Electric Propulsion

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GLOSSARY

Arcjet Device that heats a propellant stream by passing a high-current electrical arc through it, before the propellant is expanded through a downstream nozzle.

Hall effect Conduction of electric current perpendicular to an applied electric field in a superimposed magnetic field.

Inductive thruster Device that heats a propellant stream by means of an inductive discharge before the propellant is expanded through a downstream nozzle.

Ion thruster Device that accelerates propellant ions by an electrostatic field.

Magnetoplasmadynamic thruster Device that accelerates a propellant plasma by an internal or external magnetic field acting on an internal arc current.

Plasma Heavily ionized state of matter, usually gaseous, composed of ions, electrons, and neutral atoms or molecules, that has sufficient electrical conductivity to carry substantial current and to react to electric and magnetic body forces.

Resistojet Device that heats a propellant stream by passing it through a resistively heated chamber before the propellant is expanded through a downstream nozzle.

Thrust Unbalanced internal force exerted on a rocket during expulsion of its propellant mass.

THE SCIENCE AND TECHNOLOGY of electric propulsion (EP) encompass a broad variety of strategies for achieving very high exhaust velocities in order to reduce the total propellant burden and corresponding launch mass of present and future space transportation systems. These techniques group broadly into three categories: electrothermal propulsion, wherein the propellant is electrically heated, then expanded thermodynamically through a nozzle; electrostatic propulsion, wherein ionized propellant particles are accelerated through an electric field; and electromagnetic propulsion, wherein current driven through a propellant plasma interacts with an internal or external magnetic field to provide a

stream-wise body force. Such systems can produce a range of exhaust velocities and payload mass fractions an order of magnitude higher than that of the most advanced chemical rockets, which can thereby enable or substantially enhance many attractive space missions. The attainable thrust densities (thrust per unit exhaust area) of these systems are much lower, however, which predicates longer flight times and more complex mission trajectories. In addition, these systems require space-borne electric power supplies of low specific mass and high reliability, interfaced with suitable power processing equipment. Optimization of EP systems thus involves multidimensional trade-offs among mission objectives, propellant and power plant mass, trip time, internal and external environmental factors, and overall system reliability. An enduring international program of research and development of viable electric thrusters has been in progress for several decades, and over the past few years this has led to the increasing use of a number of EP systems on commercial and governmental spacecraft. Meanwhile, yet more advanced EP concepts have matured to high credibility for future mission applications.

I. CONCEPTUAL ORGANIZATION AND HISTORY OF THE FIELD

A. Motivation

The stimulus for development of electrically driven space propulsion systems is nothing less fundamental than Newton's laws of dynamics. Since a rocket-propelled spacecraft in free flight derives its only acceleration from discharge of propellant mass, its equation of motion follows directly from conservation of the total momentum of the spacecraft and its exhaust stream:

$$m\dot{v} = \dot{m}v_e, \qquad (1)$$

where m is the mass of the spacecraft at any given time, $\dot{v}$ its acceleration vector, v_e the velocity vector of the exhaust jet relative to the spacecraft, and $\dot{m}$ the rate of change of spacecraft mass due to propellant-mass expulsion. The product $\dot{m}v_e$ is called the thrust of the rocket, T, and for most purposes can be treated as if it were an external force applied to the spacecraft. Its integral over any given thrusting time is usually termed the impulse, I, and the ratio of the magnitude of T to the rate of expulsion of propellant in units of sea-level weight, $\dot{m}g_o$, has historically been labeled the specific impulse, $I_s = v_e/g_o$. If v_e is constant over a given period of thrust, the spacecraft achieves an increment in its velocity, Δv, which depends linearly on

v_e and logarithmically on the amount of propellant mass expended:

$$\Delta v = v_e \ln \frac{m_o}{m_f}, \qquad (2)$$

where m_o and m_f are the total spacecraft mass at the start and completion of the acceleration period. Conversely, the deliverable mass fraction, m_f/m_o, is a negative exponential in the scalar ratio $\Delta v/v_e$:

$$\frac{m_f}{m_o} = e^{-\Delta v/v_e}. \qquad (3)$$

Inclusion of significant gravitational or drag forces on the flight of the spacecraft adds appropriate terms to Eq. (1) and considerably complicates its integration, but it is still possible to retain relation (3), provided that Δv is now regarded as a more generalized "characteristic velocity increment," indicative of the energetic difficulty of the particular mission or maneuver. However represented, the salient point is simply that if the spacecraft is to deliver a significant portion of its initial mass to its destination, the rocket exhaust speed must be comparable to this characteristic velocity increment. Clearly, for missions of large Δv, the burden of thrust generation must shift from high rates of ejection of propellant mass to high relative exhaust velocities. Unfortunately, conventional chemical rockets, whether liquid or solid, monopropellant or bipropellant, are fundamentally limited by their available combustion reaction energies and heat transfer tolerances to exhaust speeds of a few thousand meters per second, whereas many attractive space missions entail characteristic velocity increments at least an order of magnitude higher. Thus, some fundamentally different concept for the acceleration of propellant mass that circumvents the intrinsic limitations of chemical thermodynamic expansion is required. Into this breech step the family of electric propulsion possibilities.

B. Conceptual Subdivision

So that propellant exhaust speeds in the range above 10,000 m/sec desirable for interplanetary flight and other high-energy missions can be obtained, processes basically different from nozzled expansion of a chemically reacting flow must be invoked. More intense forms of propellant heating may be employed, provided that the walls of the rocket chamber and nozzle are protected from excessive heat transfer. Alternatively, the thermal expansion route may be bypassed completely by direct application of suitable body forces to accelerate the propellant stream. Either of these options is most reasonably accomplished by electrical means, which constitute the technology of electric propulsion.

Historically, conceptually, and pragmatically, this field has tended to subdivide into three categories:

1. *Electrothermal propulsion*, wherein the propellant is heated by some electrical process, then expanded through a suitable nozzle
2. *Electrostatic propulsion*, wherein the propellant is accelerated by direct application of electrostatic forces to ionized particles
3. *Electromagnetic propulsion*, wherein the propellant is accelerated under the combined action of electric and magnetic fields

Over their periods of development, each of these approaches has spawned its own array of technical specialties and subspecialties, its own balance sheet of advantages and limitations, and its own cadres of proponents and detractors, but in serious assessment, each has validly qualified for particular niches of application, many of which do not seriously overlap. Throughout the history of EP development, the original subdivision of the field into electrothermal, electrostatic, and electromagnetic systems has remained useful, and this subdivision will be respected through the balance of this article. It should be recognized, however, that in virtually all practical systems, two or even all three of these processes function in some concert to accelerate, channel, and expand the propellant flow, and in many cases it is the efficacy of this cooperation that determines the utility of any given device.

The exhaust velocities attainable by these methods, especially the latter two, are more than adequate for many large-velocity-increment missions beyond the viable chemical range. Indeed, some restraint of their v_e capability may be required because of their associated "power supply penalty." Clearly, each of these concepts entails two functional components: the thruster itself and an electric power supply to drive it. The latter adds mass, m_p, to the composite propulsion system in some proportion to the power level of operation, P, which in turn scales with the square of the exhaust velocity:

$$m_p = \alpha P = \frac{\alpha T v_e}{2\eta} = \frac{\alpha \dot{m} v_e^2}{2\eta}, \qquad (4)$$

where α is the specific mass of the power supply (mass per unit power), and η is the efficiency with which the thruster converts its input power to thrust power, $T v_e/2$. Since the requisite propellant mass scales inversely with v_e, it follows that for any given mission requirement, Δv, there is an optimum v_e that minimizes the sum of the propellant mass and that of the requisite power supply. Relation (4) also emphasizes the importance of utilizing power systems of low specific mass and thrusters of high conversion efficiency. Overlaid on all this is the evident necessity for impeccable reliability of both components of the system over long periods of unattended operation in the space environment.

C. History of Effort

The attractiveness of EP for a broad variety of space transportation applications was recognized by the patriarch of modern rocketry, Robert H. Goddard, as early as 1906. His Russian counterpart, Konstantin Tsiolkovskiy, proposed similar concepts in 1911, as did the German Hermann Oberth in his classic book on spaceflight in 1929 and the British team of Shepherd and Cleaver in 1949. But the first systematic and tutorial assessment of EP systems should be attributed to Ernst Stuhlinger, whose book *Ion Propulsion for Space Flight* nicely summarizes his seminal studies of the 1950s.

The rapid acceleration of the U.S. space ambitions in the 1960s drove with it the first coordinated research and development programs explicitly addressing EP technology. In its earliest phase, this effort drew heavily on reservoirs of past experience in other areas of physical science and engineering that had employed similar electrothermal, electrostatic, and electromagnetic concepts to their own purposes, such as arc-heated wind tunnels and welding practice, cathode ray tubes and mass-spectroscopic ion sources, and magnetohydrodynamic channel flows and railguns. From these transposed technologies blossomed a significant new component of the burgeoning space industry that concerned itself not only with the development of viable electric thrusters, but also with the provision of suitable electric power supplies and power conditioning equipment, major ground test facilities, and sophisticated mission analyses of a smorgasbord of potential space applications.

Following a sizable number of experimental flight tests, EP entered its era of commercial application in the early 1980s, as resistojets became common options for station keeping and attitude control on tens of commercial spacecraft. In the early 1990s, electrothermal arcjets were adopted for north–south station keeping (NSSK) of many communication satellites in geosynchronous earth orbit (GEO). The year 1994 saw the first use of electrostatic ion thrusters for the NSSK of commercial satellites, and the year 1998 their application on a planetary NASA mission. Although Hall thrusters have been used on Soviet and Russian spacecraft since the mid-1970s, and there have been a few applications of pulsed plasma thrusters, electromagnetic thrusters are only now entering their era of application on Western commercial spacecraft. In total, the number of electrically propelled spacecraft has gone from single digits in the 1960s to double digits in the 1970s and 1980s and has reached the triple-digit mark

in the late 1990s. A recent emphasis in research and development has been the scaling down, in both physical size and power level (<100 W), of many EP concepts for future applications on micro-spacecraft. At the other extreme, the prospect of energetic missions—with large cargo and piloted payloads—to the planets, which stand to benefit most from EP, remains futuristic until the required high power levels (100 kW and above) become available in space.

II. ELECTROTHERMAL PROPULSION

A. Overview

Electrothermal propulsion comprises all techniques whereby the propellant is electrically heated in some chamber and then expanded through a suitable nozzle to convert its thermal energy to a directed stream that delivers reactive thrust power to the vehicle. Three subclasses of this family may be denoted in terms of the physical details of the propellant heating:

1. *Resistojets*, wherein heat is transferred to the propellant from some solid surface, such as the chamber wall or a heater coil
2. *Arcjets*, wherein the propellant is heated by an electric arc driven through it
3. *Inductively and radiatively heated devices*, wherein some form of electrodeless discharge or high-frequency radiation heats the flow

Each of these strategies relieves some of the intrinsic limitations of the chemical rocket in the sense that the propellant species may be selected for its propitious physical properties independently of any combustion chemistry, but heat transfer constraints and frozen flow losses (losses due to unrecuperated energy "frozen" in the internal modes and dissociation of the molecules) remain endemic.

The gross performance of any electrothermal thruster can be crudely forecast by means of a rudimentary one-dimensional energy argument that limits the exhaust speed of the flow from a fully expanded nozzle to $v_e \leq \sqrt{2 c_p T_c}$, where c_p is the specific heat at constant pressure per unit mass of the propellant and T_c is the maximum tolerable chamber temperature. Propellants of the lowest molecular weight thus seem preferable, and indeed hydrogen might at first glance appear optimum, but in practice its frozen flow propensities and difficulty of storage compromise its attractiveness. More complex molecular gases such as ammonia and hydrazine, which dissociate into fairly low effective molecular weights and high specific heat gas mixtures in the chamber, are currently more popular, but in

these cases also, frozen flow kinetics in the nozzle remain important to performance.
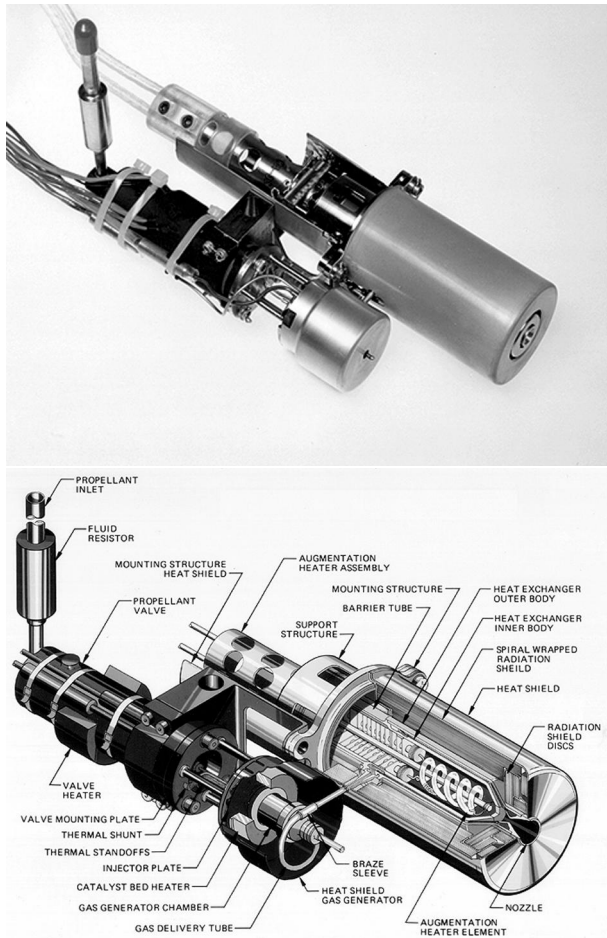
B. Resistojets

In the resistojet subclass of devices, chamber temperature is necessarily limited by the materials of the walls and/or heater coils to some $3000°$K or less, and hence the exhaust velocities, even with equilibrated hydrogen, cannot exceed 10,000 m/sec, which is nonetheless a factor of two or three beyond that of the best chemical rockets. In contemporary practice, lower performance but more readily space storable propellants, such as hydrazine and ammonia, along with biowaste gases such as water vapor and carbon dioxide, are more commonly employed because of their overall system advantages.

Beyond the frozen flow kinetics, the major practical challenge facing resistojet technology is retaining the integrity of the insulator and heater surfaces at the very high temperatures the concept demands, while still minimizing the viscous and radiative heat losses that further decrease thruster efficiency. Since the mid-1960s, many configurations of resistojet have been conceived, researched, and developed to optimize these processes, and a few, such as the flight-ready module shown schematically and in the photograph in Fig. 1, have evolved to practical space thrusters and been deployed on suitable missions. A typical resistojet uses catalytically decomposed hydrazine as its propellant and achieves an exhaust velocity of 3500 m/sec and a thrust of 0.3 N at an efficiency of 80% when operating at a power level of 750 W.

From a system point of view, resistojets are particularly attractive because they readily lend themselves to integration with previously developed and commonly used propellant storage and flow management systems for hydrazine monopropellant thrusters. Another advantage is their low operational voltage, which, unlike that in other EP systems, does not require complex power processing. For these reasons, and the fact that satellites in GEO often have excess electrical power, resistojets were among the first EP options to be used for the NSSK of communication satellites. While the earliest use of resistojets in space dates to 1965 (the Air Force Vela satellites), their adoption on commercial spacecraft did not start until the 1980 launch of the first satellites in the INTELSAT-V series. A more recent application has been for orbit insertion, attitude control, and deorbit of LEO satellites, including the 72 satellites in the Iridium constellation.

C. Arcjets

If an electrothermal thruster is to attain exhaust speeds substantially higher than 10,000 m/sec, interior portions
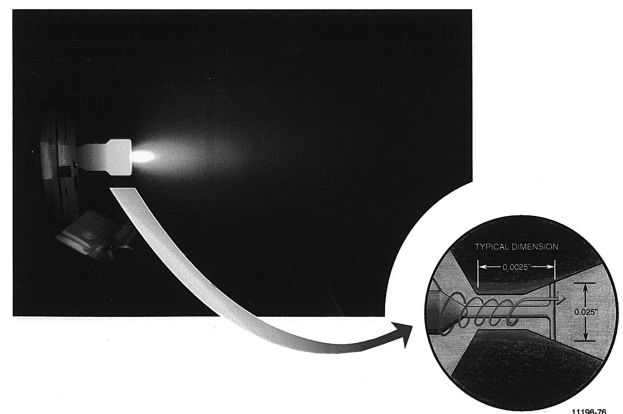
FIGURE 1 Photograph and schematic of a flight-ready hydrazine resistojet. [Courtesy of the Primex Corporation.]

of the propellant flow through the heating chamber must reach temperatures as high as 10,000°K, while being restrained from direct contact with the chamber and nozzle walls. Thus, steep radial gradients in temperature must be sustained, which renders the entire flow pattern explicitly two-dimensional. The most effective and straightforward means for achieving such profiles is by passing an electric arc directly through the chamber in some appropriate geometry. Figure 2 shows a diagram and a photograph of a prototypical thruster of this class, commonly called an electrothermal arcjet. Direct currents of tens or hundreds of amperes are passed through the gas flow between an upstream conical cathode and a downstream annular anode integral to the exhaust nozzle, generating a tightly constricted arc column that reaches temperatures of several tens of thousands of degrees on its axis. The incoming propellant is usually injected tangentially, then swirls around, along, and through this arc, expanding in the anode/nozzle to average velocities of tens of thousands of meters per second. Properly designed and operated, the

chamber and nozzle walls remain tolerably cool under the steep radial gradients, and even the arc attachment regions on the cathode and anode are somewhat protected by the electrode sheath processes, even though the cathode tip must reach incandescent temperatures to provide the requisite thermionic emission of electron current.

Analytical models of thrusters of this type usually represent the arc in three segments: a cathode fall region, which functions to heat the cathode tip and extract electrons from it; an arc column, wherein ohmic heating hboxsustains the necessary ionization against interior recombination and radiation losses; and an anode fall region, wherein the arc terminates in a diffuse annular attachment on the diverging nozzle wall, depositing thermal electron energy into the body of the thruster. Heating of the propellant actually occurs in two important modes: by direct passage of a core portion of the flow through the arc itself, and by conduction and convection to the outer flow from the chamber and nozzle walls, which themselves have been heated by radiation from the arc column and by the anode attachment. This latter, regenerative component rescues the efficiency of the thruster somewhat from the detrimental frozen flow losses associated with the failure of the hottest portion of the core flow to recover much of the energy invested in its ionization and dissociation. Aside from these frozen flow losses, the efficiency also suffers from viscous effects, nonuniform heat addition across the flow, and heat deposition in the near-electrode regions due to voltage drops in the electrode sheaths.

Arcjets on contemporary operational flights typically use catalytically decomposed hydrazine as propellant and operate at a power level of about 1.5 kW with an exhaust velocity between 5000 and 6000 m/sec and an efficiency up to 40%. While ammonia, by virtue of its lower molecular mass, can offer an exhaust velocity as high as 9000 m/sec at the same power levels, the associated complexity of the mass feeding system favors the use of

FIGURE 2 Photograph and schematic of a 1.5-kW arcjet in operation. [Courtesy of the Primex Corporation.]

hydrazine. Since these arcjets operate at a voltage of about 100 V, which is generally higher than the spacecraft bus voltage, dedicated power processing units, whose mass can exceed that of the dry propulsion system, are required.

Starting with the first of the Telstar-4 series of GEO communication satellites launched in 1993, hydrazine arcjets have quickly gained acceptance as viable propulsion options for NSSK. They represent the second evolutionary step, after resistojets, in the use of EP systems and offer substantial propellant mass savings over all previous monopropellant propulsion options. Although a recent test flight of a 30-kW ammonia arcjet (on the Air Force ESEX spacecraft) has demonstrated the potential of this higher power class of electrothermal propulsion for more thrust-intensive missions such as orbit transfers and primary propulsion maneuvers, the difficulty of providing such high power in space, combined with the lifetime-limiting problems of electrode erosion and whiskering have so far delayed such applications.

D. Inductively and Radiatively Heated Devices

The most vulnerable elements of direct current arcjets are the electrodes that transmit the high currents from the external circuit to the arc plasma, and their erosion ultimately limits the operational lifetime of these thrusters. In efforts to alleviate this basic problem, a number of more exotic concepts for electrothermal propulsion have been proposed and implemented, wherein the propellant is ionized and heated by means of some form of electrodeless discharge. These have varied widely in power levels, geometries, propellant types, and densities and have utilized applied frequencies ranging from low radio frequency (RF) to the microwave bands. In all cases, the strategy is to heat the free electron component of the ionized propellant by means of an applied oscillating electromagnetic field and then to rely either on ambipolar diffusion to direct the ions and neutrals along an appropriate exhaust channel (inductive thrusters) or on collisional and radiative heating of the neutral component by a sustained plasma upstream of the throat of a diverging nozzle (microwave thrusters). Devices of this class are thus hybrid electrothermal and electrostatic and, indeed, since some of them also employ magnetic fields to confine and direct the flow, may actually embody all three classes of interaction.

Early enthusiasm for this class of accelerators was necessarily tempered by the relatively low efficiency of RF and microwave power generation technologies of that time, which would have transcribed into intolerably massive space power supplies. More recent advances in solid-state power processing have revived some of these concepts, although none has yet been flight-tested. The most mature of these concepts is currently a microwave elec-

trothermal thruster that operates with hydrogen, nitrogen, or ammonia at exhaust velocities ranging between 4000 m/sec and 12,000 m/sec and efficiencies as high as 60%, excluding the efficiency of the microwave source. The microwave electrothermal thruster seems particularly amenable to scaling to low powers. While most of the recent development has been at the kilowatt level, scaled-down prototypes operating efficiently at 100 W and below have also been developed.

III. ELECTROSTATIC PROPULSION

A. Basic Elements

The fundamental thermal limitations on attainable exhaust speeds and lifetimes associated with the heating and expansion processes of electrothermal accelerators can be categorically circumvented if the propellant is directly accelerated by an external body force. The simplest such device, in concept, is the ion thruster, wherein a beam of atomic ions is accelerated by a suitable electric field and subsequently neutralized by an equal flux of free electrons. The essential elements of such a thruster are sketched in Fig. 3, where a collisionless stream of positive atomic ions, liberated from some source, is accelerated by an electrostatic field established between the source surface and a suitable permeable grid. Downstream of this region, electrons from another source join the ion beam to produce a stream of zero net charge, which exits the accelerator at a speed determined not only by the net potential drop between the ion source and the plane of effective neutralization, but also by the charge-to-mass ratio of the ion species employed.

A quick calculation, based on reasonable electrode dimensions, manageable applied voltages, and available ion charge-to-mass ratios, indicates that extremely high exhaust speeds, well in excess of 10^5 m/sec, are readily achievable. Indeed, given the power supply mass penalty, which scales strongly with the exhaust velocity according to Eq. (4), these devices tend to optimize their thrust efficiency at too high an exhaust velocity for most near-earth and interplanetary mission applications. A more troublesome drawback, however, is that regardless of the particular electrode configurations and propellant species employed, thrusters of this class are severely limited in their attainable thrust density by space-charge distortions of the applied electric field pattern. Specifically, it can readily be shown that the maximum ion current density that can be sustained through a one-dimensional acceleration gap d, across which is applied a voltage V, is

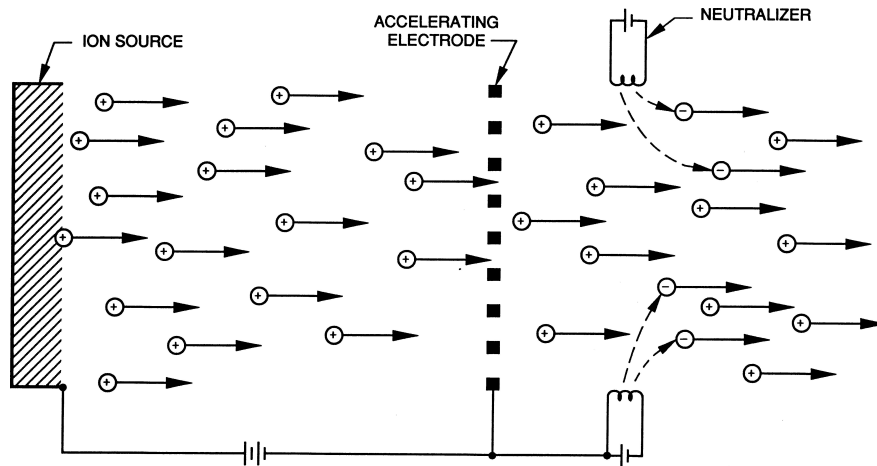$$j = \frac{4\epsilon}{9} \left(\frac{2q}{M} \right)^{1/2} \frac{V^{3/2}}{d^2}, \qquad (5)$$

FIGURE 3 Ion thruster schematic.

where ϵ is the dielectric permittivity constant and q/M is the ion charge-to-mass ratio, all in mks units. It then follows that the maximum thrust density of the emitted beam depends only on V/d:

$$\frac{T}{A} = \frac{\dot{m}v_e}{A} = \frac{jMv_e}{q} = \frac{8\epsilon}{9}\left(\frac{V}{d}\right)^2, \qquad (6)$$

where A is the area of the exhaust jet, while the corresponding exhaust speed depends only on q/M and V:

$$v_e = \left(\frac{2qV}{M}\right)^{1/2}. \qquad (7)$$

The thrust density and thrust power density that can be conveyed on the exhaust beam for attainable values of V, d, and q/M thus compute to rather small values, on the order of a few newtons per square meter and 10^5 W per square meter, respectively, at best. On the positive side, the attainable thrust efficiency is essentially limited only by the energetic cost of preparing the individual ions, which should be a small fraction of their exhaust kinetic energy. System optimization, therefore, involves a somewhat complex multidimensional trade-off among the exhaust speed, thrust density, efficiency, and power system specific mass, for any given mission application.

B. Ion Thruster Technology

1. Ion Sources

In practice, the most amenable propellants for electrostatic thrusters have proven to be cesium, mercury, argon, krypton, and most commonly xenon, and many possible sources of such ions of the requisite efficiency, reliability, and uniformity have been conceived and developed. Of these, only three, the electron bombardment discharge source, the cesium–tungsten surface contact ionization source, and one form of RF discharge source, have survived to application.

The essential elements of the bombardment sources are some form of cylindrical discharge chamber containing a centerline cathode that emits electrons, a surrounding anode shell, and a permeating azimuthal and radial magnetic field that constrains the electrons to gyrate within the chamber long enough to ionize the injected propellant gas and to direct it, once ionized, to extractor and accelerator grids downstream. One contemporary implementation of such a chamber is shown in Fig. 4. This particular device employs a hollow cathode electron source, wherein is sustained a secondary discharge that facilitates electron emission from the interior walls of the cathode cavity. The magnetic field permeating the entire chamber is provided by three ring magnets, empirically configured to establish a grossly diverging but doubly cusped field pattern that optimizes the discharge for ionization and ion extraction purposes. The magnitude of this field is adjusted in concert with the anode–cathode voltage differential to maximize the ionization efficiency and discharge stability while minimizing the production of doubly charged ions, which would be out of focus in the accelerator gap and thus tend to erode the grids through high-energy sputtering. Typical values for xenon and mercury propellants would be in the regimes of 0. 25 T and 30 V, respectively. Slightly different chamber configurations and field values have also been used successfully.

Contact ion sources rely on the difference between the electronic work function of a metallic surface and the ionization potential of alkali vapors to ionize the latter on contact with the former. Very few metal–alkali combinations have this requisite positive voltage differential, and of these the combination of tungsten and cesium provides
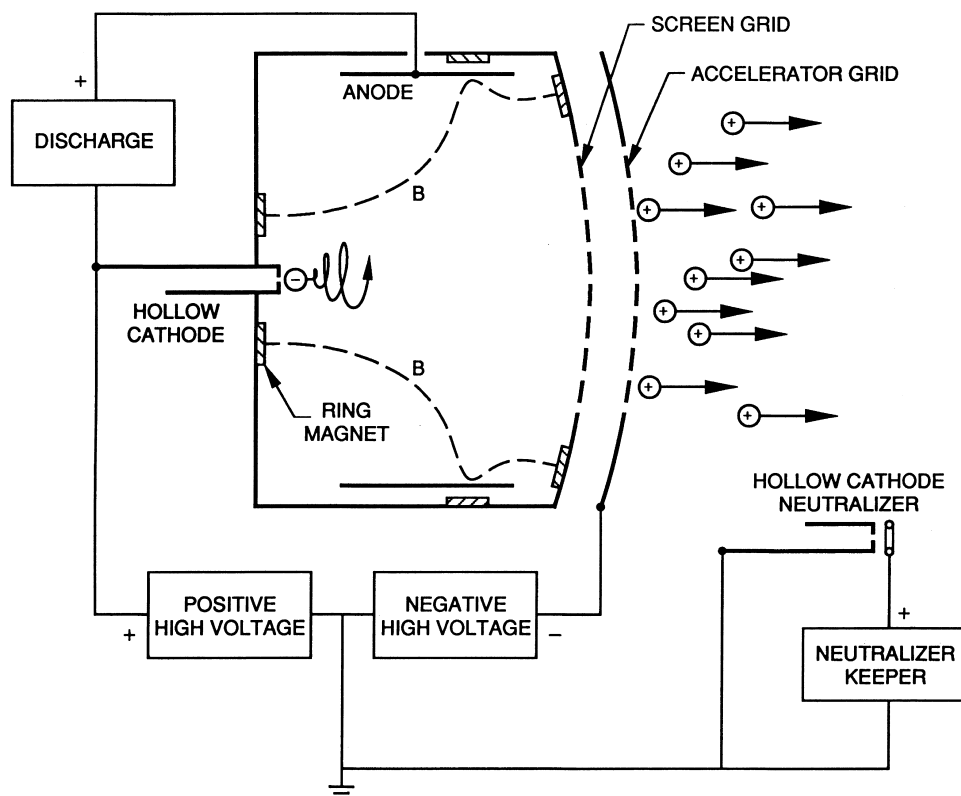
FIGURE 4 Ring-cup ion thruster. [Courtesy of Colorado State University and NASA Glenn Research Center.]

the largest. The most common implementation has been to force hot cesium vapor through a porous tungsten wafer to enhance surface contact, but problems in degradation of wafer porosity and recondensation of cesium vapor have tended to compromise the ionization efficiency and lifetime of these sources.

The RF ionization sources currently favored in Western Europe are similar in principle and configuration to the U.S. electron bombardment sources, except that the discharge is inductively driven RF rather than directly coupled dc. While the efficiency and lifetime of these RF sources seem competitive, they entail the complication of RF modules in their power processing equipment. In Japan another cathodeless ion thruster concept has been developed that uses a microwave source to create and sustain the plasma through electron cyclotron resonance (ECR) and offers some system and lifetime advantages.

2. Accelerator Grids

In virtually all classes of ion thruster, the positive ions are extracted from the source and accelerated downstream by a system of grids configured to achieve the desired exhaust velocity with minimum beam impingement. In U.S. bombardment engines, for example, a double grid configuration is usually dished downstream as shown in Fig. 4

to improve its mechanical and thermal stability against distortion. The upstream grid is maintained at a higher positive potential than required by the desired exhaust speed in order to enhance the ion extraction process and increase the space-charge limited current density that can be sustained. The downstream grid then reduces the exhaust plane potential to the desired value. This "accel–decel" scheme has the advantages of higher beam density at a given net voltage and of reducing electron backstreaming from the neutralized beam downstream.

The grid perforations are configured analytically and empirically to focus the ion stream into an array of beamlets that pass through with minimum impingement. In this process, the downstream surface of the discharge plasma in the chamber acts as a third electrode, and since this contour is not independent of the discharge characteristics and applied grid voltages, it can be a source of some instability. Further complications are introduced by the small fractions of double ions or neutrals that find their way into the beam and are henceforth out of focus and free to bombard the grid surfaces.

3. Neutralizers

If the ion beam emerging from the downstream electrode is not to stall on its own interior potential profile, it must

be electrostatically neutralized within a very few units of grid spacing. This is typically achieved by provision of a flux of electrons, usually from another hollow cathode discharge, which fortuitously mix effectively within the ion beam by means of a variety of microscopic and macroscopic internal scattering processes. Once so neutralized, this plasma constitutes a downstream "virtual electrode" that completes the axial potential pattern.

4. System Aspects and Application History

Although they are technically the most complex EP systems, ion engines like those outlined above and shown in Fig. 5 have been the most thoroughly developed and tested of all EP devices. Their appeal stems primarily from their maturity, demonstrated long lifetime ($>20,000$ hr), relatively low beam divergence (<20 deg) and high efficiency (65%) at a useful exhaust velocity (30,000 m/sec), and power levels between 200 and 4000 W. These advantages are somewhat offset by low thrust density, system complexity, and high-voltage requirements which translate into power processing unit (PPU) specific masses as high as 10 kg/kW. Their space-worthiness has been demonstrated by more than a dozen U.S. and Soviet flight tests, starting in 1962, which have provided guidance and validation to ground-based research and development efforts that have led to the optimization of their designs and materials.

The first operational use of ion thrusters occurred in 1994 on the Japanese ETS-6 and COMETS satellites, for which four 12-cm ion engines provided NSSK propulsion. This was followed in 1997 by PAS-5, which inaugurated the first U.S. commercial satellite bus to rely on ion propulsion for GEO station keeping.
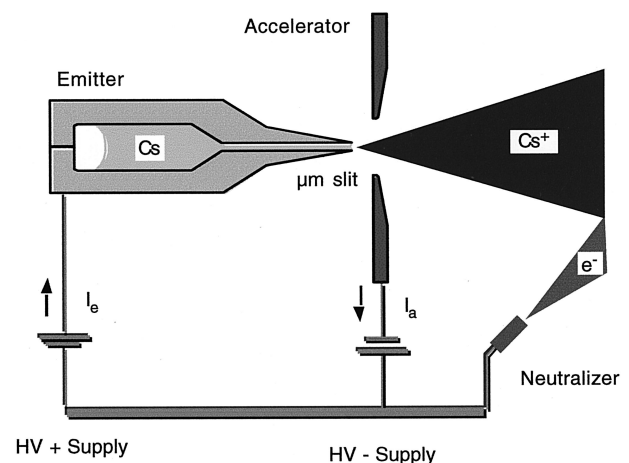
While ion engines compete well with other EP options for near-earth applications, their ability to operate efficiently and reliably at even higher exhaust velocities makes them ideally suited for energetic (i.e., high Δv) deep-space missions, where long thrusting times can be tolerated. In 1998, NASA 's Deep Space 1 became the first interplanetary mission to benefit from ion propulsion. On its way to its encounter with asteroid Braille, the spacecraft used a xenon bombardment ion propulsion system to provide the required Δv over 1800 hr of thrust, while consuming only 12 kg of propellant and demonstrating in-space performance within 1% of that measured in ground tests. Various commercial and scientific missions using ion propulsion are slated for launch in the next few years, including the world's first sample-and-return attempt from an asteroid by the Japanese MUSES-C spacecraft.

C. Other Electrostatic Propulsion Concepts

Many of the complexities of ion bombardment sources, multibeam focusing grids, electromagnets, and other subsystems of ion thrusters can be bypassed altogether if only minute thrust levels are needed. Creating a high electric field concentration at the lips of a capillary slit, as shown in Fig. 6, allows direct ionization from the liquid phase of a metal to be achieved by field emission, and the resulting ion beam can be accelerated electrostatically to very high velocities. Field emission electric propulsion (FEEP) devices of this kind have evolved in Europe since the late 1970s and have unique characteristics and advantages. In a typical FEEP device, cesium propellant from a small reservoir is allowed to wet the inside of a 1-μm capillary channel and form a free surface between the blade-edge lips of the emitter. An electric field of a few kilovolts is

FIGURE 5 A 30-cm bombardment ion engine. [Courtesy of Hughes Aircraft Co.]

FIGURE 6 Schematic of a field emission electric propulsion (FEEP) device.

applied between the emitter and an accelerator electrode, and the resulting electric field concentrations at the slit edges form protruding cusps, or "Taylor cones," on the edge of the liquid by means of the competition between the electrostatic forces and surface tension. When the electric field reaches field emission levels (10^9 V/m), these points become local ion emission sites. The extracted and accelerated ion beams are subsequently neutralized by injecting electrons from an appropriate source. For a typical extraction voltage of 10 kV, the ion exhaust velocity is in excess of 100,000 m/sec, the efficiency close to 100%, and the thrust-to-power ratio about 16 μN/W. Although FEEP thrusters with thrust levels as high as 5 mN have been developed by the European Space Agency, near-term applications are for missions requiring small and precise thrust. While no such devices have yet flown, a number of missions are planned in the United States and Europe, including systems associated with space-borne interferometers for detection of gravitational waves, and missions requiring fine pointing and formation flying of micro-spacecraft. Since these FEEP devices are operated with cesium because of its high atomic mass, low ionization potential, low melting point (28.4°C), and good wetting capabilities, a number of practical problems related to spacecraft plume interactions and propellant contamination will need to be resolved.

Another simple electrostatic thruster concept that has the advantages over FEEP of higher thrust-to-power ratios and the use of more benign propellants is the colloidal thruster. It employs similar physical processes, except that nonmetallic liquids are used and sub-micron-sized charged particles (colloids) are produced and accelerated. This yields specific impulses more compatible with near-earth missions. Much of the work on colloidal thrusters was carried out in the 1960s and identified limitations on the achievable charge-to-mass ratios and the uniformity of the the charge-to-mass distributions. The former transcribes to excessively large voltages (hundreds of kilovolts) to attain the desirable exhaust velocities (10,000 m/sec) and the latter results in large beam divergences. Nonetheless, more recent research in Russia and in the United States, driven by the advent of micro-spacecraft missions, has returned colloidal electrostatic thrusters to the arsenal of electric micropropulsion options.

IV. ELECTROMAGNETIC PROPULSION

A. Basic Concept

The third category of EP relies on the interaction of an electric current pattern driven through a conducting propellant stream with a magnetic field permeating the same
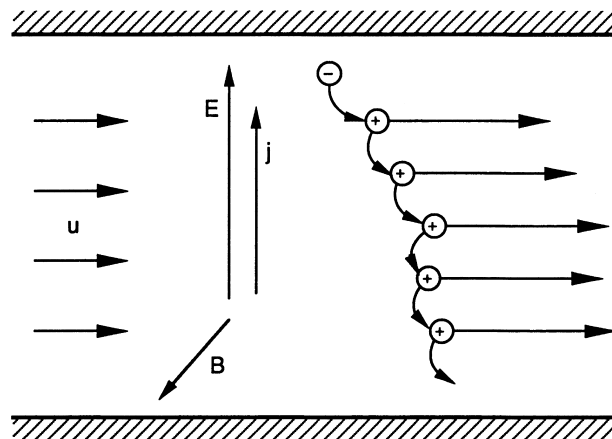

FIGURE 7 Crossed-field electromagnetic thruster schematic.

region to provide the accelerating body force. Such systems can produce exhaust speeds considerably higher than those of the electrothermal devices, and thrust densities much larger than those of the electrostatic thrusters, but are phenomenologically more complex and analytically less tractable than either of these alternatives. The essence of an electromagnetic thruster is sketched in Fig. 7, where some electrically conducting fluid, usually a highly ionized gas, is subjected to an electric field E and a magnetic field B, perpendicular to each other and to the fluid velocity u. The current density j driven by the electric field interacts with B to provide a streamwise body force $f = j \times B$ that accelerates the fluid along the channel. The process may alternatively be represented from a particulate point of view in terms of the mean trajectories of the current-carrying electrons, which, in attempting to follow the electric field, are turned downstream by the magnetic field, transmitting their streamwise momentum to the heavy particles in the stream by collisions and/or by microscopic polarization fields. It is important to note that in either representation, the working fluid, although highly ionized, is macroscopically neutral, hence not constrained in its mass flow density by space-charge limitations as in the electrostatic accelerators.

B. Varieties

Unlike the electrothermal or electrostatic classes, which offer only a few practical configurations, electromagnetic acceleration presents myriad possibilities for implementation. The applied fields and internal currents may be steady, pulsed, or alternating over a broad range of frequencies; the B fields may be externally applied or induced by the current patterns; and a broad variety of propellant types, including liquids and solids, may be employed, along with a host of channel geometries; electrode and insulator configurations; means of injecting, ionizing, and

ejecting the propellants; and modes of delivery of the requisite electrical power. As a result of ongoing research and development since the late 1950s, a huge number of such possible permutations of the electromagnetic propulsion concept have been studied, both experimentally and theoretically, but only a few have survived the gauntlet of requisite efficiency, reliability, range of performance, and system compatibility to retain enduring technological promise. Of these, most advanced are the steady or quasi-steady magnetoplasmadynamic (MPD) thrusters, the Hall-current accelerators, and the pulsed plasma devices.

C. Magnetoplasmadynamic Thrusters

1. Operating Principles

As sketched in Fig. 8, the magnetoplasmadynamic thruster (MPDT) is characterized by a coaxial geometry constituted by a central cathode, an annular anode, and some form of interelectrode insulator. Gaseous propellants are introduced into the upstream portion of the channel, whereafter they are ionized by passage through an intense, azimuthally uniform electric arc standing in the interelectrode gap. If the arc current is high enough, its associated azimuthal magnetic field is sufficient to exert the desired axial and radial body forces on the propellant flow, directly accelerating it downstream and compressing it toward the centerline into an extremely hot plasma just beyond the cathode tip. Subsequent expansion of this plasma, along with the direct axial acceleration, yields the requisite exhaust velocity.

Theoretically, these self-field accelerators can be represented in relatively simple continuum plasmadynamic form or in more elaborate three-fluid plasma kinetic formulations. Irrespective of their interior details, electromagnetic tensor analysis yields a generic thrust relation:

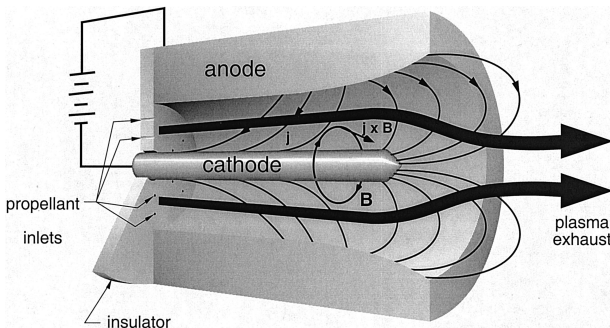$$T = \frac{\mu J^2}{4\pi} \left(\ln \frac{r_a}{r_c} + A \right), \qquad (8)$$

FIGURE 8 Magnetoplasmadynamic thruster (MPDT) schematic.

where T is the total thrust, μ the vacuum magnetic permeability, J the total arc current, r_a and r_c the effective arc attachment radii on the anode and cathode, and A a parameter slightly less than unity that depends on the finer details of the current attachment patterns on the electrodes. Note that this relation, which generally agrees with experiments, is independent of the mass flow rate and any other properties of the propellant, thus the exhaust velocity must scale as the ratio $J^2/\dot{m}$. An important nondimensional scaling parameter, ξ, which arises in several other empirical and theoretical contexts, including the onset of severe erosion and the appearance of various modes of plasma waves and instabilities, is calculated from the following formula:

$$\xi \equiv \left[\frac{\frac{J^2 \mu}{\dot{m} 4\pi} \left(\ln \frac{r_a}{r_c} + A \right)}{(2\phi_i/M)^{1/2}} \right]^{1/2}, \qquad (9)$$

where the propellant is represented by its ionization potential ϕ_i and atomic mass M. Nominal MPDT operation is achieved at $\xi \simeq 1$, while stable and low-erosion operation is typically limited to $\xi < 2$. The lifetime of such devices is limited by component erosion, most notably evaporation of cathode material, and the performance is bounded by losses associated with ionization and thermal energy frozen in the flow as well as losses in the electrode sheaths. The modeling and control of many loss mechanisms are complicated by the presence of plasma turbulence due to current-driven microinstabilities which can cause excessive ionization and heating.
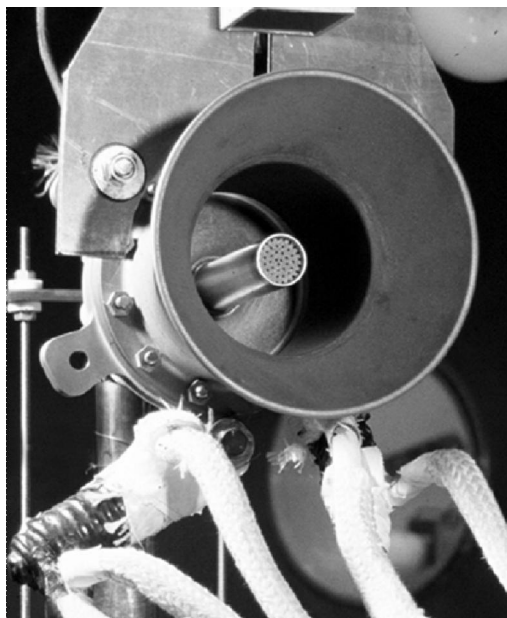
In space applications, the optimum power range of operation will be delimited on the low side by the need to ionize the propellant fully and to keep the electrode losses relatively small compared with the thrust power. The upper power limit will be set either by tolerable erosion rates and plasma instabilities, or by the realities of the overall system, including the available space power source and associated heat rejection equipment. At megawatt power levels and corresponding propellant flow rates, both ground testing and space testing of these highly promising devices present formidable technological and economic problems. Indeed, at present there are no U.S. facilities capable of long-term megawatt operation of steady-state MPDTs.

2. Present and Projected Capabilities

The MPDT has demonstrated its capability of providing specific impulses in the range of 1500–8000 sec with thrust efficiencies exceeding 40%. High efficiency (above 30%) is typically reached only at high power levels (above 100 kW); consequently, the steady-state version of the MPDT is regarded as a high-power propulsion option.

When the thruster is operated below 200 kW, the self-induced magnetic field becomes only marginally sufficient to provide the desired body force, and external fields are frequently added to enhance performance in this range. However, in its megawatt versions, the self-field MPDT has the unique capability, among all developed electric thrusters, of processing very high power levels in a simple, compact, and robust device that can produce thrust densities as high as 10^5 N/m^2. These features have rendered the steady-state MPDT particularly attractive for energetic deep-space missions requiring high thrust levels, such as piloted and cargo missions to Mars and the outer planets, as well as for nearer-term orbit raising missions.

In addition to the present unavailability of high power in space, the cathode erosion rates of the steady-state MPDT (which can be as high as 0. 2 μg/C), have slowed the evolution of steady-state MPDTs toward flight applications. A version of the steady-state MPDT, called the lithium Lorentz-force accelerator (Li-LFA), shown in Fig. 9, uses a multichannel hollow cathode and lithium propellant to substantially reduce the cathode erosion problem while significantly raising the thrust efficiency at moderately high power levels. For example, a 200-kW Li-LFA has demonstrated essentially erosion-free operation over 500 hr of steady thrusting at 12.5 N, 4000 sec I_s, and 48% efficiency. Since no other electric thruster has yet shown such a high power processing capability, the Li-LFA is at the forefront of propulsion options for nuclear-powered deep-space exploration and heavy cargo missions to the outer planets.

FIGURE 9 A 100-kW-class, lithium MPDT or Lorentz-force accelerator.

In order to benefit from the advantages of MPD propulsion on today's power-limited spacecraft, the MPDT can also be operated in a quasi-steady (QS) pulsed mode using flat-top high-current pulses long enough ($>350\,\mu$sec) for a steady-state current pattern to dominate the acceleration process. The QS-MPDT can thus benefit from the high efficiency associated with the instantaneous high power, while drawing low steady-state power from the spacecraft bus. This approach was adopted in the first MPDT to fly as a propulsion system, a 1-kW-class QS-MPDT that operated successfully in 1996 onboard the Japanese Space Flyer Unit. (Previous MPDT spaceflight tests by Japan and the Russia in 1975, 1977, 1980, and 1983 were purely experimental.)

While no present operational spacecraft employ MPD propulsion systems, ongoing research and development activities in Russia and the United States on the Li-LFA, and in Europe and Japan on the gas-fed MPDT, aim at further improving the performance and lifetime of the steady-state MPDT to a level that meets near-future advanced propulsion needs.

D. Hall Thrusters

1. Operating Principles

If any electromagnetic accelerator is operated at low enough plasma density or high enough magnetic field, the current driven through it will divert from strict alignment with the applied electric field to acquire a component in the $\boldsymbol{E} \times \boldsymbol{B}$ direction—a form of the well-known "Hall effect" that derives from the ability of the current-carrying electrons to execute significant portions of their cycloidal motions in the crossed fields before transferring their momentum to the heavy particles. In extreme cases, this Hall-current component can totally dominate the conduction or "Lorentz" component.

Low-density Hall-current accelerators exploit this effect by providing channel and field geometries that lock the plasma electrons into a nearly collisionless cross-stream drift, which leaves the positive ions free to accelerate downstream under a component of the applied electric field. In a sense, such devices are hybrid electrostatic–electromagnetic accelerators with space-charge neutralization automatically provided by the background of drifting electrons. Because the magnetic fields in these devices are externally supplied, and because the mass flow densities are intrinsically low, these thrusters optimize their performance at considerably lower powers than those of the self-field MPD devices.

2. Evolution and Present Capabilities

Coaxial Hall plasma accelerators were optimized in the Former Russia during the late 1960s to the late 1990s,
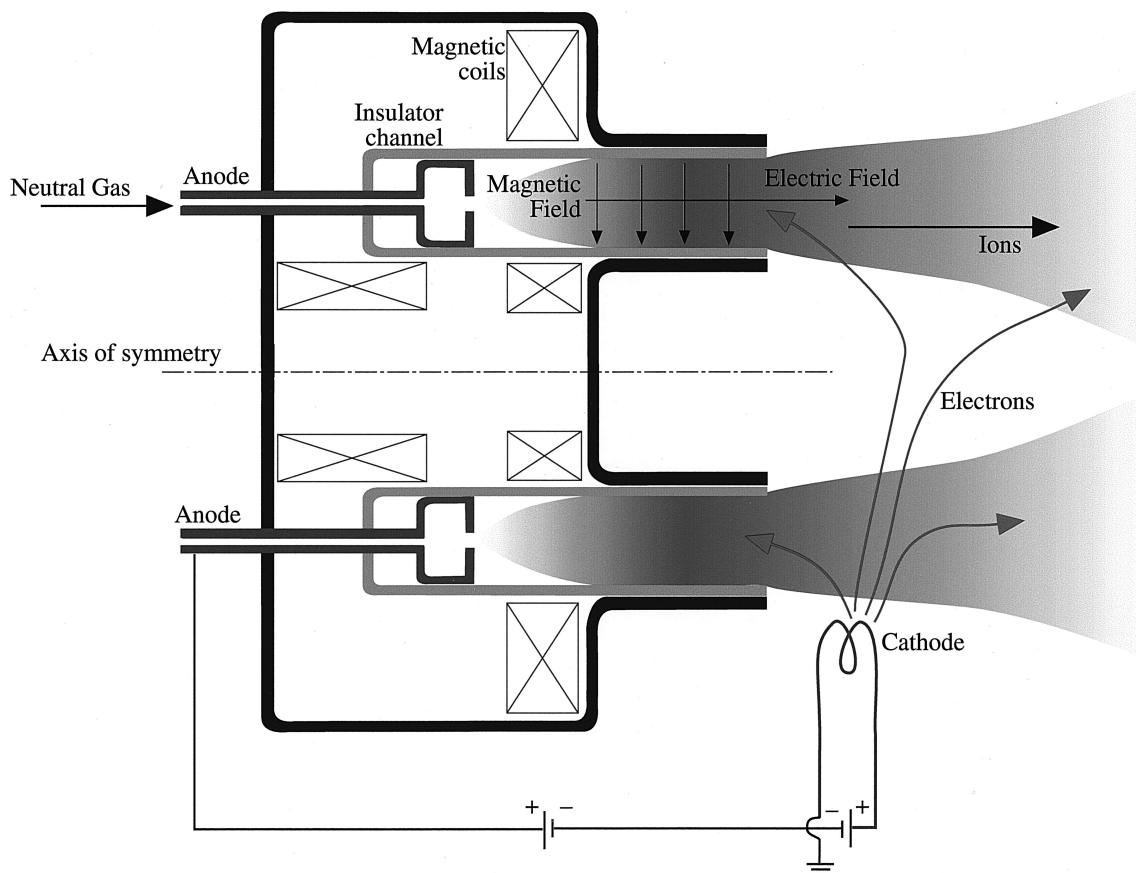
where they attained efficiencies above 50%. Some of the original work on Hall thrusters was also conducted in the United States in the early and mid-1960s, but interest in that accelerator waned in favor of more extensive ion thruster development, until a vigorous revival of interest began in the United States, Europe, and Japan in 1991.

Today's Hall thrusters are sometimes referred to as "closed-electron-drift" devices, given the azimuthal drift of electrons that is common to all present variants of such thrusters. The most common versions are the stationary plasma thruster (SPT) (also termed the "magnetic layer thruster") and the anode layer thruster (ALT). The former differs from the latter by its extended channel, the use of insulator chamber walls, and the extent of the quasi-neutral acceleration region, but both rely on the same basic principles for ionizing and accelerating the propellant. A schematic of a Hall thruster of the SPT type is shown in Fig. 10. Electrons from the cathode enter the chamber and are subjected to an azimuthal drift in the crossed radial magnetic and axial electric fields, wherein they undergo ionizing collisions with the neutral propellant atoms (typically xenon) injected through the anode. While the mag-netic field strength is sufficient to lock the electrons in an azimuthal drift, it does not significantly affect the trajectory of the ions, which are directly accelerated by the axial electric field. An axial electron flux equal to that of the ions reaches the anode due to a cross field mobility that often exceeds classical values, and the same flux of electrons is available from the cathode to neutralize the exhausted ions. Quasi-neutrality is thus maintained throughout the chamber and exhaust beam, and consequently no space-charge limitation is imposed on the acceleration, which allows relatively high thrust densities compared with those of conventional electrostatic propulsion devices. Nominal operating conditions of a common flight module (e.g., the Russian SPT-100) operating with xenon are a 2- to 5-mg/sec mass flow rate; a 200- to 300-V applied voltage, yielding a plasma exhaust velocity of 16,000 m/sec; and a thrust of 40–80 mN, at efficiencies of about 50%.

3. Applications and Flight History

Hall accelerators of the closed-drift type are at present the most commonly used plasma thrusters. Since 1972,

FIGURE 10 Schematic of a Hall thruster with an extended insulator channel (stationary plasma thruster, or SPT), showing the external cathode, the internal anode, the radial magnetic field, and typical particle trajectories.

more than 110 Hall thrusters have been flown on Russian spacecraft, and more than 52 thrusters remain in operation. They have also been used as plasma sources in active space experiments. In view of their high specific impulse, relatively high efficiency, and high thrust density, Hall thrusters continue to be developed by industry and government for purposes of orbit insertion, attitude control, and drag compensation. Because of the stringent requirements for trouble-free operation for many thousands of hours, efforts to improve performance and lifetime are currently under way in the United States, Russia, Europe, and Japan. These include efforts to lower beam divergence (which is typically between 30 and 40 degrees), to reduce electromagnetic interference (due to various types of plasma oscillations), to lower erosion rates, and to increase thrust efficiencies. Strides toward more efficient, compact, and lightweight PPUs (with specific masses as low as 5 kg/kW) are also being made.

E. Pulsed Plasma Thrusters

The power conservation strategy underlying quasi-steady operation of MPDTs can be carried further by employing small power systems to drive plasma thrusters in short ($\simeq$10-μsec) bursts of high instantaneous power ($\simeq$10 MW). The energy involved is typically stored in capacitor banks or inductor coils, then delivered rapidly to the electrodes by some form of high-speed switch.

When a gas-fed pulsed plasma thruster (GF-PPT) is operated in a predominantly electromagnetic mode, the acceleration is achieved by "snowplow" action of a current sheet driven by its self-induced Lorentz body force. Various geometries that had been used in early thermonuclear fusion experiments, including coaxial guns and linear and theta pinch discharges (Fig. 11), have been modified into propulsion configurations.

In the early and middle 1960s, GF-PPT systems were developed that had efficiencies above 20% at specific impulses near 5000 sec, using some 65 J of stored energy per pulse, but since the main focus of EP research at that time was on developing *primary* propulsion systems, GF-PPT research and development ebbed by the end of that decade due the need for massive capacitors and the energy and lifetime requirements of the fast-acting valves required to ensure high mass utilization efficiency.

However, one derivative of this concept survived in the form of the simple ablative pulsed plasma thruster (APPT), which promised to solve the mass utilization problem without the use of valves, and to save system mass by shedding the complex mass storage and control systems of their gas-fed counterparts. In the APPT, shown schematically in Fig. 12, the surface of a polymer block (most commonly
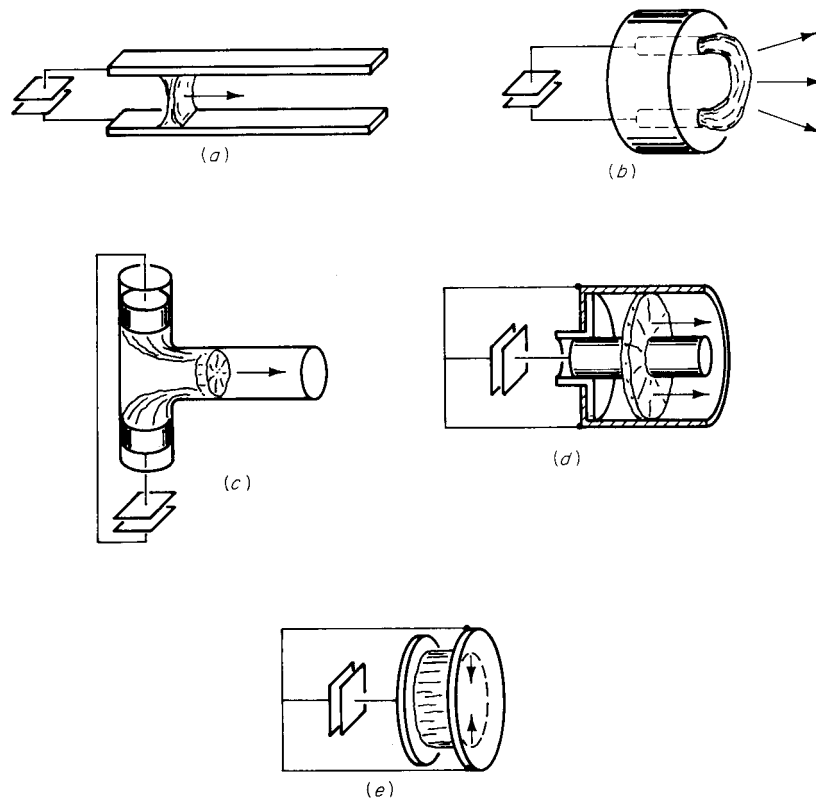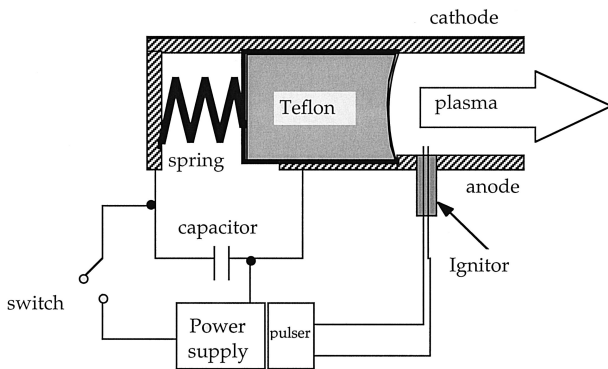

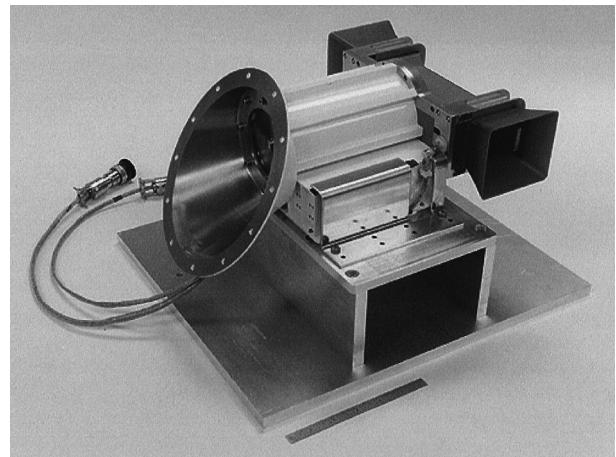FIGURE 11 Various pulsed plasma thruster (PPT) configurations.

FIGURE 12 Schematic of an ablative pulsed plasma thruster (APPT).

FIGURE 13 A flight-ready Teflon ablative pulsed plasma thruster (APPT) module using two thrusters, positioned on the opposite ends of the thrust axis. [Courtesy of the Primex Corporation.]

Teflon) is successively eroded by intermittent arc pulses driven across its exposed face, and the ablated material is accelerated by a combination of thermal expansion and self-field electromagnetic forces.

The APPT has the claim of being the first EP system to operate in orbit, when the 1964 Soviet Zond-2 spacecraft used six Teflon APPTs for sun pointing control. The United States followed in 1968 with the LES-6 satellite, which used four APPTs for east–west station keeping (EWSK). Since then, APPTs have had a sporadic history of application and, except for a small number of experimental suborbital and orbital tests by the United States and China, they have been used on only a series of five U.S. Navy satellites launched in the late 1970s and early 1980s. It was not until the mid-1990s, in the context of power-limited small satellites, that APPT research and development were rekindled. Improved capacitor technology, combined with the simplicity of the APPT and its propellant storage and feed system, and its capability of providing small and precise impulses at high specific impulse and arbitrarily low spacecraft power, made it suitable for many attitude-control chores on power-limited small satellites. A small but growing number of upcoming U.S. missions using APPTs have been planned, which will use flight-ready modules such as that shown in Fig. 13.

Two of the most severe defficiencies of APPTs, namely their low efficiency ($<15\%$) at low pulse energies and spacecraft contamination by the polymer products in the plume, have spurred some revival of their gas-fed progenitors. GF-PPTs have the advantages of compatibility with a wide range of propellants, cleaner exhaust, and a wider scalability of performance. Recent advances in low-inductance, high-frequency, and high-current pulsing technologies have relieved the low mass utilization efficiency problem that plagued the 1960s prototypes. Further improvements of these devices will depend on a better understanding of the nature and scaling of the complex dis-

sipative mechanisms in such unsteady flows, and on the the ability to control and abate the canting and instability of the accelerating current sheets.

F. Inductive Thrusters

As with the electrothermal arcjets, the most vulnerable elements of all the electromagnetic thrusters described above are the electrode surfaces. Hence there has been some on-going interest in a variety of inductive possibilities that require no electrodes directly exposed to the intense discharge environment. Again, most of these concepts have been transposed from other technologies, including pulsed inductive discharges, traveling wave accelerators of various classes, RF fringe-field accelerators, and cyclotron resonance devices. All involve inherently unsteady flows over a wide range of operating frequencies, and all must trade off the absence of electrode erosion against generally poorer coupling efficiency between the external circuitry and the accelerating propellant plasma. Like the inductive electrothermal machines, their overall systems suffer from the more complex and massive power processing equipment needed to drive them, although some of this disadvantage has been ameliorated by recent improvements in solid-state electronic technology, so that their future may be somewhat brighter than their past.
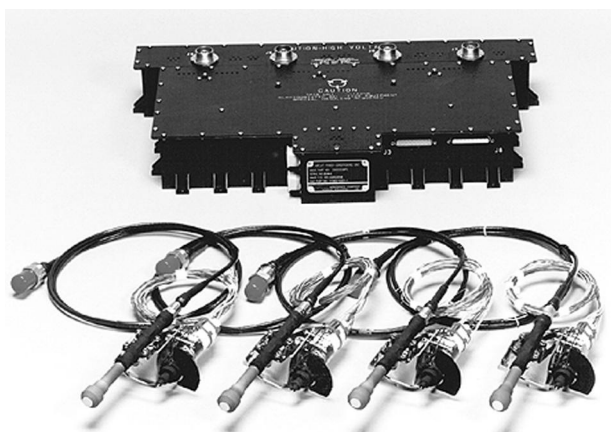
V. SYSTEMS CONSIDERATIONS

A. Power Conditioning

Almost all electric thrusters, except resistojets, operate at voltages larger than those provided by the standard

bus of solar-powered spacecraft. This necessitates the use of power processing subsystems that transpose the primary power from the space-borne source to the requisite voltages, currents, and duty cycles of the given thrusters. The PPU can have much influence over the overall efficiency, reliability, and mass of the total propulsion system. Aside from voltage conditioning, each class of thruster presents its own demands on the PPU. Many electrothermal arcs, for example, can develop negative slopes in portions of their voltage–current characteristics that are tantamount to negative impedance for the power source and must be suitably ballasted if the arc is to be properly controlled. Ion thrusters require a broad range of electrode voltages and currents for their discharge anodes, hollow cathodes, and accelerating grids and must be protected against high-voltage shorts and insulator breakdowns. High-power MPD accelerators require huge currents at relatively low voltages, and their unsteady versions present an additional overlay of energy storage and processing requirements. Almost all electric thrusters require some ignition mechanism that inevitably complicates the power package.

Only the most mature thruster concepts have had significant flight-qualified PPU development. For a hydrazine arcjet, whose typical operating voltage peaks at about 100 V, the PPU is about 91% efficient and has a specific mass of about 2.5 kg/kW. A photograph of four flight-ready arcjets and their PPU is shown in Fig. 14. The PPU penalty becomes worse for higher-voltage devices such as the xenon Hall thruster and the xenon-ion engine, whose PPUs have efficiencies of 93% and 88%, respectively, and specific masses as high as 10 kg/kW. Almost all electric thruster PPUs developed to date are for use with solar panel power sources and would need to be reconsidered when nuclear and other higher-power sources are deployed.

FIGURE 14 Four flight-ready arcjets with their power processing unit (PPU). [Courtesy of the Primex Corporation.]

B. Primary Power Sources

All this power conditioning technology must relate to a primary source that is itself reliable, compatible, and of sufficiently low specific mass to function in the given space application. Although various nuclear, chemical, and solar thermal conversion cycles have been studied as potential spacecraft power sources, practically all present satellites rely exclusively on solar panels and batteries for power sources. Consequently, progress in photovoltaic cell technology is critical to the continued growth of EP applications on near-earth spacecraft. Present standard silicon solar cells cost about \$1500/W and have a power density of about 140 W/m^2, which corresponds to a specific power of 40 W/array kg. The most likely near-term improvement on these are gallium arsenide (GaAs) cells, which have been proven to yield 220 W/m^2. When such solar cells are augmented with aluminized Mylar concentrators, they promise specific powers of 100 W/array kg, which should greatly enhance the proliferation of EP systems. Another critical improvement in solar power technology is the decrease of solar cell degradation from accumulated environmental radiation dose. This is particularly crucial for the EP missions (such as orbit raising) which require substantially longer transfer times than those of the impulsive maneuvers effected by high-thrust chemical rockets.

While for many near-earth missions of relatively modest Δv solar or chemical energy sources may be viable, for the interplanetary, megawatt scale of operations, nuclear power systems of major dimensions will inevitably be required, which will further complicate the total system with attendant environmental and safety hazards.

VI. APPLICATIONS

Although the primary motivation for development of space-worthy EP systems is the conservation of propellant mass for missions of large characteristic velocity increments, electric thrusters offer a number of attractive secondary operational benefits, including precision and variability of thrust levels and impulse increments, generous shutdown and restart capabilities, and the use of chemically passive propellants. Their major limitations are the need for sophisticated external power sources, very low to modest thrust density capabilities, and little empirical experience with unattended operation in the space environment. All these characteristics serve to circumscribe the classes of missions for which EP may reasonably be considered.

Obviously, the limited thrust densities predicate thrust-to-mass ratios for electric systems that are not propitious for rapid maneuvers in strong gravitational fields. There are no launch or ascent–descent capabilities near

planetary surfaces, and even outer orbit transfer exercises can be performed only very slowly over gentle spiral trajectories. Thus, near-planet applications will be limited to those attitude-control, station-keeping, drag-reduction, and modest orbit-changing functions (such as orbit phase changes in LEO constellations) where the minuteness and precision of thrust, propellant conservation, and long lifetime give them superiority over chemical options. Many such applications have been recognized and evaluated, and in several cases appropriate electric systems have been, or soon will be, deployed.

In the domain of interplanetary flight, however, EP offers much more substantial advantages over chemical systems, which extend in several important cases to enabling missions that simply could not be performed by means of any other reasonably projected propulsion technology. These include heavy cargo and/or piloted missions to Mars and the outer planets and many unpiloted probes beyond the solar system and out of the ecliptic plane.

Comparison of electric and chemical systems for any ambitious mission, piloted or unpiloted, reveals that the basic dynamical distinction is between essentially impulsive thrust increments provided by the latter and much more protracted, lower-level thrust profiles necessarily, and in some cases advantageously, provided by the former. Analytical optimization of extended variable thrust trajectories for interplanetary transportation is a complex task, far from fully rendered in contemporary mission analyses. Superimposed on the strictly dynamical aspects are a host of systemic considerations, such as the importance of flight time to crew, other payload considerations, internal and external environmental hazards, utility of the power supply at destination, fraction of payload to be returned, secondary mission goals in flight, human and structural compatibility with flight maneuvers such as aerobraking and swingby, and in-flight adjustment, service, and emergency return capabilities. Since many of these, especially those involving human factors, currently have inadequate fundamental databases and theoretical representations, the composite mission assessments are shaky, at best, and much more sophisticated analyses will be required before definitive mission projections.

Finally, as already mentioned, before any such ambitious EP missions can seriously be contemplated, nonsolar alternatives for high-power sources in space must be developed. For mostly political reasons, plans for deployment of nuclear high-power sources in space have so far failed to materialize, and consequently the use of electric thrusters for primary propulsion in energetic missions has had a cyclical history of false starts and disappointments. Indeed, the recent vigorous rejuvenation of the field of electric propulsion can be attributed, at least in part, to a conscious shift in emphasis away from the high-power missions envisaged during the 1960s and 1970s toward the less ambitious but more realistic power-limited small satellites of today. Now that many EP systems have entered the mainstream of astronautic technology, their role in helping to expand human ambition beyond the inner part of the solar system, although still dependent on the hitherto unrealized development of high-power sources, is perhaps on more credible ground.

SEE ALSO THE FOLLOWING ARTICLES

LIQUID ROCKET PROPELLANTS ● RAMJETS AND SCRAMJETS ● ROCKET MOTORS, HYBRID ● ROCKET MOTORS, LIQUID ● ROCKET MOTORS, SOLID ● SOLID PROPELLANTS ● SPACECRAFT CHEMICAL PROPULSION ● SPACECRAFT DYNAMICS

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[1] A special issue of the *Journal of Propulsion and Power*, Vol. 14, No. 6, 1998, contains review articles on general EP, EP for solar system exploration, ion thruster development, PPTs, Hall thrusters, MPDTs, high-power arcjets, and field emission micropropulsion.