AIAA/SAE 7th Propulsion Joint Specialist Conference

THIS Survey Paper, the next three papers, and the first two Engineering Notes in this issue were presented at the AIAA/SAE Propulsion Joint Specialist Conference in Salt Lake City, Utah, June 14–18, 1971. Additional papers from this meeting will be published in a later issue of the *Journal of Spacecraft and Rockets*.

The Propulsion Conference has addressed some of the major technical propulsion problems and presented information for the high performance propulsion systems of the future. In reviewing the papers from the Conference, it is evident that a great deal of sophisticated knowledge exists in the propulsion industry today. Propulsion systems are now highly developed and are capable of delivering design performance over a wide range of operating conditions. In particular, the available knowledge regarding chemical rockets can be utilized for the design of a low cost, highly reliable space shuttle engine.

Future efforts will focus not only on performance improvements and cost reductions, but also on modifications to satisfy current environmental requirements. Several papers presented at the Conference were devoted to pollution generated by jet propulsion. Modeling and prediction techniques for gas turbine emissions were shown to be capable of yielding qualitatively accurate results. Sufficient understanding of combustor nonuniformities appears to be available so that the prediction techniques for exhaust gas composition can be applied to more realistic combustor systems. R and D efforts on reducing the noise of aircraft powerplants also continue to receive strong support, as substantial noise reduction is achieved with prototype configurations.

The Conference proceedings indicate that more effort is required in the development of advanced air-breathing propulsion systems. These advanced systems include not only devices such as higher Mach number turbojets and ducted afterburning rockets but also powerplants for hypersonic flight. Relevant information for this flight regime was discussed and should eventually lead to the design and development of a propulsion system for a hypersonic vehicle. Such a vehicle might well incorporate an advanced cryogenic rocket boost system and a supersonic combustion ramjet.

The 1971 Conference has given us a glimpse of future innovations, e.g., vortex film cooling for rocket chambers, an extendable nozzle skirt, a reliable performance prediction technique for tank injection pressurization, control devices for solid rocket motors, and a staged combustion rocket motor. In addition, the Conference provided us with further information in certain unresolved areas, e.g., combustion instability in rockets, gas turbines, and thrust augmenters. The research programs discussed may eventually provide us with sufficient information to predict motor stability behavior and to evolve design criteria for acoustic absorbers.

The wealth of information included in the Conference papers and exchanged during the discussions should prove useful in the design of advanced propulsion systems. As we progress into the decade of the 70's, we can hopefully look forward to further advances in propulsion in spite of the reductions that have occurred in aerospace funding.

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Survey of Satellite Auxiliary Electric Propulsion Systems

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Introduction

WITH the success of the experimental electric thrusters that are on board the Application Technology Satellite (ATS) IV, Lincoln Experimental Satellite (LES) 6, and Space Electric Rocket Test (SERT) II, along with the applications of state-of-the-art electric propulsion technology to LES 8 and ATS F and G, auxiliary electric propulsion is emerging from experimental to flight status. The satellite auxiliary propulsion designer is now faced with the evaluation of present state-of-the-art auxiliary electric thrusters for application to unmanned satellites, particularly long-life synchro-

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nous satellites. The purpose of this survey is to provide program managers and systems engineers with comparative information on auxiliary electric propulsion systems.

In an earlier report,[1] the thrusters surveyed were limited to those that rely on gasdynamic pressure forces to accelerate propellant, e.g., chemical and inert gas. Electric thrusters surveyed herein rely on electrostatic or electromagnetic forces to accelerate propellant. The varied characteristics of electric thrusters separate these devices into distinct classes. Ion and colloid thrusters comprise the electrostatically accelerated group, while magnetoplasmadynamic (MPD) and pulsed plasma thrusters comprise the electromagnetically accelerated group.

This survey is divided into three sections for each of the four thruster types: 1) general description of the thruster; 2) space flight experience; and 3) survey of state-of-the-art thrusters.

The thrusters discussed were developed by Hughes Research Laboratory (HRL), Malibu, Calif.; Electro-Optical Systems (EOS), Pasadena, Calif.; Fairchild Hiller Corp., Farmingdale, N.Y.; TRW Systems Group, Redondo Beach, Calif.; NASA Lewis Research Center (LeRC), Cleveland, Ohio; General Electric Company, Space Division, Philadelphia, Pa.; and Cornell Aeronautical Labs., Buffalo, N.Y. These discussions are followed by comparisons of surveyed thrusters for use on satellites, to provide east-west and north-south stationkeeping.†

Ion Thrusters

What is an Ion Thruster?

The ion thruster is defined as a device that electrostatically accelerates charged single atoms or ions. Ion thrusters may be categorized according to the process by which the ions are produced. Bombardment ionization results from the collision of an electron with a single propellant atom in the gaseous phase. Contact ionization is an interaction between a single propellant atom and a heated surface of a metal with a high work function. These sources are sufficiently different in their operational requirements and in the nature of the ion beam they produce to influence the design of other elements in the accelerator and the design of the electric power supply required. Ion engines are therefore normally classified in terms of their source: contact-ion thrusters and electron-bombardment thrusters.

A conceptual diagram of a contact-ion thruster is presented in Fig. 1. Liquid cesium propellant is fed to a vaporizer by capillary action. The Cs vapor produced at a pressure of less than $7 \times 10^3 N/m^2$ (1 psia) flows through a heated porous tungsten ionizer (877–1077°C), and contact ionization proceeds with 99–100% effectiveness. (Effectiveness in this sense means percent of propellant flow ionized.) The ionizer is maintained at a voltage V_A above spacecraft potential and the accelerator electrode is maintained at a voltage V_B below spacecraft potential to preclude electron migration (from neutralizer) into the acceleration gap and provide sufficient potential gradient for proper ion focusing. The ratio of beam power to thrust is proportional to exhaust speed (I_{sp}); therefore, thruster power requirements can become excessive at high exhaust speeds. A decelerator electrode is sometimes provided. During thruster operation, positive ions are being

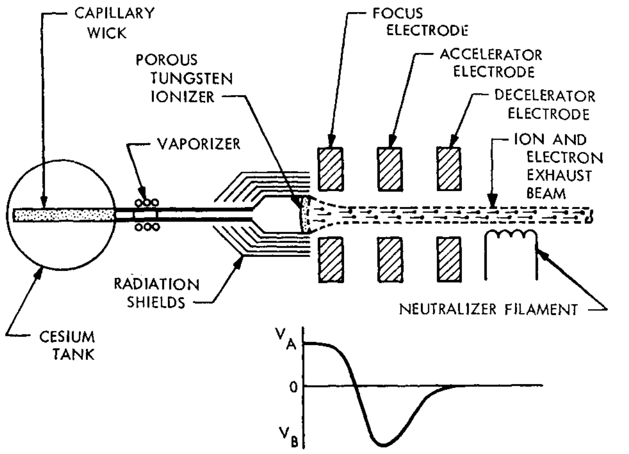

Fig. 1 Conceptual diagram of the contact-ion thruster.

emitted from the device; hence, if no provision for neutralization of the iron beam were made, the spacecraft would acquire a negative potential. The ion exhaust beam is neutralized by electrons (electrons drawn into beam by positive ions) supplied from a thermionic filament of plasma-bridge neutralizer.

Principles of operation of the electron-bombardment thruster are shown in Fig. 2. Liquid cesium or mercury propellant is fed to a vaporizer. The propellant vapor produced is then fed to a discharge chamber. Electrons are emitted from the cathode and spiral about magnetic field lines in the discharge chamber. Electrons suffer collisions with propellant in the discharge chamber and ions are formed. Cathode configurations differ between designs; however, a hollow cathode with a fraction (10–100%) of the propellant flowing through it has been incorporated in most recent electron-bombardment ion discharge chambers. Anode-magnet configurations differ between thruster designs, with the simplest arrangement presented in Fig. 2. Most discharge chambers have a screen to provide a physical surface to which the plasma boundary can attach itself. The accelerator and neutralizer of the electron-bombardment thruster are similar to those discussed in the contact-ion thruster system.

Space Experience of Ion Thrusters

In August 1958, the first ion thruster experiments were conducted. On July 20, 1964, the Space Electric Rocket Test (SERT I) spacecraft was launched. A four-stage Scout launch vehicle followed a ballistic trajectory over the Atlantic Ocean, providing two ion thrusters 47 min of space environment. One of the two thrusters, a cesium contact-ion thruster, was unable to operate because of a high-voltage short circuit. The other, a mercury bombardment thruster, successfully

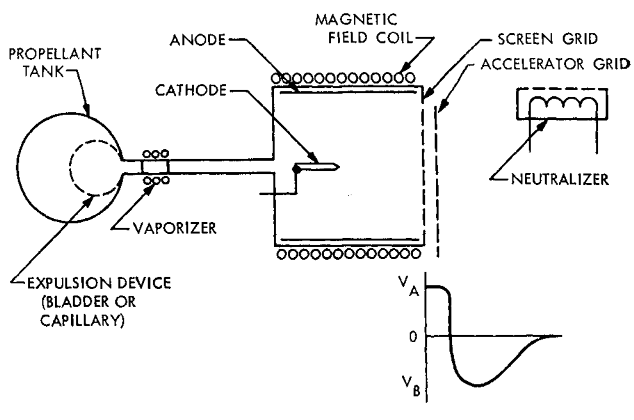

Fig. 2 Conceptial diagram of the electron-bombardment thruster.

† In an effort to abridge this survey, detailed discussion of the following auxiliary electric thruster systems has been deferred: HRL ion micro-thruster, HRL dual-beam contact-ion thruster, EOS 450-μN (100-μlbf) cesium bombardment thruster, General Electric's Solid Propulsion Electric Thruster (SPET), and Cornell Aeronautical's Pulsed Vacuum-Arc Thruster (PVAT); however, these thrusters have been considered in the conclusions section of this survey. A more detailed discussion and additional information on the thrusters will be included in an Addendum to Ref. 1.

operated for over 31 min. Three flights of the Blue Scout vehicle were used to place cesium contact thrusters (EOS) into ballistic trajectory. The second flight was successful, and data on the cesium contact-ionization thruster were returned. An EOS cesium contact ion thruster was launched onboard the Snapshot satellite; operation of its engine was unsuccessful. More recently, a cesium contact thruster was flown on ATS-D and operated successfully. The SERT-II was orbited on February 3, 1970. Data on the two mercury bombardment thrusters on this satellite have been returned. One of the two thrusters failed after more than 3800 hr of operation; at this point the second thruster was made operational. The second thruster failed after 2000 hr. Failure of both thrusters has been identified as a result of impingement of neutralizer generated particles on the accelerator electrode.

EOS ATS-D and -E Contact-Ion Microthruster

Electro-Optical Systems has designed several contact-ion microthruster systems. The most recent thruster design is the 0- to 89-μN (0- to 20-μlbf) cesium contact-ion microthruster system developed for the ATS-D and -E satellites.[2,3] The thruster employs a single-button porous-tungsten ionizer. The thruster system is complete with propellant feed system, power conditioning, and 12 telemetry channels. Thrust vectoring ($\pm 10°$) is accomplished by electrostatic beam deflection. A tantalum (50 ppm yttrium) hot-wire neutralizer is employed. Two neutralizers are provided; one is a backup in case of failure of the active neutralizer. The complete thruster system is depicted in Fig. 3.

The cesium propellant is stored in a cylindrical reservoir enclosing a fin array. There are 120 fins with spacing 0.16 cm (0.064 in.) at the outer edge of the array. Cesium is fed to the center of the array where a porous nickel rod carries Cs through a sheathed vaporizer heater (0 to 4 w required for vaporization of Cs). Propellant is sealed in the propellant tank assembly with an Invar tube with a nickel tip, which seats into an aperture in the vapor feed line. During vaporizer and ionizer operation, the feed line is automatically opened because of the difference in thermal expansion between the Invar tube and its stainless steel housing. The power conditioning efficiency is 60%. Three high voltages are required: beam voltage (+3000 v), deflection voltage, and accelerator voltage (−2000 v), along with low-voltage power to the ionizer, vaporizer, and neutralizer. The complete system, including power conditioning, propellant feed system, and thruster, has a mass less than 2.7 kg (6 lbm).

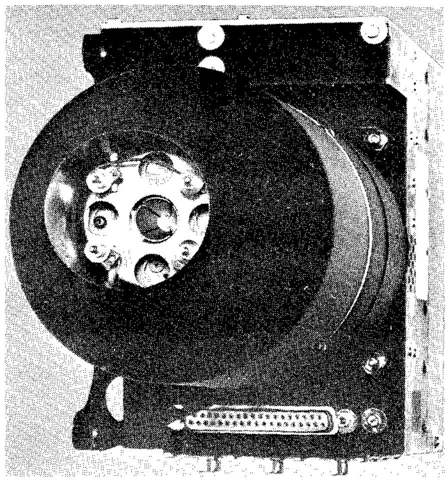

Fig. 3 EOS ATS-D and -E contact-ion microthruster (photo courtesy of EOS)

The system is capable of thrust level selection with nominal preset points of 0, 22, 44, 67, and 89 μN (0, 5, 10, 15, and 20 μlbf) thrust. Operation at 89 μN (20 μlbf) thrust has led to problems; the system operates best below this value. Typical operating data are presented in Table 1.[2] These data were taken from flight acceptance testing where the thruster operated for 12 hr in vacuum at 5 to 30°C with three on-off cycles. The ionizer requires a constant 9.7 w to maintain the porous tungsten button at approximately 1040°C. The tantalum neutralizer requires a fixed 3.5 w. The electrostatic beam deflection system uses an insignificant amount of power.

Life testing of two ATS-D and -E contact-ion thrusters can be used as an indicator of thruster expected life.[4] Program goal was 3000 hr of steady-state operation at 67 μN (15 μlbf) thrust. One unit accumulated 2245 hr of operation at 67μN (15μlbf) during a total test time of 3481 hr; however, high accelerator current drain conditions led to the decision to terminate the test. An investigation of the drains was undertaken. Sputterback from the test chamber walls was the suspected cause of accelerator current drain. The conclusion

Table 1 Ion thruster performance data

Thruster type	Thrust, μN (μlbf)	Thruster power, w	Specific impulse, N-sec/kg (lbf-sec/lbm)	Power/thrust w/mN (w/mlbf)	Mass utilization efficiency, %	Engine efficiency, %	Power conditioning efficiency, %	Total power, w
EOS ATS-D and -E cesium contract-ion micro-thruster	22 (5)	15.0	76,982 (7850)	1290 (5510)	~100	5.7	54.3	27.6
	44 (10)	16.6	103,950 (10,600)	671 (2980)	~100	13.9	55.7	29.8
	67 (15)	18.1	70,118 (7150)	455 (2020)	~100	12.9	58.9	30.7
	89 (20)	26.0	65,705 (6700)	369 (1640)	~100	14.6	61.7	32.4
EOS 4.5-mN (1-mlbf) cesium bombardment thruster	4530 (1020)	117.0	23.732 (2420)	26 (115)	90.0	46.0	83.0	141.0
HRL 5-cm mercury bombardment thruster	1643 (370)	54.5	18.829 (1920)	32 (140)	75.0	26.2	85.0	64.0

yields no prediction of how long the thruster is expected to operate in a space environment where sputterback is not a problem. Test data returned from the ATS-D satellite indicate low accelerator current drain during the satellite's brief (less than 10 hr) testing periods.[3] From these data, a thruster lifetime (space environment) of 3000 to 5000 hr can be expected without large accelerator electric current drains.

Extended life tests of tantalum and tantalum (50 ppm yttrium) neutralizer filaments were conducted. Four tantalum-yttrium filaments operated in excess of 10,000 hr, while two pure tantalum filaments failed at less than 10,000 hr. All six were tested at a pressure of 8×10^{-6} N/m^2 or lower, to avoid hydrocarbon vapors that could embrittle the filaments. The neutralizer performed well during the space tests of the ATS-D experiment. Emission-limited neutralizer operation was observed at one point during the test. It was later postulated that one of the gravity-gradient booms passing through the ion thruster exhaust beam caused this anomaly.

EOS 4.45-mN (1-mlbf) Cesium Bombardment Thruster

Electro-Optical Systems has recently developed a 4.45-mN (1-mlbf) thrust cesium-bombardment ion thruster for north-south stationkeeping on the synchronous ATS-F satellite.[5] The system includes an ion engine, zero-*g* feed system, neutralizer, and power conditioning. Thrust vectoring ($\pm$ 7°) is accomplished by accelerator grid translation. The ion engine employs the highly efficient magnetoelectrostatic containment (MESC) discharge chamber and a plasma-bridge neutralizer. The 4.45-mN (1-mlbf) cesium electron-bombardment thruster system is depicted in Fig. 4.

The thruster employs magnetoelectrostatic plasma containment (Fig. 5).[6] Bounding surfaces of the discharge chamber act as a magnetic wall to reflect most of the ions and electrons approaching them. This greatly reduces ion losses to the walls, increases mass utilization efficiency, and provides a uniform plasma for efficient ion extraction. Ten percent of the propellant flows through a hollow cathode (-500°C) which acts like an autocathode after discharge power (-3 w) is initiated. The remainder (90%) of the cesium propellant is introduced to the discharge chamber through a feed ring based at the potential of the boundary anodes. Accelerator displacement is used as a means of thrust vectoring. Thermal expansion forces are used to displace the accelerator perpendicular to the engine axis creating $\pm$ 9° of thrust vectoring. Cesium propellant is stored in a cylindrical reservoir enclosing a fin array. Three feed systems are required for the MESC thruster, one for the plasma-bridge neutralizer, one for the thruster cathode, and one for the thruster anode. Each feed system is sealed with a thermally actuated bimetal valve similar to that used on the ATS-D contact-ion microthruster. Capacity of the feed systems can be increased or decreased with the ATS-F feed system storing 3.4 kg (7.4 lbm) of cesium (0.64 kg or 1.4 lbm for cathode) for the discharge chamber and 0.09 kg (0.2 lbm) for the plasma-bridge neutralizer. Feed system container mass is 2.0 kg (4.5 lbm). The plasma-bridge neutralizer requires approximately 8 w power and consumes 24 mg of cesium per hr of operation. The power conditioning efficiency is 83% for this system, which provides high voltage to the discharge chamber and accelerator grid along with power to grid displacement heaters, vaporizers, cathode and neutralizer. Power conditioning system mass was 8.6 kg (19 lbm), with a total system mass of 15 kg (33lbm). These mass data include 3.5 kg (7.6 lbm) of cesium.

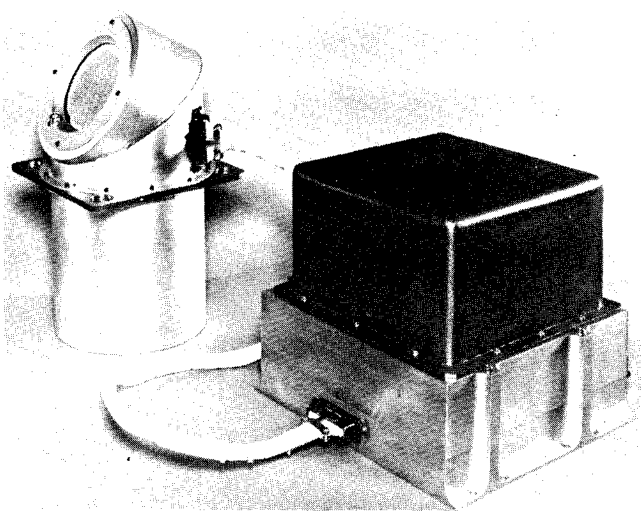

Fig. 4 EOS 4. 45-mN (1-mlbf) cesium bombardment thruster (photo courtesy of EOS).

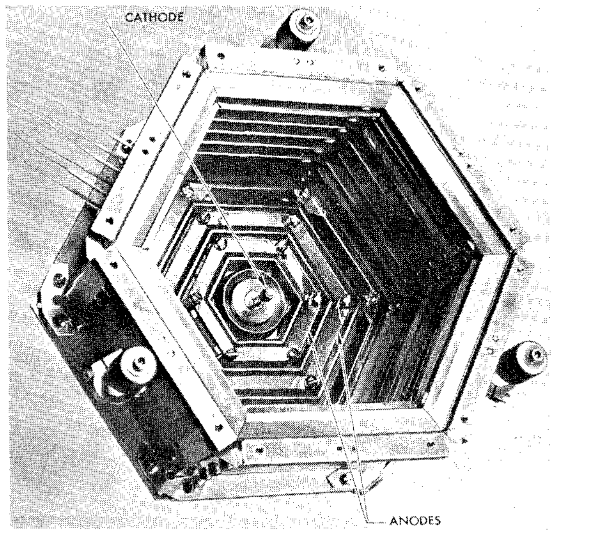

Fig. 5 Discharge chamber geometry of the EOS magnetoelectrostatic containment (MESC) thruster (photo courtesy of EOS).

The MESC electron-bombardment thruster develops a nominal 4.57-mN (1.03-mlbf) thrust at 24,500-N-sec/kg (2500-lbf-sec/lbm) specific impulse with a discharge chamber mass efficiency of 93%. These data must be adjusted to account for the unaccelerated cesium propellant expelled from the plasma-bridge neutralizer. This results in a system specific impulse of 23,700 N-sec/kg (2420 lbf-sec/lbm) and a 90% mass efficiency (percent unaccelerated propellant). Detailed thruster system performance data taken from a 465-hr integration test are presented in Table 1.[5] The neutralizer required 5 to 11 w.

The system has undergone 465 hr of test with successful system operation. No lifetime data of the MESC thruster are presently available. Life tests of a 28.9-mN (6.5-mlbf) cesium electron bombardment thruster for more than 8000 hr indicate long life of cesium-bombardment thruster grids.[7] A lifetime of 20,000 hr was predicted from measurement of accelerator grid erosion. The plasma-bridge neutralizer failed after 8100 hr of test, because the neutralizer propellant orifice became clogged. Backsputtered material from the test chamber walls was blamed for neutralizer clogging, and operation in a space environment should alleviate this mode of neutralizer failure.

5-cm Mercury Bombardment Thruster

Low-thrust mercury-bombardment thruster research has been underway at NASA Lewis Research Center (LeRC) for several years. The most recent low-thrust mercury electron-bombardment thruster[8, 9] employs a 5-cm-diam discharge chamber and a glass-coated grid. A series of preliminary thruster tests was conducted at LeRC with a variable-geometry discharge chamber. Hughes Research Labs. are presently developing this experimental thruster into a flight-prototype system capable of providing north-south station keeping of a synchronous satellite. The system includes ion engine, zero-g feed system, and neutralizer. Thrust vectoring ($\pm 10°$) is accomplished by electrostatic beam deflection. The ion engine utilizes a plasma-bridge neutralizer. The LeRC 5-cm mercury electron-bombardment thruster system is depicted in Fig. 6.

The thruster employs a 5-cm-diam permanent magnet discharge chamber (Fig. 7). A hollow enclosed cathode and a conical cathode pole piece are used to improve discharge chamber performance at the low thrust. Glass-coated grids are incorporated to reduce discharge power losses at low specific impulses. All the propellant flow goes through the hollow enclosed cathode. The method of thrust vectoring is not firmly established; however, an electrostatic thrust vectoring scheme is being extensively investigated at HRL. Glass grids capable of electrostatic beam vectoring are not presently developed; however, experimental two-grid systems have demonstrated the technique.[10] The 5-cm mercury bombardment thruster could rely on a standard glass grid and engine gimbaling, or a two-grid system and grid translation to provide thrust vector control. The feed system is similar to the SERT-II mercury feed system. The propellant is stored in a reservoir and expelled by a pressurant gas. An elastomeric bladder is used for propellant-pressurant separation. The feed system of the HRL design has a mass of 1 kg (2.2 lbm) and a capacity for up to 6.2 kg (13.6 lbm) of mercury. An isolator is provided so that the main cathode and the neutralizer can share the same propellant tank. The plasma-bridge neutralizer is designed to provide sufficient electrons to neutralize a 30-ma beam with 6.3% of the main propellant flow required for the neutralizer. A power conditioning system mass of 2.7 kg (6.0 lbm) is estimated,[11] with an efficiency of 85% expected. The total system has an estimated mass of 2.3 kg (5.1 lbm) with a maximum propellant capacity of 6.2 kg (13.6 lbm) of mercury.

The HRL 5-cm mercury bombardment thruster develops 1640-μN (370-μlbf) thrust at a nominal 18,800-N-sec/kg (1920-lbf-sec/lbm) specific impulse with a discharge chamber mass efficiency of 75%. Discharge chamber mass efficiency includes neutralizer propellant consumption. Detailed thruster performance data taken from Ref. 8 are presented in Table 1. The plasma-bridge neutralizer requires a maximum of 12.0 w.

Life test data of this thruster system are not available, since it is still in the developmental stage. Grid lifetime seems to limit thruster life at present. During a 1000-hr life test of a 15-cm glass-coated grid,[12] weight loss measurements indicated that the grid would be 100% eroded in 22,000 hr. This life

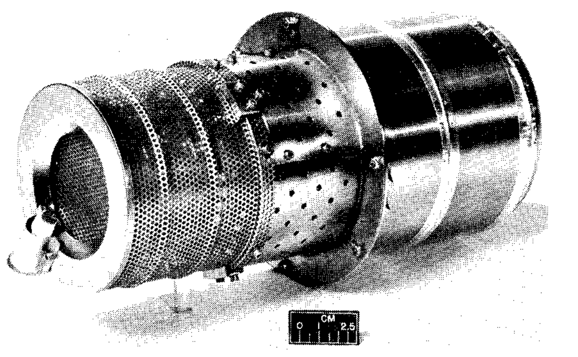

Fig. 6 Five-cm mercury bombardment thruster (photo courtesy of LeRC).

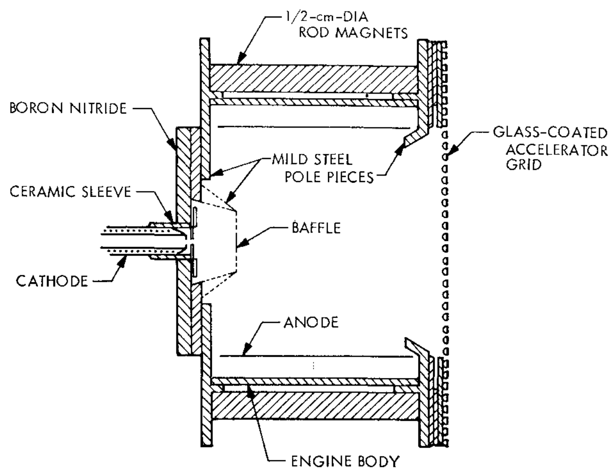

Fig. 7 Five-cm mercury bombardment thruster discharge chamber (representative of HRL and LeRC designs).

test was divided into two parts: 500 hr of normal operation followed by 500 hr of grid-edge termination studies. During the second 500 hr of testing, this grid was modified with masks over the edges of the grid. Although lifetimes of 10,000 hr have been projected for glass-coated grids, a great number of glass grids have failed after a few hours of test. Long-life glass grids seem to depend heavily on extremely tight quality control, since irregularities in glass coating induce premature failure.[12] It has also been speculated that interactions with the test facility may be responsible for such failures, making ground test of these devices very difficult.[13]

Two-grid bombardment thrusters (these operate at a somewhat higher voltage or specific impulse than glass-coated grid thrusters) have undergone extensive life testing. The 15-cm SERT-II mercury bombardment thruster has been life-tested at several facilities. Thruster tests have exceeded 5000 hr; however, excessive erosion near the neutralizer was noted. Proper design and placement of the neutralizer should alleviate this problem in future thrusters. In earlier studies of mercury bombardment thruster lifetime, grid erosion rates were estimated.[14] Lifetimes of 10,000 hr for a 20-cm mercury bombardment thruster were estimated on the basis of observed grid erosion rates.

Sputter yield (i.e., number of metal atoms eroded per incident ion) is an increasing function of ion energy or the potential difference between the accelerator grid and neutralizer. Therefore, increased grid life can be expected for the 5-cm mercury bombardment thruster since the negative accelerator voltage (-250v) is nearer spacecraft potential than the SERT-II negative accelerator voltage (-1750v). Accelerator grid erosion is also a function of current density; however, the SERT-II and the 5-cm thruster are of comparable current density. The two-grid 5-cm mercury bombardment thruster with proper neutralizer geometry should provide 10,000 hr of life.

Plasma-bridge neutralizers have been tested in conjunction with thruster life tests. One mercury bombardment thruster test was in excess of 5000 hr. An endurance test of a SERT-II[15] design neutralizer has been conducted. The plasma-bridge neutralizer operated for 12,000 hr, emitting 250 ma to a collector plate.

Colloid Thrusters

What is a Colloid Thruster?

The colloid thruster is defined as a device that electrostatically accelerates multiatom or multimolecular charged particles. Solid and liquid heavy particles have been studied;

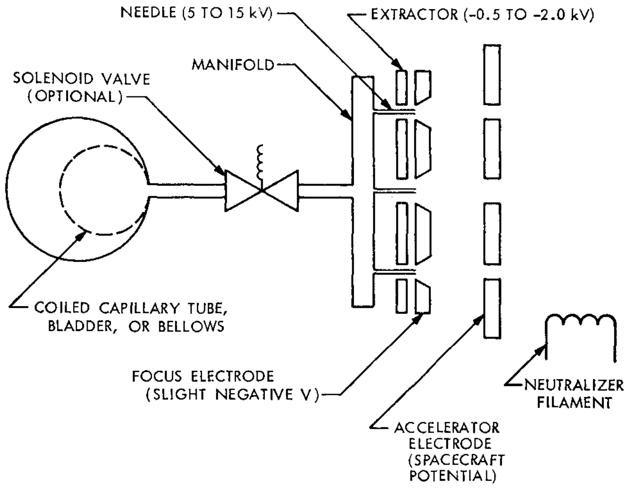

Fig. 8 Conceptual diagram of a colloid thruster.

however, liquid colloids have proved most successful. A conceptual diagram of a colloid thruster is presented in Fig. 8. Liquid propellant (glycerol to which sodium iodide is added) is stored in a reservoir, which can be a rigid tank, a coiled feed tube, or a bellows tank. If the propellant is pneumatically fed as in the system presented in Fig. 8, then a propellant isolation valve is required. If a bellows is used, then the propellant can be fed mechanically, and therefore an isolation valve is required only to protect the propellant from an atmosphere prior to launch. The propellant is distributed in a manifold to needles, linear slits, or annular slits (Fig. 9). A potential is maintained between the needle and the extractor electrode. The intense electric field in the vicinity of the thin needle rim exerts a force upon the fluid surface, producing electro-hydrodynamic spraying of charged propellant droplets. These multimolecular droplets are charged positive

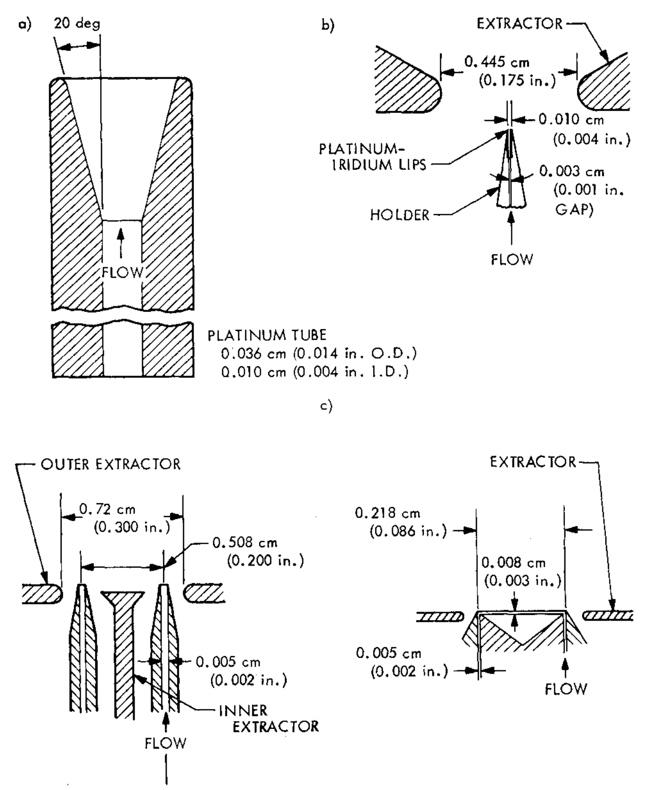

Fig. 9 Geometry of a) standard needles, b) linear slits, and c) annular slits.

when the capillary is maintained at a positive potential and negative when the voltage potential is reversed. A focus electrode is required to contain the charged droplets within the accelerator gap. The accelerator electrode is maintained at spacecraft potential. In recent thrusters, the focus and accelerator electrodes have been removed. Again as in the ion thrusters, a neutralizer is required; however, the lower charge/mass ratio for colloid systems as compared to ion thruster systems dictates substantially lower neutralizer requirements for a colloid thruster of comparable thrust.

Colloid Microthruster

To the time of this study there have been no orbital tests of colloid thrusters; however, TRW Systems Group has developed a colloid micro-thruster system[16,17] consisting of a three-needle array thruster, neutralizer, zero-g propellant feed system, and power conditioning unit. This colloid system develops a nominal 36-μN (8-μlb) thrust at 9320-N-sec/kg (950-lb-sec/lbm) specific impulse and was originally developed for providing yaw control on a multipurpose satellite in synchronous orbit. When plans for the satellite were cancelled, the colloid thruster was redirected to laboratory testing. The complete prototype system is shown in Fig. 10.

The propellant, glycerol doped with a sodium iodide, is stored in a coiled Inconel tube. The pressurant gas (argon) is contained in an annulus surrounding the propellant storage coils. The annulus gas volume is twice the initial propellant volume, so that feed pressure is reduced by 33%: 4800 to 3200 N/m^2 (36 to 24 torr) near complete expulsion. Propellant storage and expulsion tank mass is 0.8 kg (1.7 lbm) with 0.2 kg (0.5 lbm) of propellant. Since the propellant is constantly under pressure, an in-line propellant valve is required. Because of the very low propellant flow rates (e.g., 4 μg/sec), a valve design that minimizes "pumping" of propellant through the capillary array during actuation is desired. A tungsten carbide ball seat is employed. Valve actuation is implemented with a thermal actuator (2 w). Dual valve assembly mass is 0.14 kg (0.3 lbm). The power conditioning unit must supply power to the propellant valve, neutralizer, and system heaters, along with high-voltage power to the propellant needles (4 to 8 kv) and the extractor electrode (400 to 800 v). With a nominal input of 5 w (spacecraft voltage) the unit can supply 1.4 w of conditioned power at a conversion efficiency of 28%. The mass of the power conditioning unit is 1.3 kg (2.9 lbm). A small thermal control heater is provided to

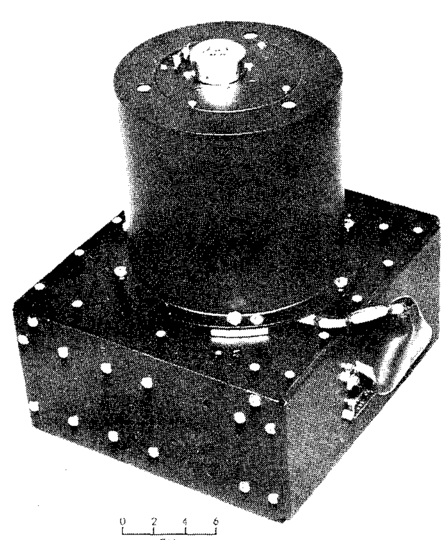

Fig. 10 Colloid microthruster system (photo courtesy of TRW System Group).

maintain the thruster at a preset 25 or 30°C. Thruster mass is 0.05 kg (0.1 lbm) and structure mass is 0.7 kg (1.5 lbm). This result in a total system mass of 3.2 kg (7.0 lbm) including propellant.

Typical operating data are presented in Table 2. These data are for a 36-μa beam current and a thruster temperature of 29°C. Both the beam current and temperature remain fixed. Since the feed system operates from a blow-down propellant tank, the feed pressure decreases with propellant consumption. As feed pressure decreases, the mass flow rate of propellant decreases. The mechanism of electrohydrodynamic spraying is not well understood; however, at low mass flow rates particles are extracted at a somewhat larger average charge-to-mass ratio, which results in a decrease in thrust and an increase in specific impulse.

A life test of this thruster has been performed. Results from over 1750 hr of microthruster operation are presented in Refs. 17 and 18. Thruster operation was good through 1300 hr of log time with a slight decrease in propellant mass flow rate. (Time thruster was in vacuum). For a period of 70 hr, a 0.3- to 0.5-μa extractor current persisted. The current leakage was finally corrected by operating the thruster with a closed valve and a high operating temperature. During the log hours period 1484 to 2348, a constant decrease in mass flow rate was noticed and thruster temperature was increased in attempts to improve the mass flow rates. Decreased mass flow rate was later explained by a leak in the pressurant gas tank. Another life test of the colloid microthruster is reported in Ref. 19. A three-needle array was tested for 1300 hr when a power failure permitted the propellant to form a droplet between the needle and the extractor. A second test of the array lasted 2400 hr with a similar power failure. At test termination, needle tips were in excellent condition; however, prevention of glycerol flow to the extractor and tar deposits due to electron bombardment of glycerol are necessary for long-life colloid thrusters.

In Refs. 17 and 18, the tungsten neutralizer was tested in conjunction with the thruster, and filament voltage was monitored during the test. A rapid increase in neutralizer voltage was noticed at 25 hr into the test. This was attributed to carbiding of the tungsten filament due to the presence of hydrocarbons (colloid beam products) in the test chamber environment. A controlled oxygen leak was introduced to the chamber to counter the tungsten carbiding. Neutralizer operation was terminated after 950 hr because of electron leakage to high voltage. The leakage was subsequently attributed to holes through the support insulators between the feed system and the thruster housing. Electron leakage was not evidenced after covering the support insulator holes during post-test analysis.

TRW 4. 4-mN (1-mlbf) Colloid ADP Thruster

A 4. 4-mN (1-mlbf) thrust colloid system is presently under development at TRW Systems Group under sponsorship of the Air Force Aero Propulsion Lab.[20] The thruster will

Table 2 Colloid thruster performance data

Thruster type	Geometry	Thrust, µN (µlbf)	Thruster power, W	Specific impulse, N-s/kg (lbf-s/lbm)	Mass flow rate, µg/s	Reference	Power/thrust, W/mN (W/mlbf)	Engine efficiency, %	Power conditioning efficiency, %	Total power, W
TRW colloid micro-thruster[a]		39 (8. 7)[b]	1. 39[c]	9611 (960)	4. 0	17	36 (160)	13. 1	28. 0	5. 0
		23 (5. 2)	1. 43[c]	16, 867 (1720)	1. 5	17	70 (275)	13. 6	29. 0	5. 0
		48 (10. 8)	1. 38[c]	7257 (740)	6. 5	17	29 (128)	12. 6	28. 0	5. 0
Typical colloid thrusters	36 needles	371. 0 (83. 5)	3. 91	14, 710 (1500)	24. 9	Estimate	10. 54 (46. 8)	70. 0		
	36 needles	383. 0 (86. 0)	3. 80	14, 691 (1498)	25. 6	21	9. 95 (44. 2)	70. 0		
	36 needles	306. 0 (68. 9)	3. 30	14, 318 (1460)	21. 4	21	10. 79 (47. 9)	66. 0		
	36 needles	345. 0 (77. 6)	3. 80	14, 837 (1513)	23. 3	21	11. 01 (48. 9)	67. 0		
	6 annular slits	534. 0 (120. 0)	5. 54	15, 200 (1550)	35. 0	21	10. 41 (46. 2)	72. 0		
	6 annular slits	512. 0 (115. 0)	5. 52	14, 906 (1520)	36. 0	21	10. 81 (48. 0)	70. 0		
	6 annular slits	489. 0 (110. 0)	5. 22	14, 514 (1480)	35. 0	21	10. 70 (47. 5)	69. 0		
	1 annular[d] slit (EOS)	89. 0 (20. 0)	0. 87	13, 141 (1340)	6. 8	22	9. 80 (43. 5)	73. 0		
TRW 4. 5-mN (1-mlbf) ADP colloid	(footnote)[e]	4, 450 (1000)	58. 00[f]	14, 710 (1500)	298. 8	Estimate	13. 0[f] (58)	56. 0[f]	85. 0	68. 2

[a]Data at 29°C, 36 mA.
[b]Design thrust level.
[c]Includes 0. 82 W for neutralizer and 0. 30 W for thruster heater.
[d]Data at 0°C.
[e]12 modules of 36 needles each with the performance of the fourth line of this table.
[f]Includes 5 W for neutralizer, 6 W for thruster heater and mass flow rate controller.

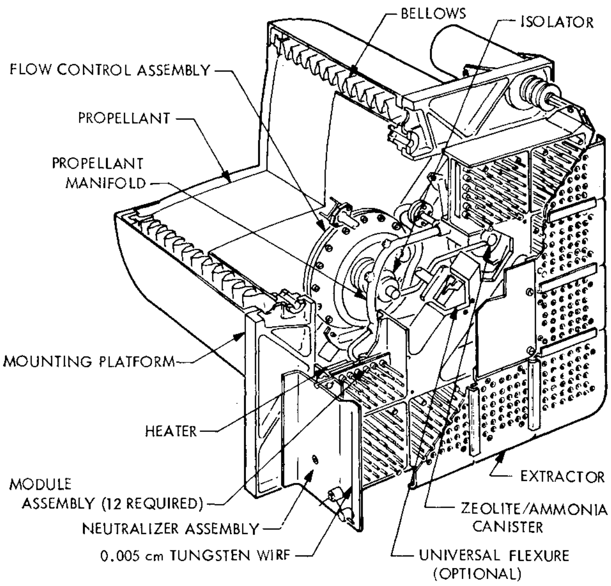

Fig. 11 TRW 4.45-mN (1-mlbf) colloid ADP thruster concept.

comprise twelve 36-needle modules. Each module will deliver approximately 355 μN (80 μlbf) thrust at a nominal 14,710 N-sec/kg (1500 lbf-sec/lbm) specific impulse. This system is designed for north-south station keeping of a synchronous satellite. A program goal is a 10,000-hr thruster lifetime. An artist's conception of this thruster is depicted in Fig. 11.

The sodium iodide doped glycerol is stored in a bellows tank, with propellant flow rate mechanically controlled. Feed system mass is estimated at 2.9 kg (6.4 lbm) with a maximum propellant load of 11.3 kg (25 lbm). This allows for 10,000 hr of thrusting at 14,710 N-sec/kg (1500 lbf-sec/lbm) specific impulse and a 95% mass expulsion efficiency. The mechanical flow controller operates on a feedback loop from the beam current. Average power requirement for the controller is 1 w with a maximum predicted peak power required of 2 w. The power conditioner must provide high voltages for the needles and extractor (up to 12.3 kv) along with power for the thruster heater, neutralizer, and propellant flow rate controller. The power conditioning subsystem is estimated at 3.1 kg (6.9 lbm) with a 85% efficiency. It will provide 70 w of conditioned power. Thruster mass is 0.95 kg (2.1 lbm) with a 2.0 kg (4.5 lbm) structural mass estimated. Total fixed system mass is 9.1 kg (20 lbm) with up to 11 kg (25 lbm) of propellant capacity.

Typical operating data for a 36-needle module is presented in Table 2. The fourth line corresponds to the design data used in estimated 4.4-mN (1-mlbf) thruster system performance. The following lines are test data obtained on several types of modules. The annular slit performance was taken TRW Systems Group[21] and EOS[22] data. The final line in Table 2 is estimated performance of the 4.4-mN (1-mlbf) colloid thruster system. A heater power of 3.0 w is estimated for synchronous orbit of north-south station keeping thrusters. Neutralizer power is estimated at 5.0 w, providing 3.8 ma of negative current for 10,000 hr.

Life test data are not available for a 4.4-mN (1-mlbf) thruster array. The life test of a 3-needle microthruster system has previously been discussed. Tests of 36-needle modules have exceeded 1000 hr with little or no degradation in thruster performance.[21] Current leakage between the needle and extractor were noticed several times during the thruster life test. The vacuum pressure was increased, propellant feed pressure was lowered, and the beam was vectored in an effort to "clean up" the drainage currents. This was an effective solution; however, it is not the procedure that would be used in space. At 930 hr the test was terminated when failure of vacuum equipment led to severe erosion of the thruster module. A six-annular-slit array was tested for 500 hr. Tar formation was present and eventually led to termination of the test. New designs are being studied in an effort to eliminate long-term tar deposits in annular slits.

Neutralizers were tested in conjunction with the 1000-hr 26-needle tests reported in Ref. 21. The neutralizer functioned properly for the first 180 hr of the test when the chamber pressure was increased in an effort to "clean up" the thruster array. However, the neutralizer then burned up due to the presence of atmosphere.

MPD Thrusters

What is an MPD Thruster?

The magnetoplasmadynamic (MPD) arc thruster has evolved through experiments that incorporate arcjet technology and magnetogasdynamic channel flow. The arcjet utilizes a high-current-density discharge between a cathode and anode producing a hot gas that is expelled through a nozzle. The hot gas is ionized (i.e., plasma) and can therefore be accelerated electromagnetically. Magnetogasdynamic channel flow, sometimes referred to as magnetohydrodynamic acceleration (MHD), relies primarily on crossed field ($\mathbf{j} \times \mathbf{B}$) forces to accelerate a plasma. The MPD arc thruster combines an arc discharge with a crossed-field accelerator.

The MPD has been investigated primarily as a steady-state device; however, it may also be pulsed. Pulsed MPD thrusters with pulse lengths of 1 msec or greater are referred to as quasi-steady-state MPD thrusters. The geometry of the quasi-steady-state MPD thruster does not differ greatly from that of a conventional steady-state MPD thruster.

A conceptual diagram of an MPD thruster is presented in Fig. 12. The primary components are the gas flow passages, the anode, the cathode, and the external field coil. Propellant, usually a gas, is fed from a storage tank through a valve (rapid pulse valve for quasi-steady-state device) to the discharge chamber. A discharge takes place from the anode to the cathode. This may be a steady or a quasi-steady discharge. The resultant plasma is then accelerated out the nozzle by the crossed field force arising from the self-field and the discharge current. For low current devices, an external field coil is required. Although the device will operate without an external field (i.e., self-induced magnetic field), low-current density operation tends to be more stable with an external coaxial solenoid. There is no fundamental space-charge limitation on the mass flow density like that arising in pure electrostatic accelerators; therefore, a beam neutralization is not necessary.

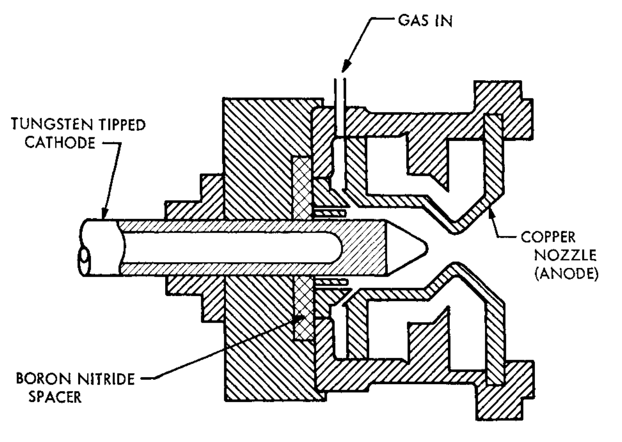

Fig. 12 Conceptual diagram of the MPD thruster.

LeRC Low-Power MPD Thruster

The LeRC low-power MPD arcjet has been under development for several years.[23] The low-power MPD with upstream cathode has been operated with pure gaseous propellants (H_2, Ar, Xe) and biowaste propellants (CO_2-H_2). The most recent device incorporates a downstream cathode to improve thruster performance. Xenon has been used exclusively for tests of the downstream cathode low-power MPD. The thruster operates over a thrust range of 5.8 to 16 mN (1.3 to 3.6 mlbf) with specific impulses of 5880 to 20,590 N-sec/kg (600 to 2100 lbf-sec/lbm). The system is being developed for application to satellite stationkeeping and attitude control. The thruster system does not include power conditioning since it is still in the development stage. Thrust vectoring is implemented by using two pairs of off-center magnets. A slight current bias to a pair of skewed coils will create a slightly skewed magnetic field. The plasma will then be deflected off centerline, resulting in thrust vectoring of the beam. This technique has been demonstrated with an effective beam vectoring of 5 deg. The downstream cathode MPD thruster is presented in Fig. 13.

A schematic diagram of the thruster is presented in Fig. 14. The discharge chamber is surrounded by two electromagnets; however, in most recent tests, only a downstream edge wound electromagnet was required, for which 48 w was required.

Table 3 Plasma thruster performance data

Thruster type	Thrust, μN (μlbf)	Thruster power, W	Specific impulse, N-s/kg (lbf-s/lbm)	Power/thrust, W/mN (W/mlbf)	Specific thrust, μN-s/J (μlbf-s/J)	Engine efficiency, %	Power conditioning efficiency, %	Total power, W
Low-power MPD arc thruster[a]	4181 (940)	193	5296 (540)	46 (206)		5.8	86.0	224.0
	9608 (2160)	345	12,425 (1267)	36 (159)		17.3	86.0	400.0
	14,457 (3250)	625	18,456 (1882)	44 (194)		21.3	86.0	727.0
Fairchild Hiller LES 6 thruster	16.9[b] (3.8)	1.41[c]	3040 (310)	84 (371)	13.3 (3.0)	1.8	56.0	2.52
	17.8[d] (4.0)	1.41[c]	3040 (310)	80 (353)	14.2 (3.2)	1.9	56.0	2.52
	17.3[e] (3.9)	1.41[c]	1863[e] (190)	81 (361)	13.3 (3.0)	1.2	48.0	2.91
Fairchild Hiller LES 7 thruster[f]	398[g] (89.5)	19.6	10,081 (1028)	49 (219.0)	20.5 (4.6)	10.2	85.0	23.0
	3185[h] (716.0)	156.8	10,081 (1028)	49 (219.0)	20.5 (4.6)	10.2	85.0	184.0
Fairchild Hiller high energy pulsed plasma[f]	7325 (1646)[i]	257	12,200 (1245)	35 (156)	28 (6.4)	17.4	85.0	302.0
	6470 (1454)[i]	244	13,000 (1328)	37 (168)	26 (5.9)	17.2	85.0	287.0
	6600 (1483)[j]	309	18,600 (1900)	46 (208)	21 (4.8)	19.9	85.0	364.0
	3515 (790)[i]	190	32,600 (3330)	54 (241)	19 (4.2)	30.2	85.0	224.0
TRW pulsed inductive thruster	5769[k] (1297)	288.0	5884[l] (600)	50 (222)	20.0 (4.5)	5.9	85.0	339.0
	5769[k] (1297)	288.0	19,613 (2000)	50 (222)	20.0 (4.5)	19.6	85.0	339.0
	3674[m] (826)	200.0	5884[l] (600)	55 (242)	18.0 (4.1)	5.4	85.0	235.0
	3674[m] (826)	200.0	19,613 (2000)	55 (242)	18.0 (4.1)	17.0	85.0	235.0
	1833[n] (412)	128.0	5884[l] (600)	70 (311)	14.0 (3.2)	4.2	85.0	150.6
	1833[n] (412)	128.0	19,613 (2000)	70 (311)	14.0 (3.2)	14.0	85.0	150.6

[a] Includes 48 W for edge wound magnet power and 2 W for a solenoid valve.
[b] Reference 26 at 0.667 pps.
[c] 0.1 W for telemetry.
[d] Reference 28 at 0.667 pps.
[e] Tom Williams, NASA Goddard, at 0.69 pps; one million pulses
[f] Supplied by W. J. Guman, Fairchild-Hiller.
[g] At 1 pps.
[h] At 8 pps.
[i] At 0.531 pps.
[j] At 0.971 pps.
[k] 12 kV on 4 μF capacitor at 1 pps.
[l] Corrected for 30% mass utilization.
[m] 10 kV on 4 μF capacitor at 1 pps.
[n] 8 kV on 4 μF capacitor at 1 pps.

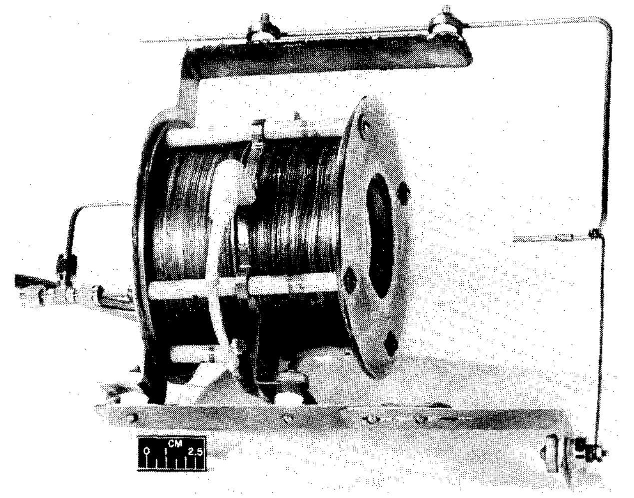

Fig. 13 NASA-Lewis low-power MPD thruster (photo courtesy of LeRC).

A pole piece is incorporated to intensify the magnetic field in the discharge chamber. A hollow cathode is placed in the exhaust beam downstream of the discharge chamber. The development thruster does not include a feed system, but the propellant feed system should be identical to a xenon inert gas system. Taking advantage of the compressibility of xenon at 1.0×10^7 N/m^2 (1500 psia), a pound of gas requires about 0.14 kg (0.3 lbm) of tankage. Solenoid valves have been developed for inert gas systems (e.g., Mariner) and could be used in the MPD feed system. Leakage of gaseous propellant through the solenoid is a problem, particularly during long-term storage in space. Thruster mass is approximately 2.2 kg (5 lbm) excluding the propellant feed system. Power conditioning is estimated at 3.2 kg (7 lbm) with an efficiency of 86%.

The system is capable of 4.4- to 18-mN (1- to 4-mlbf) thrust. Thrust level and efficiency increase with specific impulse. Low-power MPD performance data are presented in Table 3.[23] Power to the magnet can be reduced for low-thrust, low-specific-impulse operation (12 w). Since a neutral plasma is ejected from the thruster, no neutralizer is required. A power of 1 to 3 w is required for the solenoid valve in the feed system.

At present, this system has not achieved long life. Endurance and erosion of an upstream hollow cathode configura-

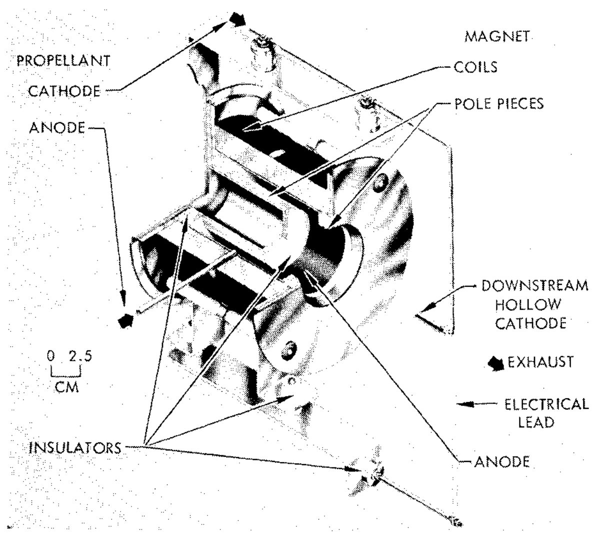

Fig. 14 Schematic of low-power MPD arc thruster with a downstream cathode.

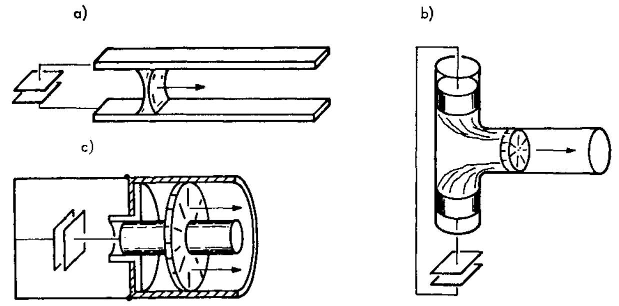

Fig. 15 Various pulsed plasma accelerators: a) parallel-rail accelerator, b) T tube, and c) coaxial gun.[25]

tion were conducted with argon as the propellant.[24] After 638 hr of operation, the cathode tip had eroded 1.33 cm, leading to failure of the thruster. The cathode erosion rate has been verified by several additional life tests. At present, this form of erosion seems to be the thruster life determinant. Life tests of the downstream cathode have not yet been performed; however, cathode life is not expected to differ greatly from the previous life test data.

The initiation of thruster discharge is a problem, and at present a lightweight arc initiator for space application has not been developed. An experimental lightweight arc initiator is presently under study at LeRC.

Pulsed Plasma and Pulsed Inductive Thrusters

What are Pulsed Plasma and Pulsed Inductive Thrusters?

The pulsed plasma thruster is a device that uses a "burst" of electrical energy to produce, accelerate, and eject a plasma wave. Pulsed plasma accelerators have electrodes in direct contact with the gas, thus requiring a zone of conducting plasma to complete the circuit, while pulsed inductive accelerators have a complex external circuit that induces a current in the gas flow. Pulsed plasma acceleration has been studied most widely, although there is some research in pulsed inductive acceleration.

Pulsed plasma devices are usually classified by their discharge chamber geometry. The parallel rail accelerator in Fig. 15a is the simplest pulsed plasma device. Two other pulsed plasma geometries are the "T tube" (Fig. 15b) and the "coaxial gun" (Fig. 15c). The pulsed inductive thruster depicted in Fig. 16 is called a "loop inductor" pulsed inductive accelerator.

The typical pulsed plasma thruster is presented in Fig. 17. The propellant, a solid, liquid, or gas, is placed in the discharge gap. Solid propellant is maintained in the gap region by mechanical springs, liquid is fed by capillary action, and gas is placed in chamber with a gas injection valve. A capacitor or inductor is required for energy storage. The anode-cathode configuration will vary as those presented in Fig. 15; however, the rail-type geometry is presented here. As for the MPD plasma thruster, no neutralizer is required for the pulsed plasma thruster.

The pulsed inductive thruster is similar; however, the capacitor discharge is through an inductor loop as in Fig. 16.

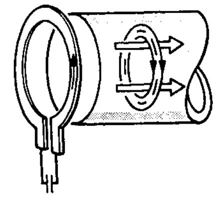

Fig. 16 Pulsed inductive accelerator, loop inductor geometry.[25]

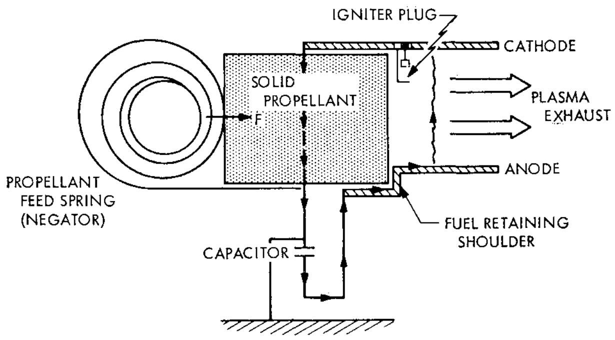

Fig. 17 Conceptual diagram of the pulsed plasma thruster.

A spark gas is required to initiate the main capacitor discharge. The propellant, usually a gas, is fed to the inductor face with a gas injection valve simultaneous to the inductor pulsing.

Space Experience of Pulsed Plasma Thrusters

A solid-propellant pulsed plasma propulsion system having four thrusters was launched aboard the MIT Lincoln Lab. LES-6 Satellite in September 1968.[26] After more than 2 years in space, the pulsed plasma thrusters continue to function well. In a simulated life test at vacuum facilities, intermittency or failure to discharge capacitor energy was observed. The intermittency would not correct itself and the unit stopped firing completely. After more than 3 million pulses on each of the four flight thrusters, intermittency has been observed in three of the four thruster units. Complete failure (100% intermittency) has occurred to only one thruster unit. More recent designs have corrected this intermittency problem. The propulsion system has functionally kept the LES-6 Satellite within the $\pm 2°$ longitude design band.

Fairchild Hiller LES-6 Pulsed Plasma Thruster

Republic Aviation Division of Fairchild Hiller Corp. has developed a solid-propellant pulsed plasma thruster for the MIT Lincoln Lab., LES-6.[26, 28] This thruster is capable of providing a 27-μN (6-μlbf-sec) impulse bit per discharge of the 2-μF capacitor. The flight system is charged to 1360 v, which is equivalent to a 1.85-joule discharge. A solid block of Teflon is used as propellant. The thruster system is complete with feed system, power conditioning, and telemetry circuits. This system does not have thrust vectoring capability. The LES-6 thruster is presented in Fig. 18.

A schematic diagram of the solid-propellant LES-6 pulsed plasma thruster is presented in Fig. 17. A small capacitor discharge ($\frac{1}{8}$ joule) across the spark igniter provides sufficient plasma to initiate a discharge between the anode and cathode. The main discharge evaporates, ionizes, and accelerates Teflon propellant. A spring is employed to feed the Teflon to the thruster. A retaining shoulder maintains the propellant between the anode and cathode. Open failure of valved liquid or gaseous systems can lead to forces or moments applied to the spacecraft. Solid Teflon propellant remains passive when not used, or in the event of a failure. The LES-6 thruster configuration employs one 2-μF capacitor and two separate anode, cathode, spark igniter and propellant subsystems. The two complete subsystems provide redundancy with little addition to total mass. Each thruster system, including case and Teflon propellant, has a mass of 1.4 kg (3.0 lbm). Each thruster includes 0.1 kg (0.3 lbm) of Teflon, which provides 280 N-sec (64 lbf-sec) total impulse. A power conditioner was developed which provides a 1.32 w of power to the main and igniter capacitors of any of four thruster units. The conditioner has a mass of 0.9 kg (1.9 lbm) and has an efficiency of 56%.

The thruster can operate at several discharge energies and different pulse rates. The nominal discharge energy of 1.85 joules provides 25- to 27-μN-sec (5.7- to 6.0-μlbf-sec) impulse bit per discharge is achieved for this thruster.[29] Typical performance data are presented in Table 3.

Radio frequency interference noise has been observed on the LES-6 thrusters. Measurements of this noise are discussed in Ref. 30. This noise has not affected normal operation of the LES-6 satellite.

Prototypes of the LES-6 thruster designed in 1968 were tested extensively in the laboratory.[26] Ground testing was performed at 1 to 4 pps. On several tests the thruster achieved more than 8 million discharges without failure of the capacitor or discharge initiator. Intermittency of discharge to occur is common with the flight thruster after 1 to 2 million discharges. An intermittent thruster does not draw power, and thus the capacitor can be discharged later. However, when the intermittency becomes more frequent the thruster may never recover. During tests of the LES-6 pulsed plasma thruster at Lincoln Labs., one thruster became intermittent after 1 million pulses and did not recover.[31]

Fairchild Hiller LES-7 Pulsed Plasma Thruster.

This thruster was developed for use on the LES-7; however, the planned spacecraft has been cancelled.[32] This thruster, which is a scaled-up version of the LES-6 thruster, is capable of providing a 270- to 400-μN (60- to 90-μlbf-sec) impulse bit per discharge of the capacitor. The performance of the device increases with discharge energy. The thruster is complete with feed system. The two thrusters per capacitor are skewed with respect to each other, providing thrust vector control. The LES-7 thruster system is depicted in Fig. 19.

Features of the thruster are similar to those described in the LES-6 system. Two igniter plugs are provided in each of the two nozzles and the parallel accelerators are skewed 30° to each other as depicted in Fig. 19. Total thruster mass is 4.8 kg (10.5 lbm) with 0.7 kg (1.6 lbm) of Teflon propellant included in each system. A 1.3 kg (2.9 lbm) power

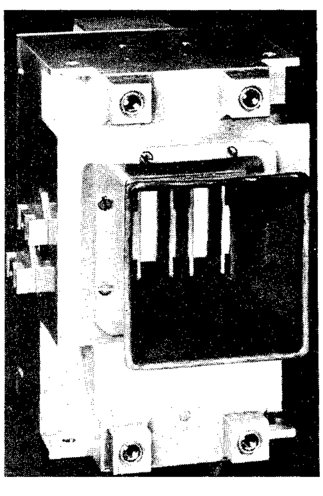

Fig. 18 Fairchild-Hiller LES-6 pulsed plasma thruster (photo courtesy of Fairchild-Hiller).

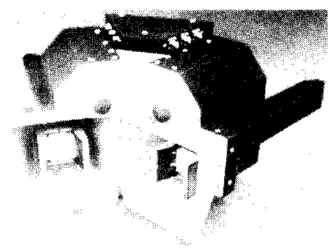

Fig. 19 LES-7 pulsed plasma thruster (photo courtesy of Fairchild-Hiller).

conditioner with 85% electrical efficiency is estimated[33] to be capable of providing 20 to 40 w of conditioned power to the thruster.

Typical performance data are presented in Table 3. The specific impulse of this thruster is near 9800 N-sec/kg (1000 lbf-sec/lbm), three times greater than that of the LES-6 thruster. This improvement was due to increased discharge energy, increased Teflon surface area, and small changes in nozzle geometry. Included also in the performance table are four experimental data points from a high-power, high-impulse-bit, pulsed plasma thruster for north-south stationkeeping. The four high-discharge energy data points are experimental and are averaged over at least 4 hr of continuous operation. The two data points with high specific thrust employ side feed of Teflon.

Radio frequency interference (RFI) noise seems to increase with discharge energy; RFI measurements from an experimental thruster similar to the LES-7 pulsed plasma thruster indicate a much higher level of noise than observed in the LES-6 thrusters.[34]

Life tests of thrusters resembling the LES-7 thruster configuration have been performed.[35] These tests suggest that 5 to 10 million pulses can be achieved without thruster failure. The problem of feeding large quantities of Teflon to the high-power, high-impulse-bit thruster has been analyzed and shown to be feasible.[32] Coiling of Teflon propellant and feeding by springs has been suggested. However, the problem of possible Teflon binding in the coiled position after long storage times is presently unresolved. Some life testing of this feed system is required before it can be used in space.

TRW Pulsed Inductive Thruster

TRW Systems Group has developed a pulsed inductive thruster.[36] This thruster, shown schematically in Fig. 20, uses a variety of gaseous propellants; however, xenon will be discussed herein. A fast-opening gas-injection valve is employed to feed propellant to the thruster. The propellant does not form a zone of conducting plasma to complete the primary circuit as in pulsed plasma thrusters. A 20-cm (8-in.) flat spiral coil is used to induce a current sheet in the propellant. A 4-μF capacitor charged to a voltage of 6 to 12 kv is discharged across the spiral coil. This thruster provides a nominal 4.45-mN-sec (1-mlbf-sec) impulse bit per discharge. The thruster system is complete with gas injection valve, field coil, ignition circuit, and capacitor.

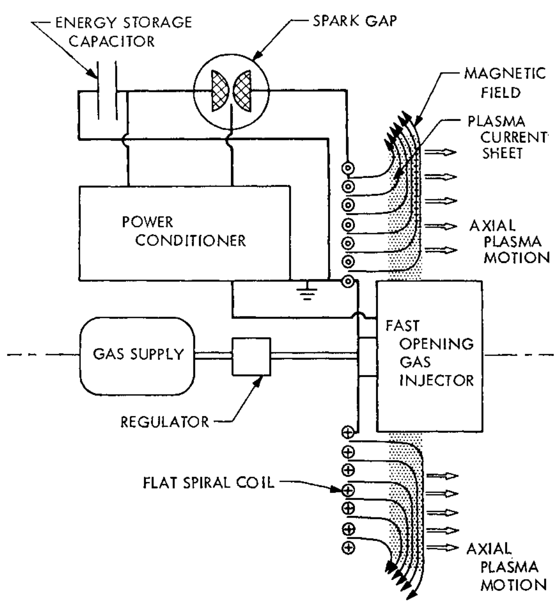

Fig. 20 Pulsed inductive thruster schematic.

Gas is fed from a storage tank to a small plenum upstream of the gas-injection valve. A small capacitor is discharged across the fast-acting valve along with the initiation of a delay generator. A variable delayed trigger pulse is sent to the spark gap several hundred μsec after valve initiation, when gas distribution over the coil is at a maximum. The capacitor then discharges through the spark gap and thruster coil. The mass utilization, or percentage of injected gas distributed over the coil during the acceleration phase, is only 30%. This is due to the long valve actuation time ($\sim$800 μsec) compared to the short time for the thruster acceleration process (10 μsec). A delay of about 600 μsec before thruster acceleration begins is typical for this system. A complete thruster package including 4-μF capacitor, thruster valve, 20-cm coil, trigger, and solenoid capacitor has a mass of 11.4 kg (25.0 lbm). A mass of 0.1 kg (0.3 lbm) can be estimated for a regulator, and 0.1 kg (0.3 lbm) of tankage can be assumed for each 0.5 kg (1 lbm) of Xe propellant at 10×10^6 N/m^2 (1500 psia). Power conditioning mass is estimated at 3.1 kg (6.8 lbm) with an efficiency of 85%.

The thruster can operate at several discharge energies and different pulse rates. The nominal discharge energy of 200 joules (10 kv) provides 3.7-mN-sec (826-μlbf-sec) impulse bit per discharge.[36] A specific impulse of 19,600 N-sec/kg (2000 lbf-sec/lbm) has been measured; however, it should be corrected to include the 30% mass utilization efficiency. Typical performance data are presented in Table 3.

As of this writing, no RFI noise or life test data are available. Present efforts are underway to improve the performance and mass utilization of this thruster.

Conclusion Based on Demonstrated Life, Mass, Performance and Power

The spacecraft systems engineer, when faced with the selection or integration of an auxiliary-electric propulsion system for application to a mission, will first ask: What is the system mass? What is the thrust level? How much total impulse has it demonstrated? How much power does it require? These questions can best be answered by summarizing the contents of the preceding auxiliary-electric thruster survey. The thruster systems surveyed can be divided into two groups: microthrusters and north-south stationkeeping thrusters.

Microthrusters are low-thrust (e.g., 0 to 223-μN) devices capable of providing thrust for spacecraft attitude control, orientation, east-west stationkeeping, and momentum wheel unloading. The total impulse requirements of microthruster systems may be as much as 4400 N-sec (1000 lbf-sec).

The second group of auxiliary-electric thrusters is the one considered for north-south station keeping and other spacecraft propulsion requirements such as orbit modifications. Satellite size will serve to define a minimum thrust for north-south stationkeeping thrusters. For each 454 kg (1000 lbm) of spacecraft, a minimum of 670 μN (150 μlbf) thrust is required if the north-south stationkeeping thruster is on 100% of the time. Thus north-south stationkeeping thrusters fall into a thrust range of 450 μN to 4.5 mN (100 μlbf to 1 mlbf), where the low-thrust range is applicable only to satellites below 454 kg (1000 lbm). A total impulse of 21,000 N-sec (4730 lbf-sec) per year is required for each 454 kg (1000 lbm) of spacecraft mass. If orbit corrections are required, the time required for orbit correction and the amount of spacecraft movement will define thruster requirements.[37]

Thruster systems are compared on plots of total thruster system mass as a function of the total impulse required (Figs. 21 and 22). The total mass shown includes a power penalty

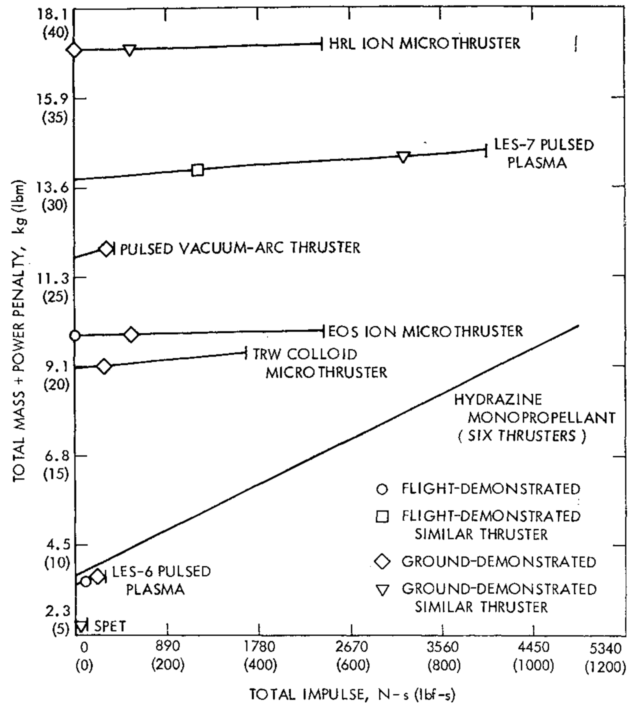

Fig. 21 Mass of microthruster systems (two thrusters per system, each capable of the plotted total impulse).

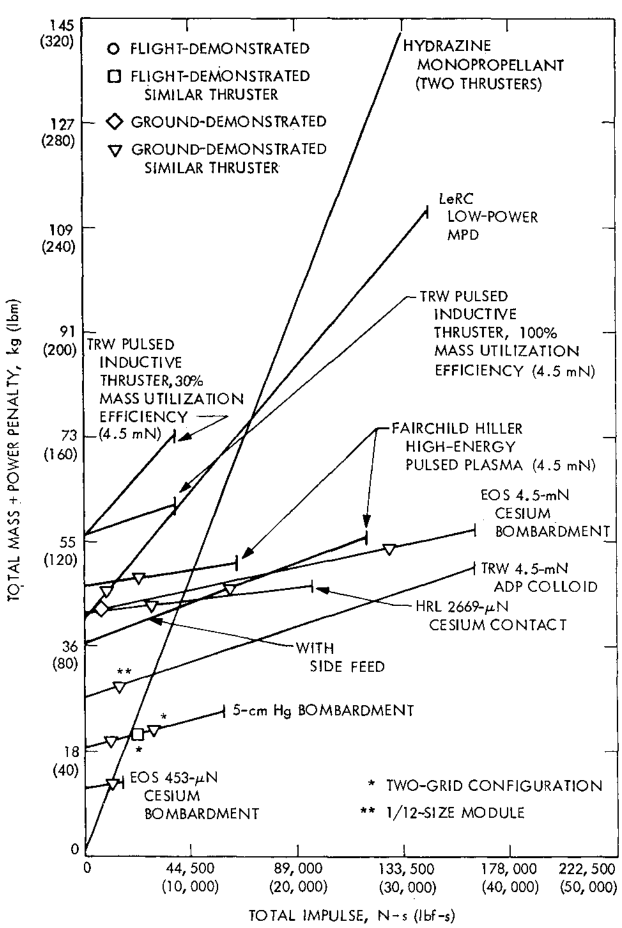

Fig. 22 Mass of complete north-south stationkeeping thruster systems (two thrusters per system, each capable of the plotted total impulse).

of 0.1 kg/w (solar cells at ~3 w/lbm). Shown for comparison on these plots is a 450-mN (100-mlbf) monopropellant hydrazine system sized for the same functions. All electric thruster systems include two complete thrusters, power conditioning, and propellant. The demonstrated total impulse has been separated into four categories: 1) flight-demonstrated thruster life, 2) flight-demonstrated life of a similar-type thruster, 3) ground-simulated life tests in high vacuum, and 4) ground-simulated life tests of a similar-type thruster.

The total impulse plotted is per thruster. Since there are two thrusters per system, the spacecraft will therefore receive twice the plotted total impulse, if both thrusters use all their propellant. For a comparison of thruster total impulse capability, each thruster mass line has been terminated with an end point that corresponds to 10,000 hr for steady-state thrusters and 10,000,000 pulses for pulsed thrusters. The life estimate of the high-energy plasma thrusters was increased to 20,000,000 pulses.

Auxiliary-electric microthrusters surveyed include the Hughes Research Lab. and Electro Optical Systems cesium contact-ion thrusters, the TRW Systems Group colloid microthruster, the Fairchild Hiller LES-6 and LES-7 pulsed plasma thrusters, the General Electric SPET, and the Pulsed Vacuum-Arc thruster. Several conclusions can be drawn on microthruster systems (Fig. 21): 1) The ion (HRL and EOS) and colloid microthrusters in general have greater fixed mass (including penalty for power) than the pulsed plasma microthrusters. 2) Demonstrated life of the pulsed plasma system exceeds the other auxiliary-electric microthrusters. 3) For three-axis-stabilized spacecraft, either two electric microthrusters with beam deflection (e.g., ion microthrusters), or six monopropellent hydrazine thrusters could provide east-west station keeping. 4) The mass of the hydrazine catalyst system with six thrusters is less than the mass of any of the electric microthrusters, with the exception of the low-total-impulse (450-N-sec) pulsed plasma thrusters. 5) Pulse plasma thrusters should be considered for spinning satellites and other spacecraft that require extensive pulsing.

Auxiliary-electric thrusters surveyed, which are suitable for north-south stationkeeping, are the HRL 2.7-mN (600-μlbf) linear strip ion thruster, the EOS 450-μN (100-μlbf) and 4.5-mN (1-mlbf) cesium bombardment thrusters, the HRL and LeRC 5-cm Hg bombardment thruster, the TRW 4.5-mN (1-mlbf) ADP colloid thruster, the LeRC low-power MPD, Fairchild Hiller's experimental pulsed plasma, and TRW's pulsed inductive thruster. The following conclusions are drawn for auxiliary-electric north-south stationkeeping thrusters (Fig. 22):

1) The EOS 450-μN (100-μlbf) cesium bombardment thruster for low-total-impulse requirements (small satellites) offers a small mass advantage over monopropellant hydrazine, when the thruster is sized for its assumed maximum capability. Below that size, it is heavier than an equivalent hydrazine system. 2) The 5-cm mercury bombardment thruster offers some improvements over a hydrazine system above 22,250 N-sec (5000 lbf-sec) total impulse. Present demonstrated life of glass grids is less than 22,250 N-sec (5000 lbf-sec) total impulse; however, the SERT II life tests indicate possible long life with a two-grid arrangement. 3) The 4.5-mN (1-mlbf) colloid thruster offers the greatest mass savings of any of the auxiliary-propulsion systems plotted for missions with total impulse requirements of more than 66,750 N-sec (15,000 lbf-sec). The mass of the 4.5-mN (1-mlbf) colloid thruster is estimated since this thruster has not yet been fabricated. Testing of the high thrust colloid to date has been limited to a 1000-hr test of a $\frac{1}{12}$-size thruster. 4) The 2.7-mN (600-μlbf) dual-beam cesium-contact thruster system offers some mass advantages over a monopropellant; however, its high power-to-thrust ratio penalizes this system. 5) The 4.5-mN (1-mlbf) cesium bombardment thruster has been tested for only about 500 hr; however, previous tests

of a slightly larger thruster exceeded 8000 hr. The 4.5-mN (1-mlbf) cesium bombardment thruster has promise of large total impulse life. 6) The low-power MPD thruster was compared at a thrust and power point that unfortunately is accompanied by a low specific impulse. Higher-power operation places an excess power penalty on the system. Development of an MPD with maximum performance at 4.5-mN (1-mlbf) thrust will make this thruster more attractive for north-south stationkeeping. The present optimum MPD thrust of 9 to 13 mN (2 to 3 mlbf) is attractive only for very large satellites. 7) The experimental pulsed plasma thrusters are heavier in general than the ion and colloid thrusters but may offer more total impulse life. 8) The total impulse of the pulsed plasma thruster is dependent on capacitor and gas injection valve technology. These thrusters are each plotted with an assumed life of 10^7 pulses per thruster and 2×10^7 pulses per capacitor. The figure of 10^7 pulses for the TRW pulsed inductive thruster is based on estimates of gas injection valve/spark gap life. The Fairchild Hiller thruster life is based on 2×10^7 discharges of a capacitor with specific mass of 22 joule/kg (10 joule/lbm). Without attainment of this assumed specific mass or an increase in the number of discharges possible, the high-energy pulsed plasma system will become non-competitive on a mass basis. These conclusions are based on mass, power, and performance considerations only and do not include thruster cost, reliability, and spacecraft/thruster interactions.

Conclusions Based on Cost and Reliability

Cost and reliability are important factors in the selection of an auxiliary-propulsion system. There is not enough information on costs or component reliabilities to make detailed estimations of either cost or reliability of auxiliary electric thrusters, but qualitative estimates can be made.

Hardware cost of a microthruster system (one thruster with associated power conditioning) is approximately \$100,000, while the hardware cost of a north-south stationkeeping thruster system is approximately \$150,000. Flight qualification and thruster testing requirements really set the thruster costs. The extent of flight qualification and thruster testing will vary with the mission as well as the thruster.

Without component reliabilities based on sufficient test data, it is impossible to compute realistic quantitative reliabilities for electric thrusters. However, by comparing critical components and subsystems of electric thrusters, a qualitative reliability ranking (highest reliability listed first) can be estimated: 1) pulsed plasma, solid propellant; 2) pulsed plasma, gaseous propellant; 3) magnetoplasmadynamic (MPD); 4) ion; and 5) colloid.

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