be flown in the 1993 to 1994 timeframe on GE's Series 7000 and Japan's ETS-VI spacecraft, respectively. Since most spacecraft have power capabilities less than 5 kW, most of the electric propulsion opportunities for the next 10 years will likely involve stationkeeping, maneuvering, and repositioning of geosynchronous and low-Earth orbit satellites. At least two arcjet and three ion propulsion flights are scheduled during the 1992-1995 period. Ground demonstration technology programs are also focusing on the development of 0.2 to 1.8 kW hydrazine and ammonia arcjets and xenon ion thruster systems for power limited spacecraft. The low power arcjet work involves fundamentals of arc stability and requisites for reliable, long-life operation. Ion propulsion technology efforts focus on reduced complexity of the thruster and power processor system, lower system mass, reduced cost, and increased lifetime. In both electric propulsion disciplines, integration technology work is ongoing to understand spacecraft compatibility issues related to potential plume contamination from electric thrusters, thrust losses due to plume impingement on the spacecraft, electromagnetic compatibility, and impact of plumes on up- and down-link communications.

REFERENCES

1. Smith, W. W., et al., "Low Power Hydrazine Arcjet Flight Qualification," IEPC Paper 91-148, presented at AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

2. Schrieb, R., "Readiness Appraisal: Ion Propulsion for Communication Satellites," AIAA Paper 88-0777, March 1988.

3. Bartoli, C., "Review of European Activities on Electric Propulsion," IEPC Paper 91-001, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

4. Yoshikawa, T., "Electric Propulsion Research and Development in Japan," IEPC Paper 91-004, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

5. Curran, F. C., Sovey, J. S., and Myers, R. M., "Electric Propulsion: An Evolutionary Technology," IAF Paper 91-241, October 1991.

6. Bassner, H., et al., "Status of the RITA-Experiment on EURECA," IEPC Paper 88-029, Proceedings of the DGLR/AIAA/JSASS 20th International Electric Propulsion Conference, October 1988, pp. 180-185.

7. Shimada, S., et al., "Ion Engine System Development of ETS-VI," IEPC Paper 91-145, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

8. Patterson, M. J. and Foster, J. E., "Performance and Optimization of a Derated Ion Thruster for Auxiliary Propulsion," AIAA Paper 91-2350, June 1991.

9. Beattie, J. R., Robson, R. R., and Williams, J. D., "18 mN Xenon Ion Propulsion System," IEPC Paper 91-010, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

10. Sovey, J. S., et al., "The Evolutionary Development of High Specific Impulse Electric Thruster Technology," AIAA Paper 92-1556, March 1992.

11. Feconda, R. T. and Weizman, J. I., "Satellite Reaction Control Subsystems with Augmented Catalytic Thrusters," AIAA Paper 84-1235, July 1984.

12. Bober, A. S., et al., "State of Work on Electrical Thrusters in USSR," IEPC Paper 91-003, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

13. McKevitt, F. X., "Design and Development Approach for the Augmented Catalytic Thruster (ACT)," AIAA Paper 83-1255, June 1983.

14. Dressler, G. A., et al., "Flight Qualification of the Augmented Electrothermal Hydrazine Thruster," AIAA Paper 81-1410, July 1981.

15. Rawlin, V. K. and Majcher, G. A., "Mass Comparisons of Electric Propulsion Systems for NSSK of Geosynchronous Spacecraft," AIAA Paper 91-2347, June 1991.

16. Gruber, R. P., "Power Electronics for a 1kW Arcjet Thruster," AIAA Paper 86-1507, June 1986.

17. Yoshikawa, T., et al., "Development of a Low Power Arcjet Thruster - Thrust Performance and Life Evaluation," IEPC Paper 91-043, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

18. Deininger, W. D., et al., "Review of Plasma Propulsion Activities in Italy," IEPC Paper 91-005, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

19. Cruciani, G. and Deininger, W. D., "Development Testing of a 1 kW Class Arcjet Thruster," AIAA Paper 92-3114, July 1992.

20. Zube, D. M., et al., "Development of a Low Power, Radiatively Cooled Thermal Arcjet Thruster," IEPC Paper 91-042, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

21. Kurtz, H. L., et al., "Low Power Hydrazine Arcjet Thruster Study," AIAA Paper 92-3116, July 1992.

22. Bassner, H., Berg, H.-P.. and Kukies, R.. "RITA Development and Fabrication for the ARTEMIS Satellite." IEPC Paper 91-057. presented at the AIDAA/AIAA/DGLR/ JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

23. Fearn. D. G., "The UK-10 Ion Propulsion System - A Technology for Improving the Cost-Effectiveness of Communications Spacecraft." IEPC Paper 91-009, presented at the AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio. Italy, October 14-17. 1991.

24. Bassner, H., Berg, H.-P., Kukies, R., "Recent Results on Qualification of the RITA Components for the ARTEMIS Satellite," AIAA Paper 92-3207, July 1992.

25. Beattie, J. R., Matossian, J. N., and Robson, R. R., "Status of Xenon Ion Propulsion Technology," AIAA Paper 87-1003, May 1987.

26. Patterson, M. J., "Low-I_{sp} Derated Ion Thruster Operation," AIAA Paper 92-3203, July 1992.

27. Patterson, M. J. and Verhey, T. R., "5-kW Xenon Ion Thruster Life-test," AIAA Paper 90-2543, July 1990.

28. Kitamura, S., Miyazake, K., and Hayakawa, Y.. "Cyclic Test of a 14 cm Diameter Ring-Cusp Xenon Ion Thruster," AIAA Paper 92-3146, July 1992.

29. Patterson, M. J. and Rawlin, V. K., "Derated Ion Thruster Design Issues," IEPC Paper 91-0150, presented at the AIDAA/AIAA/DGLR/ JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991.

30. Byers, D. C., "Advanced Onboard Propulsion Benefits and Status," NASA TM-103174, March 1989.

31. Haag, T. W. and Curran, F. M., "Arcjet Starting Reliability: A Multistart Test on Hydrogen/Nitrogen Mixtures." AIAA Paper 87-1061, May 1987.

32. Curran, F. M. and Haag, T. M., "An Extended Life and Performance Test of a Low Power Arcjet," AIAA Paper 88-0310, July 1988.

33. Smith, R. D., et al., "Development and Demonstration of a 1.8 kW Hydrazine Arcjet Thruster," AIAA Paper 90-2547, July 1990.

34. Zana, L. M., "Langmuir Probe Surveys of an Arcjet Exhaust," AIAA Paper 87-1950, July 1987.

35. Carney, L. M. and Sankovic, J. M., "The Effects of Arcjet Thruster Operating Condition and Constrictor Geometry on the Plasma Plume." AIAA Paper 89-2723. July 1989.

36. Sankovic, J. M., "Investigation of the Arcjet Plume Near Field Using Electrostatic Probes," NASA TM-103638, October 1990.

37. Carney, L. M., "Evaluation of the Communications Impact on a Low Power Arcjet Thruster," AIAA Paper 88-3105, July 1988.

38. Ling, H., et al., "Near Field Interaction of Microwave Signals with a Bounded Plasma Plume." Final Report, NASA Grant NCC3-127, January 1991.

39. Zafran, S., "Hydrazine Arcjet Propulsion System Integration Testing," IEPC Paper 91-013. presented at the AIDAA/AIAA/DGLR/ JSASS 22nd International Electric Propulsion Conference, Viareggio. Italy. October 14-17, 1991.

40. Bogorad, A., et al., "The Effects of 1 kW-Class Arcjet Thruster Plumes on Spacecraft Charging and Spacecraft Thermal Control Materials," Presented at the 29th Annual International Nuclear & Space Radiation Effects Conference, New Orleans, LA, July 1992.

41. Morren, W. E. and Curran, F. M., "Preliminary Performance and Life Evaluations of a 2-kW Arcjet," AIAA Paper 91-2228, June 1991.

42. Morren, W. E. and Lichon, P. J., "Low-Power Arcjet Test Facility Impacts," AIAA Paper 92-3532. July 1992.

43. Klass, P. J., "Low-Earth Orbit Communications Satellites Compete for Investors and U.S. Approval." Aviation Week and Space Technology, May 18, 1992. pp. 60-61.

44. Curran, F. M. and Sarmiento, C. J., "Low Power Arcjet Performance," AIAA Paper 90-2578, July 1990.

45. Hamley, J. A. and Hill, G. M., "Power Electronics for Low Power Arcjets," AIAA Paper 91-1991, June 1991.

46. Rawlin, V. K., "Characterization of Ion Accelerating Systems on NASA LeRC's Ion Thrusters." AIAA Paper 92-3827, July 1992.

47. Forrester, A. T., Large Ion Beams, John Wiley & Sons, New York, 1988, p. 100.

48. Sarver-Verhey, T. R., "Extended Testing of Xenon Ion

Thruster Hollow Cathodes," AIAA Paper 92-3204, July 1992.

49. Patterson M. J. and Mohajeri, K., "Neutralizer Optimization," NASA TM-105578, October 1991.

50. Divan, D. J. and Kheraluwala, M. H., "High Power Density DC-DC Converters for Aerospace Applications," NAG3-804 Final Report, 1991.

TABLE I. - NEAR TERM APPLICATIONS OF ARCJET AND ION PROPULSION FOR STATIONKEEPING

PROPULSION SYSTEM	APPLICATION	POWER FOR PROPULSION	STATUS	SPONSOR
Radio Frequency Ion Thruster Asssembly (RITA) - xenon	Experiment on European Retrievable Carrier, (EURECA 1), a free-flyer in a ~ 580 km orbit. 2000 h operation. Six month mission life.	0.44 kW	Launched in July 1992.	ESA (Germany)
Hydrazine arcjet	Perform NSSK on AT&T's Telstar 4 communications satellite.	3.6 kW	Flight qualified. Launch planned in 1993.	AT&T (GE and Rocket Research Company - USA)
Xenon ion thruster	Engineering Test Satellite (ETS VI). Prime propulsion for NSSK.	1.6 kW	Launch planned in 1994.	NASDA (MELCO - Japan)
Ammonia arcjet	Arcjet demonstration for stationkeeping. AMSAT - P3 - D program.	~ 0.7 kW	Launch planned in 1995.	Germany
RITA and UK ion thruster systems - xenon	Operational SK subsystem (10 yr) for experimental communications platform ARTEMIS.	0.6 kW	Launch planned ~ 1995.	ESA/ESTEC (Germany)

TABLE II. - TYPICAL THRUSTER PARAMETERS (FLIGHT APPLICATIONS)

	GE ARCJET	EURECA ION	ETS VI ION	ARTEMIS RF ION / UK 10 ION
Propellant	Hydrazine	Xenon	Xenon	Xenon
Power/thruster, kW	1.6	0.44	0.6	< 0.6
Thrust, mN	~ 210	10	23	15/18
Specific impulse, s	502	3300	2910	> 3000
Design life, h	830	> 1700	6500	11,000
Thruster mass, kg	1.0	1.5	3.7	1.6/ *
Power processor mass, kg	5	*	*	9.3/~ 9
Longest life demo, h	1258	*	> 7160	*
Reference	1	6	7	22,23

* Information not available

TABLE III. - TYPICAL ARCJET PARAMETERS, EUROPEAN AND JAPANESE GROUND DEMONSTRATIONS

	SEGAMI - I ARCJET (ISAS)	OSAKA - IHHI ARCJET	BPD/CENTRO-SPAZIO ARCJET	UNIVERSITY OF STUTT-GART ARCJET
Propellant	$N_2 + 2H_2$	$N_2 + 2H_2$	$N_2 + 2H_2$	$N_2 + 2H_2$
Power/thruster, kW	1.8	1.2	1.0	1.2
Thrust, mN	176	~150	130	230
Specific impulse, s	600	400 - 550	440	520
Design life, h	*	*	*	*
Thruster mass, kg	*	*	*	~1.3
Power processor mass, kg	*	*	*	*
Longest life demo, h	*	50	*	*
Reference	4	17	3	20

NAL - National Aerospace Laboratory, Japan; ISAS - Institute of Space and Astronautical Science, Japan

* Information not available

TABLE IV. - TYPICAL ION THRUSTER PARAMETERS (GROUND DEMONSTRATIONS)

	HRL 25 cm	HRL 13 cm	NASA LeRC	NAL 14 cm
Propellant	Xe	Xe	Xe	XE
Power/thruster, kW	1.3	0.44	1.5	~0.6
Thrust, mN	62	18	72	25
Specific impulse, s	2800	2600	2130	~3500
Design life, h	*	~10,000	~10,000	~8000
Thruster mass, kg	9.7	5	~7	*
Power processor mass, kg	~10	6.8	*	*
Longest life demo, h	4350	*	900**	1859
Reference	25	9	8,26	28

HRL - Hughes Research Laboratories

* Information not available ** Tested at 5.5 kW

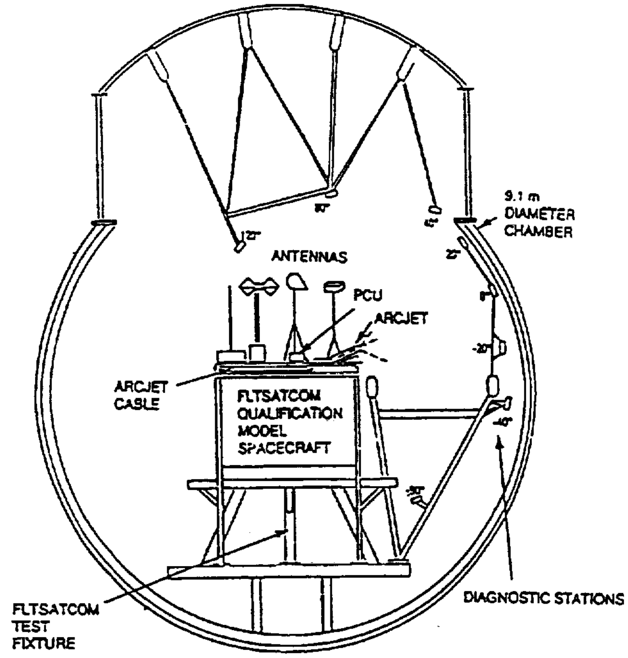

Figure 3. - Test-bed for arcjet impacts on FLTSATCOM qualification model spacecraft (ref. 39).

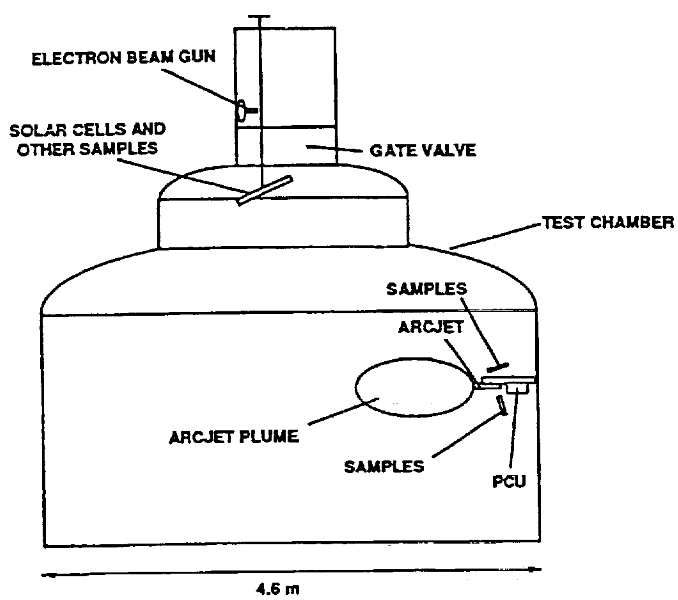

Figure 4. - Diagram of setup for GE/LeRC plume interactions test (ref. 40).

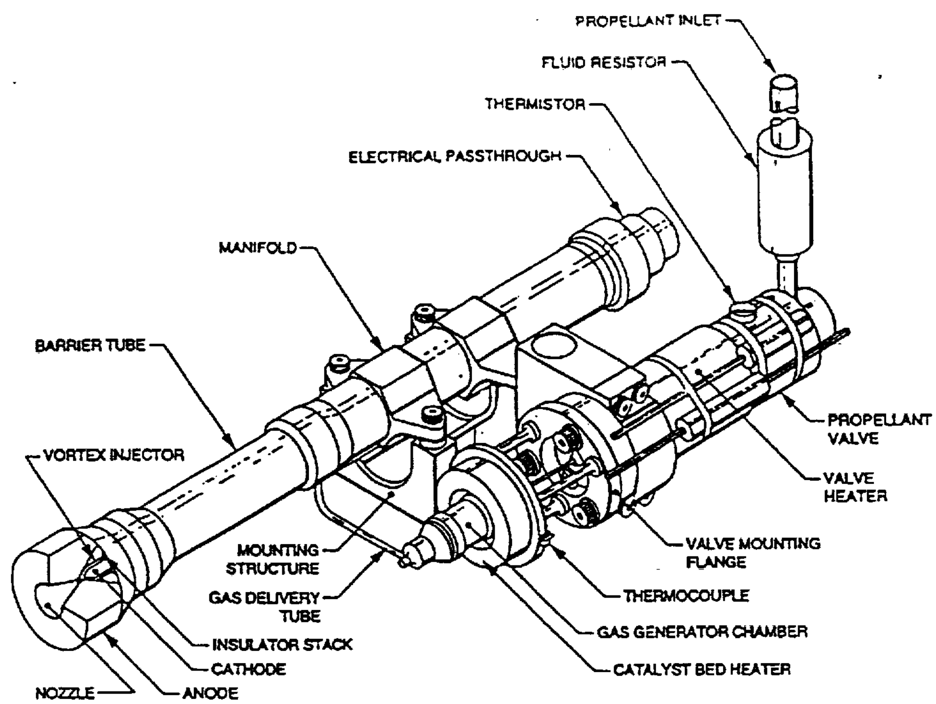

Figure 1. - Arcjet thruster assembly.

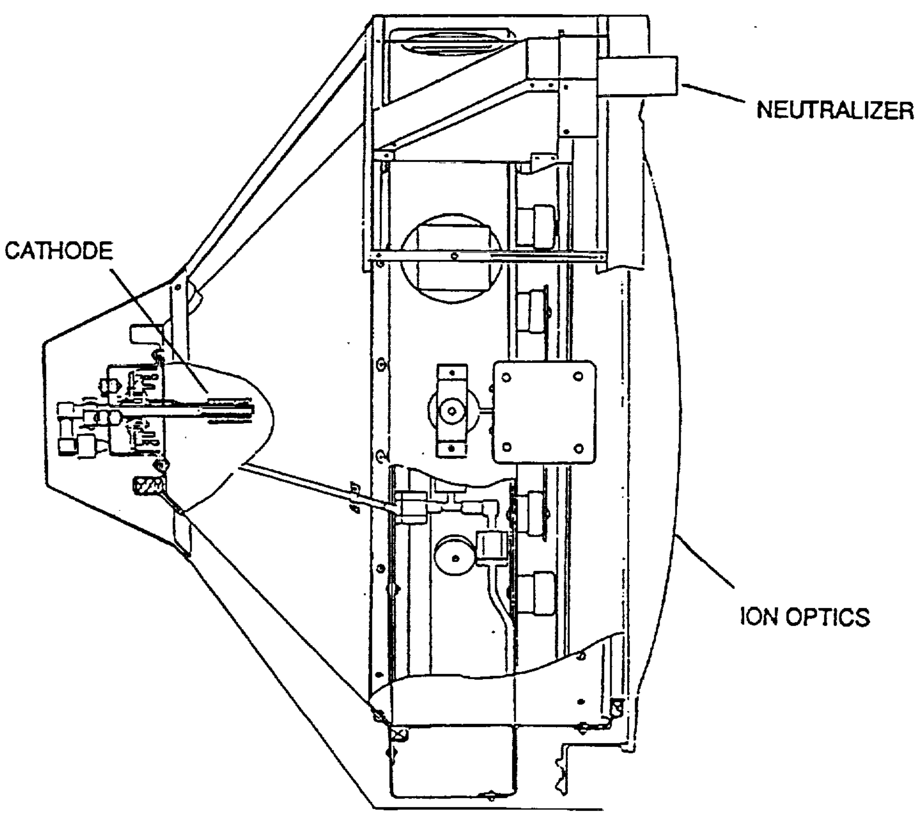

Figure 2. - Lightweight 30-cm ion thruster.

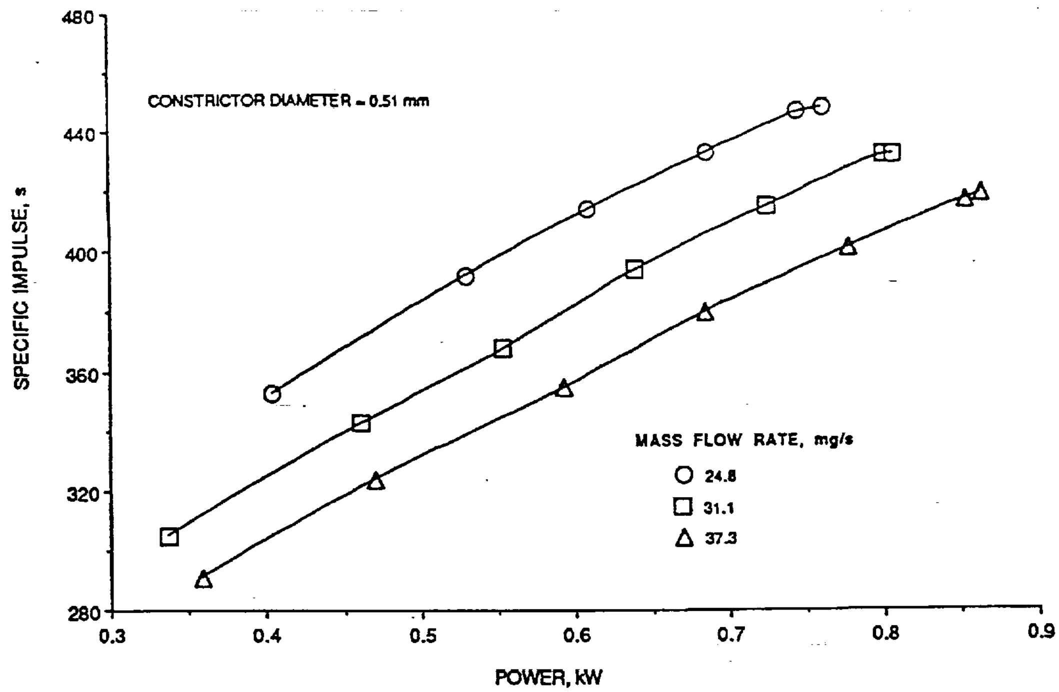

Figure 5. - Specific impulse versus power for low power arcjet (ref. 44).

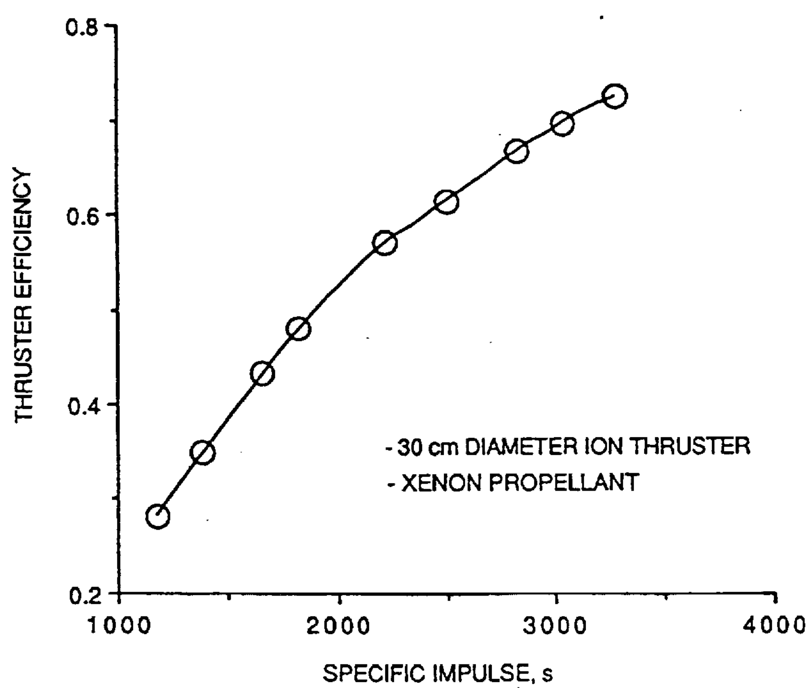

Figure 6. - Thruster efficiency versus specific impulse over a 0.5 to 6 kW input power range (ref. 26).

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James S. Sovey, Francis M. Curran, Thomas W. Haag, Michael J. Patterson, Eric J. Pencil, Vincent K. Rawlin, and John M. Sankovic

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13. ABSTRACT *(Maximum 200 words)*

Near term flight applications of arcjet and ion thruster satellite station-keeping systems as well as development activities in Europe, Japan, and the United States are reviewed. At least two arcjet and three ion propulsion flights are scheduled during the 1992-1995 period. Ground demonstration technology programs are focusing on the development of kW-class hydrazine and ammonia arcjets and xenon ion thrusters. Recent work at NASA Lewis Research Center on electric thruster and system integration technologies relating to satellite stationkeeping and repositioning will also be summarized.

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