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Ion Thruster Development Trends and Status in the United States

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The current interest in ion thrusters for near-Earth and deep-space missions has occurred because of long-term development efforts yielding an understanding of the physical phenomena involved in thruster operation and hardware that is suited to a wide range of missions. Features of state-of-the-art thrusters are described in terms of the various physical processes that occur within them. Tradeoffs that must be considered to arrive at thruster designs suited to commercialization are discussed.

Introduction

THE operation of ion thrusters in space began in 1964 with the launch of two thrusters on a brief ballistic flight designated Space Electric Rocket Test I (SERT I).[1] This test and similar Soviet tests[2] that were carried out at about the same time demonstrated 1) that a beam of high-velocity positive ions could be ejected continuously along with low-velocity electrons that neutralized the ion current and space charge and 2) that thrust was produced. Subsequent experiments like SERT II[3] and an Advanced Technology Satellite (ATS 6)[4] launched in 1970 and 1974, respectively, were intended to demonstrate the long lifetimes in space that would be required to accomplish missions of interest. The two SERT II mercury ion thrusters each performed properly for several months before high-voltage shorting problems developed and limited testing that could be done with ion beam extraction. Successful lifetime and functional tests of many thruster systems and components continued to be performed periodically, however, as long as the mercury propellant supply lasted (11 years). Even after that, periodic tests continued to be performed on some components such as heaters. Successful tests were conducted with SERT II for almost 22 years and 5792 h of full-power thrusting were accumulated on the two thrusters.[3]

The 1974 launch of the ATS-6 satellite with its two cesium ion thrusters was not as successful. Both thrusters failed to restart after a brief period of operation, apparently because of failures in the cesium feed system.[4] Before this event most people in the ion thruster community anticipated that opportunities to use ion thrusters would evolve rapidly toward the high total impulse missions where the high specific impulse capabilities of these thrusters would be most beneficial. However, potential users of this new technology were cautious; they desired a more mature product and the failures may have reinforced a general concern among them that ion thrusters were too complex. Also, few spacecraft could provide the power required by ion propulsion without major redesign. In this climate a period of declining interest in the technology evolved.

The ion propulsion community in the U.S. is, however, quite tenacious, and over the intervening period of almost 25 years researchers and technologists continued to refine their product for auxiliary and primary propulsion applications and to address user concerns. In particular, the SERT II thruster was scaled down to produce the Ion Auxiliary Propulsion System (IAPS) suitable for north–south stationkeeping of a U.S. Air Force satellite and scaled up to produce the Solar Electric Propulsion System (SEPS) for primary propulsion applications.[5] Both of these systems were the subject of extensive ground-based life testing, even though neither was used in space. During the IAPS and SEPS programs, extensive supporting experimental work was done and models of thruster processes were developed and applied to analyze thruster problems. In addition, the community began to pay greater attention to matters of cost, compatibility of thruster and spacecraft, reliability, and lifetime.

Presently, interest is high in NASA's Solar Electric Propulsion Technology Application Readiness (NSTAR) thruster for the New Millennium Deep Space-1 (DS-1) mission,[6] the Xenon Ion Propulsion System (XIPS-13)[7] is being marketed and deployed for near-Earth applications on a regular basis, and the XIPS-25 system[8] is coming on line. It appears that the ion thruster community goal of widespread user acceptance is near. A key to the realization of this goal may have been gaining the understanding necessary to realize a proper balance between a wide range of competing physical phenomena that are inherent in these devices. An objective of this paper is to point out some of these phenomena, to identify tradeoffs they infer, and to indicate how they have been addressed to arrive at thruster designs suited to deep-space and near-Earth missions. Contributions to the physics and technology of ion thrusters have been made by workers throughout the world. This paper is, however, focused on work that has led to the NSTAR and XIPS thrusters rather than being an exhaustive examination of ion thruster research and development.

Principles of Ion Thruster Operation

Essential elements of a cylindrical, direct-current (dc) ion thruster, which are shown in Fig. 1, enable the production,

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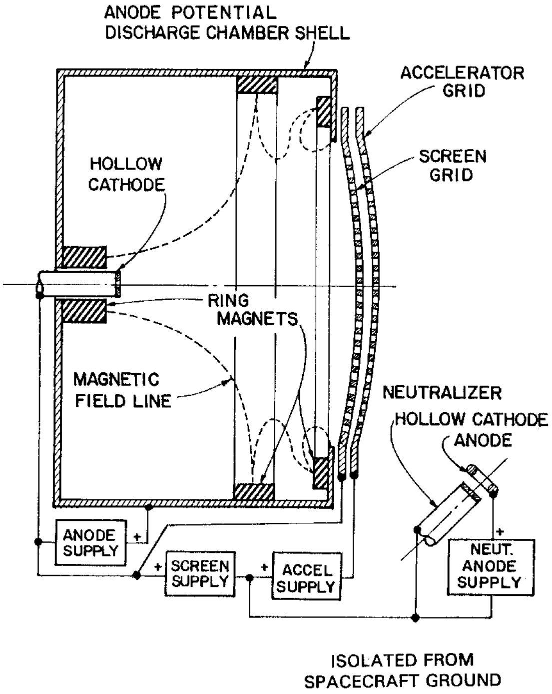

Fig. 1 Typical direct current ion thruster schematic diagram.

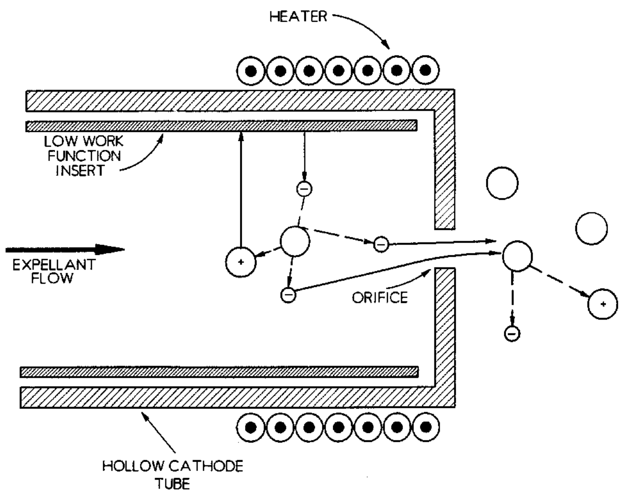

Fig. 2 Hollow cathode schematic diagram.

acceleration, and neutralization of a continuous stream of positive ions, i.e., an ion beam. As the figure suggests, at least four power supplies are required and they enable one to effect relatively independent control of the parameters associated with each of these processes.

The components of the main discharge chamber, wherein the ion production occurs, include a hollow cathode and an anode positioned within a magnetic field. The chamber is bounded on one side by an ion acceleration (grid) subsystem through which ions are extracted and accelerated into a high-velocity, well-collimated beam. A second cathode positioned downstream of the grids serves to eject electrons into the ion beam at the current required to prevent spacecraft charging and excessive ion beam divergence.

Orificed hollow cathodes, like the one shown in Fig. 2, have evolved as the preferred electron sources for the discharge chamber and the neutralizer.[9] These devices are operated by flowing gas through the cathode tube and orifice and heating the tube and low work-function insert with an external heater. When the cathode reaches its design operating temperature, it emits a substantial electron current thermionically, and electrons are extracted by applying a positive voltage to a downstream electrode, e.g., the anode. Under these conditions a dense plasma develops within the tube. Ions, which are produced within this internal plasma by sequential electron bombardment, are drawn into and heat the insert. Neutral atoms flow and electrons are accelerated through the orifice. In so doing, the electrons acquire sufficient energy to ionize neutral atoms downstream of the orifice, thereby producing either discharge chamber or neutralizer plasmas.

Electrons from the discharge chamber cathode, which are accelerated through a discharge voltage, i.e., a cathode-anode potential difference of a few tens of volts, produce ions within an enclosing magnetic field that enhances thruster efficiency. The ring-cusp magnetic field shape shown in Fig. 1, which is typical of current designs, has evolved as the preferred one.[10] Its specific purpose is to limit the migration of electrons and ions toward surfaces within the discharge chamber upon which they could recombine, thereby losing the energy expended in producing them. The field shape is also designed to facilitate ion diffusion toward the grids.

The ion extraction and acceleration process can be understood best by considering an exploded view of a single pair of apertures from the screen and accelerator (accel) grids. The apertures are shown in Fig. 3a, along with corresponding plots (Fig. 3b) of the potential variation through the solid webbing of the grids (solid line) and along the centerline of the apertures (dashed line). As this figure suggests, ions are created at a potential that is maintained above that of the ambient space plasma by the screen and anode power supplies. Many of the ions drift toward the screen grid, where they are first accelerated to a potential near that of the accel grid and then decelerated back up to space plasma potential. The electrons that reach the screen-grid sheath are reflected back into the chamber. Space plasma potential is realized at a downstream, fictitious surface (the neutralization surface), where electrons from the neutralizer mix in and the beam-ion plasma is formed. As Fig. 3a suggests, almost all of the ions remain within an envelope that has a low divergence, provided the proper balance is maintained between the ion current supplied to the screen-grid sheath from the discharge chamber and the current that can be accommodated by the grids. This latter current, which

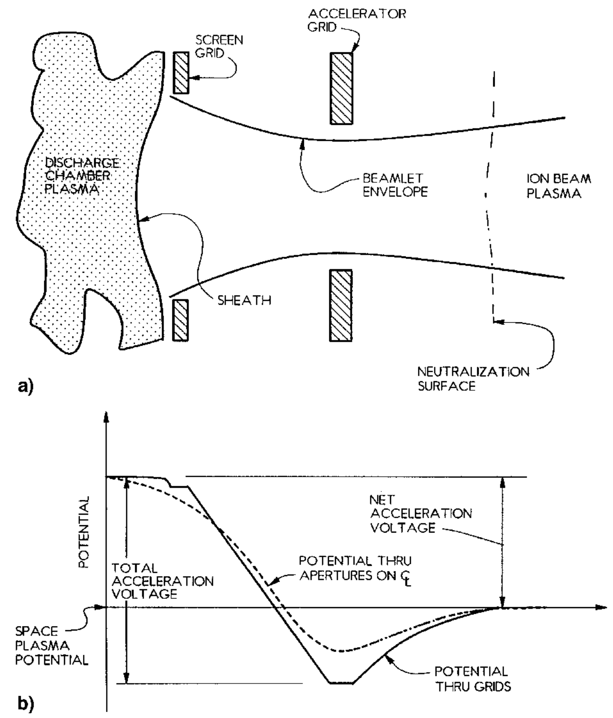

Fig. 3 Ion extraction/acceleration schematic for a single aperture pair.

is given by the Child-Langmuir law, is determined by the total acceleration voltage (Fig. 3b) and the geometrical parameters associated with the grids (separation, thicknesses, and hole diameters and shape).[11]

The ion current extraction capability through a pair of holes is increased primarily by reducing the spacing between the grids, while holding constant the ratio of hole diameter to grid spacing, or by increasing the total acceleration voltage. The magnitude of the negative bias applied to the accel grid can, therefore, be increased to increase the ion-current-carrying capacity of the grids. This bias also serves the important function of preventing electrons from the beam-ion plasma that exists downstream of the grids from being accelerated through the accel grid apertures and into the high positive potential surfaces within the discharge chamber. Although the net accelerating voltage could also be increased to increase the extracted current, it generally is not, because it determines the beam ion velocity. This velocity, in turn, determines the thruster specific impulse that is broadly fixed by system and mission considerations.

Variations on the dc ion thruster arrangement shown in Fig. 1 include 1) the radio frequency (rf) design,[12,13] in which electrons are accelerated to ionization energies using power radiated from an antenna, rather than by steady dc power; 2) the microwave (μW) design in which the ionization power for the discharge chamber and the neutralizer are both transmitted through waveguides[14]; and 3) the field emission electrostatic propulsion (FEEP) design, in which ions are formed from liquid propellant via direct ion emission that occurs in the same electric field that serves to accelerate them.[15] These designs are advantageous because they are conceptually simpler than the dc design. In the cases of the rf and μW designs, this is so because ionizing electrons are produced by accelerating electrons from previous ionizations in the radiated fields, and as a consequence, the hollow cathode in the discharge chamber can be eliminated. Discharge plasma ignition is achieved using electrons that are accelerated from the neutralizer. The FEEP thruster is even simpler because neither the discharge chamber nor its cathode are required. The disadvantages of these designs are centered in the fact that elimination of cathodes and/or discharge chambers reduces the flexibility of the thruster to operate efficiently over a wide range of thrust levels and specific impulses with many propellants. This paper will focus primarily on dc ion thrusters because of their state of development and the authors' familiarity with them.

Critical Development Trends

To increase the appeal of ion thrusters to the general user, a development trend over the past two decades has been toward simplification through elimination of components in both discharge chambers and power supplies, thereby, enhancing system reliability.[16,17] With the near-Earth mission focus that has prevailed, this has led to thrusters that operate at or near a single design point on inert gas propellant (xenon being preferred). Other changes have involved the replacement of electromagnets with permanent magnets and, in some instances, elimination or alteration of another electrode (the discharge cathode keeper), which is used in some designs to assure stable discharge operation.[3] These changes have made it possible to eliminate up to six power supplies. Beyond that, other power supplies have been simplified through the elimination of unnecessary components and control circuits.

In addition to these general trends, specific efforts have been focused on thruster subsystems that are critical because of their effect on cost, lifetime, reliability and performance of overall thruster systems. Critical changes that have been made to subsystems have been focused on 1) the grid configuration including alignment, fabrication, and mounting considerations; 2) the grid system lifetime; 3) the lifetimes of cathodes and adjacent structures; 4) thruster performance; and 5) the propellant. The development trends associated with each of these will be addressed separately.

Ion thrusters are known to be enabling for high total impulse missions, e.g., to the planets and beyond, because they involve optimum specific impulses that are high. They have now also become competitive for near-Earth, lower-total-impulse missions related to orbit raising, stationkeeping, and attitude control of satellites. The lower specific impulses that are optimal for these missions are achieved readily by simply lowering the net accelerating voltage. However, this also reduces thruster power and without other improvements, more thrusters would be required and propulsion system costs would be greater. The development of ion thrusters that operate at a lower specific impulse while maintaining a high-thrust density for near-Earth missions has been a greater challenge than development for the more ambitious high total impulse ones.

Grid System Configuration

The mechanical design of the grids, through which ions are extracted and accelerated and to which the voltages are applied is critical because the grids determine the current density, divergence, and kinetic energy of the extracted ion beam. These factors in turn determine the thrust density, thrust loss caused by divergence, and specific impulse (I_{sp}). Generally, it is best to operate at a high current density to minimize the number of thrusters required on a spacecraft. Because there is an optimum specific impulse for each mission and it is determined to first order by the net-accelerating voltage, high current density requirements are met by reducing the thickness of the screen grid and the spacing between it and the accel grid. As this is done, aperture diameters can be reduced and, consequently, the number of thrust-producing aperture pairs in the grids can be increased. The degree to which a close grid spacing can be maintained is limited by the thruster diameter and is complicated by the fact that the temperature distributions on the two grids induce thermal distortions as discharge chamber operating conditions change. These considerations have stimulated the following sequence of grid system development:

1) A transition from flat molybdenum grids that could sustain a span-to-gap ratio of near 50 to ones that were dished so they would deform predictably with temperature changes and could sustain span-to-gap ratios about an order of magnitude greater.[18,19]

2) The use of chemical etching rather than drilling to fabricate apertures in molybdenum grids, thereby reducing cost and improving aperture position accuracy and quality.[18]

3) Introduction of slight radial misalignment between apertures in the screen and accel grids to deflect the individual beamlets electrostatically, thereby correcting the overall beam divergence and attendant thrust loss introduced by dishing the grids.[20]

4) Introduction of compliant grid-mounting schemes that allow differential radial expansion between the grids and support frame to minimize the grid surface distortion that induces aperture misalignments.[21]

5) Substitution of graphite–graphite composite materials in place of molybdenum.[22,23] Composite materials, which are the subject of current study, are advantageous because they can be fabricated so that they are thin and have a high stiffness that restrains grid deflection toward each other. A composite material can also be made so it has an essentially zero thermal expansion coefficient over the operating temperature range of an ion thruster. This, in turn, may mean that no dishing or compensation will be required to maintain the alignment of the very small holes that are required at very close grid spacings. It may also be possible to extract the ion beamlets through slots rather than circular holes in composite grids.[24] It is expected that this would simplify graphite-fiber layup and hole drilling, thereby reducing composite-grid costs.

Grid System Lifetimes

In discussing lifetimes of individual components that could limit the overall thruster lifetime, it should be noted that the low-thrust characteristics of ion thrusters typically necessitate long lifetimes to accomplish missions of interest (of the order of 10^4 h). Hence, the critical wear processes typically proceed at very slow rates and the times required to conduct life tests can become very long, thereby making the tests themselves expensive. The use of computer-monitored and controlled facilities for life tests has become essential to the cost effectiveness of these tests.[25]

The lifetimes of the screen and accel grids are determined by the process of sputter erosion by ions that bombard a grid surface with sufficient energy to cause atoms of grid material to be ejected. For the screen (upstream) grid, the sputtering is caused by the direct impact of low-energy ions from the high-density plasma produced within the discharge chamber. The discharge (cathode-to-anode) voltage within this chamber determines the kinetic energy at which these ions strike the grid and it can be and generally is held at a sufficiently low value so that the singly charged ion kinetic energy will be near or below the sputtering threshold of the grid material. There are, however, doubly charged and possibly triply charged ions produced in the discharge chamber with, respectively, double and triple the usual (singly charged) ion kinetic energy that will still sputter erode the screen grid.

The need to have the thinnest possible screen grid to assure the shortest ion acceleration distance and a high-thrust density has necessitated operating condition changes that assure very low densities of multiply charged ions. Specifically, the discharge voltage and ion-to-neutral-atom ratio at which the thruster operates have been reduced to levels where erosion rates yield acceptable mission lifetimes for the thin screen grids required to yield good thruster performance.[26] It is also important to hold the screen grid at or above cathode potential to minimize the energy and, therefore, the sputter erosion rate of the screen grid.[27]

The grid system is designed so that ions being extracted into the ion beam do not strike the accel grid directly and sputter erode it. There are, however, both fast-moving ions and slow-moving neutral atoms escaping through the grids, and a small fraction (of the order of 1%) of the ions will capture an electron from an adjacent neutral during the brief interval when the ion is being accelerated into the exhaust beam. Typically, such a charge-exchange event yields a fast-moving neutral that escapes downstream and a slow-moving ion that is drawn into and sputter erodes the accel grid.[28,29] Even though the current of these ions is relatively small, they impact at high energies and they tend to be focused into specific regions between holes on the downstream surface of the accel grid when only an accel and a screen grid are used. As a result, they can cause it to erode and fail. The thickness of the accel grid doesn't affect ion extraction performance significantly, and so they are made thicker than screen grids. Their lifetimes have been extended by reducing the magnitude of the negative voltage applied to this grid, thereby reducing the kinetic energy of the charge-exchange ions that bombard it.[29]

The extent to which the magnitude of the accel grid voltage can be reduced is limited by the fact that electrons can be drawn upstream from the beam plasma through the grid apertures and into the discharge chamber if the potential on the accel grid is not sufficiently negative.[30,31] This phenomenon is called electron backstreaming and it is undesirable not only because it reduces thruster efficiency and may cause excessive component heating, but also because it gives a false indication of ion beam current and, therefore, a false indication of thrust and specific impulse.

If it were possible to increase the magnitude of the negative voltage applied to the accel grid without adversely affecting accel grid erosion, the ion beam current density and, therefore, the thrust density could be increased. Research has shown, however, that increasing the magnitude of the accel voltage causes an increase in the divergence of the ion beam.[32] The problems of both the divergence and accel grid erosion on its downstream surface can be addressed by installing a third (decel) grid downstream of the accel grid. This third grid is maintained at a potential near that of the ambient space plasma. To some extent, it provides a physical shield against erosion on the downstream surface of the accel grid, and it helps to maintain the focus of the ion beamlets, thereby reducing the overall beam divergence over a wide range of accel-grid potentials.[32] A decel grid adds the mechanical complication of maintaining the position of a third closely spaced grid, but it has the beneficial effect of shifting most of the accel grid charge-exchange sputter erosion from its downstream surface to the cylindrical (barrel) regions on the apertures.[33] Barrel erosion has a lesser effect on accel grid lifetime because the initial accel grid hole diameter is quite small and the time required to erode it to the point where the accel grid would fail mechanically is long. Enlargement of these holes does, however, bring about an ever-increasing loss of neutral propellant from the discharge chamber that results in a corresponding reduction in overall thruster efficiency over its lifetime. Barrel erosion is also less troublesome because much of the sputtered material is redeposited in the grid apertures and eventually ends up on the upstream side of the decel grid, the downstream side of the screen grid, or within the discharge chamber. The net effect is that only about 12% of the sputtered material escapes the thruster, whereas nearly all of the material sputtered from the accel grid of a two-grid system escapes into the beam-plasma region and becomes a potential spacecraft contaminant.

Lifetimes of Cathodes and Adjacent Structures

Although early versions of ion thrusters used refractory wire filaments[34] and oxide cathodes[35] as electron emitters, they were eventually abandoned in favor of orificed hollow cathodes. Hollow cathodes are presently used as electron sources for both the main and neutralizer discharges because they 1) are mechanically simple and rugged, 2) exhibit long lives, 3) can be shut down and restarted readily and repeatedly, and 4) lend themselves to ground testing and subsequent exposure to the atmosphere before they are launched into space.[36] These cathodes employ porous tungsten inserts that are impregnated with a low work-function (barium–oxygen) material.[37] Although overheating and contamination of the electron emission surface within the cathode itself can cause rather rapid failures, these events can be controlled through proper design and adequate propellant purity.[38] The lifetimes of hollow cathodes that are designed and operated properly are determined by sputtering phenomena similar to those that limit grid lifetimes. Cathode life tests conducted at electron emission current levels needed for missions of current interest have, however, been shown to exceed 20,000 h for both mercury and xenon propellants.[39–41]

Despite the fact that present hollow cathode designs appear suitable for missions of interest, there is a need to understand the details of cathode sputtering phenomena so that effects of design changes can be predicted, and long, expensive life tests of cathodes or thrusters will not be required each time such changes are made. The mechanism by which sputter erosion does occur has, however, been somewhat elusive because the voltage differences applied between electrodes in the cathode region are insufficient to accelerate ions to energies above the sputtering threshold levels and multiply charged ions are not expected. One theory that has been proposed involves the development of a potential peak immediately downstream of the cathode.[42] According to this theory, ions produced near the summit of such a hill could have sufficiently high energies and could, depending on the operating condition, be expected to flow away and induce sputtering in all directions. This would be expected to cause erosion of surfaces on both the cathode itself and on structures adjacent to the cathode. Erosion on these surfaces has been observed after thruster and thruster-component life tests that are generally consistent with energetic ion flux measurements. Rates of erosion on components are strongly dependent on the electron emission rate from the cathode, on propellant flow rate, and on the pressure immediately downstream of the cathode, where it is postulated that the potential hill develops. There is evidence that an auxiliary electrode that encloses the cathode (an enclosed keeper) most likely reduces sputter erosion because it increases the pressure and, hence, the ion scattering near the orifice.

Discharge-Chamber Performance

Overall ion thruster efficiency, η_t, defined in terms of thrust, F, propellant flow rate, $\dot{m}_p$, and input power, P, is related to the efficiencies for the utilization of electrical power, η_e, and of propellant fed into the discharge chamber, η_u, and to the effects of beam divergence and multiply charged ions, η_d, through the expression

$$\eta_t = F^2/2\dot{m}_p P \approx \eta_e \eta_u \eta_d^2 \tag{1}$$

It should be noted that the approximate equality sign appears in Eq. (1) because discharge chamber rather than total propellant utilization efficiency has been used. The propellant utilization efficiency based only on discharge chamber flow is, however, only slightly greater than the one based on total flow because neutralizer and fixed flow losses are generally small.

Ion beam divergence effects, which can be controlled through proper design and by maintaining appropriate operating voltages, are typically small as is the effect of multiply charged ions. Consequently, the η_d term in Eq. (1) is near unity, particularly in state-of-the-art thrusters that are designed to operate at the low discharge voltage condition that assures a long screen grid lifetime. Discharge-chamber propellant utilization and electrical efficiencies, on the other hand, generally have values in the 70–90% range and they are dependent upon each other. Their interdependence develops because thrust-producing ions and unionized neutral atoms escape through the grids. The relative amount of each is determined by the discharge-chamber power per unit propellant supply rate. Specifically, the fraction of propellant supplied that escapes as ions, i.e., η_u, can be increased by expending more discharge chamber (ionization) power. This power, most of which does not appear directly as kinetic power in the beam, is frequently expressed as the energy required to produce an ion. Because η_e is the ratio of the kinetic energy given to an average beam ion over the total energy required to both produce and accelerate it, the thruster efficiency from Eq. (1) can be rewritten

$$\eta_t \approx \frac{\eta_u \eta_d^2}{1 + (\varepsilon_B / V_{\mathrm{net}})} \tag{2}$$

In this expression, V_{net} is the voltage difference through which ions are accelerated into the exhaust beam, and ε_B is the energy cost of producing an average ion (eV/ion = watts/ampere) that undergoes the acceleration.

As Eq. (2) suggests, it is the balance between the effects of the beam ion energy cost and propellant utilization efficiency that determines to first order the overall thruster efficiency. The nature of the relationship between these two quantities (ε_B and η_u) is illustrated qualitatively in Fig. 4. This figure reflects the fact that ions are produced via energetic electron bombardment of propellant atoms in the discharge chamber. When the rate of electron supply from the main cathode is low, i.e., at low electron emission, most of the energy being supplied to the electrons goes into producing ions and ε_B is at its baseline value, but many neutral atoms also being supplied will not be ionized and will escape through the grids as atoms. Thus, η_u will also be low. As the rate of electron supply is increased, the fraction of these electrons that fail to ionize a propellant atom and reach the anode with most or all of their kinetic energy also increases. The power loss associated with these electrons causes ε_B to increase, but their greater density also induces increased ionization, thereby causing η_u to increase.

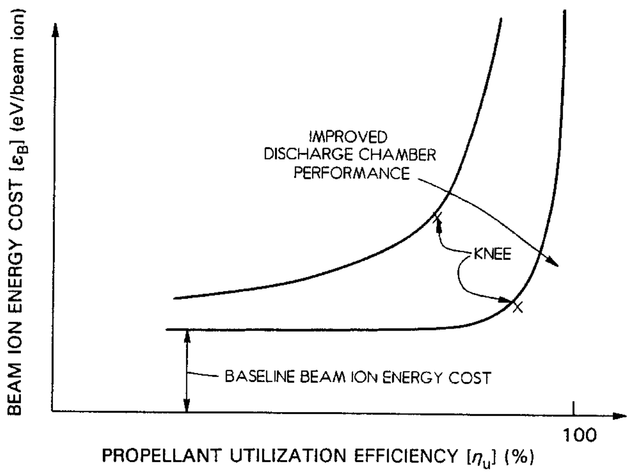

Fig. 4 Discharge chamber performance characterization.

As electron emission levels reach high values, most of the propellant is being ionized (η_u approaches unity), and many of the electrons fail to ionize atoms. As a result, they carry their energy to the anode and the beam ion energy cost rises abruptly. Generally, it is best to operate at the knee of the performance curve shown in Fig. 4. The quality of a discharge chamber is determined by the extent to which this knee can be moved to greater propellant utilization efficiencies and lower beam ion energy costs. It is noteworthy that the net accelerating voltage can be varied without affecting the performance values at the knee because the processes of ion production and acceleration are separable in gridded ion thrusters.

Equation (2) shows that it becomes increasingly important to have a low energy cost per beam ion as the net accelerating voltage is reduced, i.e., as less energetic missions characterized by lower optimum specific impulses are pursued. This trend toward lower net accelerating voltages has necessitated the following changes in discharge chamber design to mitigate the drop in thruster electrical efficiency that would have otherwise occurred:

1) Reductions in the diameters of apertures in the accel grids[43]: Because the ions are focused into beamlets that neck down to a minimum diameter as they pass from the screen grid through the accel grid (Fig. 3), it is possible to reduce the accel aperture diameters, thereby limiting the loss of neutral atoms without limiting the ejection rate of beam ions. These so-called small-hole accelerator grids (SHAG) effect a substantial improvement in performance by increasing the propellant utilization efficiency at all beam ion energy cost levels.

2) Improvements in the configuration of the magnetic field that serve to limit the migration of energetic electrons to the anode until after they have given up most of their energy in ionizing collisions: Steady reductions in beam ion energy costs have been realized as ion thrusters have developed by changing the magnetic fields through a sequence of simple solenoidal,[44] divergent,[3] radial,[45] and line- or ring-cusp[10] configurations. Currently, ring-cusp configurations are generally used. They employ rare-Earth magnets and are designed to produce fields that are strong near the chamber walls where anode-potential surfaces are located and weak throughout a relatively large central region of the chamber.[10] It is postulated that this magnetic configuration, like the magnetoelectrostatic containment concept that preceded it,[46] effect a level of electron confinement that is sufficient to induce local electric fields that also cause ion reflections back toward the center of the chamber. By so doing, ions are prevented from reaching surfaces on which they could recombine and are instead directed ultimately toward the grids. Hence, fields have been designed that prevent not only the direct loss of energetic electrons, but also the loss of ions that give up their ionization energy when they recombine. Both effects reduce the beam ion energy cost.

Propellant Trends

An ion thruster propellant with a high atomic weight is preferred because more massive atoms have a lower thermal velocity and, consequently, a lower rate of loss through the grids in the unionized state. Further, the ionization energy is similiar for most atoms and a greater mass requires a greater net acceleration voltage to achieve a given specific impulse. Equation (2) shows that these trends cause thruster efficiency to increase with atomic weight. They also cause thrust-to-power ratio to increase with atomic mass. Propellants with a high second ionization potential and a low charge-exchange cross section are also desirable because they enable long thruster lifetimes. Finally, it should be convenient to store the propellant without boiloff or the need for a complicated tank and propellant control and handling system.

Consideration of these factors led to the early use of cesium and mercury as propellants. Both have now been rejected in favor of xenon; mercury because of human toxicity that led to substantial costs associated mostly with its handling for ground-based testing; and cesium because of a high surface tension that enabled migration and eventual coating of insulators during space tests.[4] Xenon, which gives the best balance in meeting all of these compatibility and performance goals, is used on thrusters now being launched for Earth-orbital applications, even though it is somewhat costly compared with other propellants. An ultimate goal for planetary missions is to use spacecraft waste or material mined from space as the propellant, but for the present time it appears best to use one that is a simple, inert element like xenon.

Recent interest in a fullerene (C_{60}) propellant was driven by its large molecular mass (720 amu). Its high mass necessitates higher net acceleration voltages and, therefore, makes it particularly attractive for low specific impulse missions. Unfortunately, the tendencies of these molecules to fragment and form negative ions, thereby degrading the performance of the discharge chamber, appear to make C_{60} unsuitable as a propellant.[47]

Thruster Comparisons

The extent to which both changes in propellant and discharge chamber performance have changed over the past three decades is indicated by the data in the last two columns of Table 1. This table, which gives characteristics of U.S. thrusters that have been launched into space, also shows trends in specific impulse and nominal discharge voltages. The tabulated propellant utilization efficiency is associated with the discharge chamber alone and does not reflect other propellant losses such as those associated with the neutralizer. It is noteworthy that lowering the discharge voltage over the range given in Table 1 degrades discharge chamber performance slightly. However, if it is done along with other changes, e.g., SHAG optics and good magnetic field design, including reductions in the area of surfaces on which ions can recombine, the discharge voltage, V_D, can be lowered without substantial performance degradation. Thus, V_D has been reduced for each new generation of thrusters to increase the screen grid lifetime.

Other mature ion thruster designs that were or are being flight qualified but have not been flown are listed in historical sequence along with their characteristics in Table 2. These thrusters, which have contributed substantially to the development of the technology, include those associated with SEPS,[48] IAPS,[49] the NSTAR program,[27] and the 25-cm-diam XIPS-25.[8]

The data of Table 2 also illustrate the historical trend for V_D and ε_B to decrease at a nearly constant propellant utilization efficiency, although the trend is masked somewhat by changes in thruster size (beam diameter). As size is increased, for example, the constant-η_u values of V_D required to sustain a stable discharge and the beam ion energy cost tend to decrease. The sizes of ion thrusters that have been built and tested actually range beyond those given in Table 2 from 5 cm diameter[50] to 150 cm diameter.[51]

Miscellaneous Lifetime Considerations

Other thruster components such as heaters,[52] propellant feed systems,[53] and control and power supply subsystems have also been tested and refined in individual programs and as part of overall thruster development efforts to assure that they will meet lifetime requirements. Finally, it is noteworthy that the sputter erosion of discharge chamber components results in the deposition of sputtered atoms throughout the discharge chamber. These atoms will be resputtered from cathode-potential surfaces that are not well shielded by magnetic fields, and they will tend to accumulate on anode-potential surfaces.[54] This accumulation can, after long periods of operation, result in flakes of sputtered, electrically conducting material that can detach and drift into the grids. When this happens, arcing and even shorting between the grids occurs. This potential problem has been addressed in the thrusters by fabricating anode-potential surfaces with textures that hold the flakes or limit their sizes to very small dimensions. In addition, high-voltage power supplies or grid-clearing circuits are designed so that they can vaporize flakes of sputtered material that may lodge between the grids.

Contamination of spacecraft surfaces by sputtered materials or of spacecraft electronic functions by electromagnetic noise associated with thruster plasmas have also been the subject of extensive numerical studies, which have suggested that neither is likely to represent a problem. The strongest evidence that these effects are benign, however, comes from SERT II ex-

Table 1 Characteristics of key U.S. thruster generations

Flight thruster	Launch date	Beam diameter, cm	Propellant	Specific impulse, s	V_D, V	η_u, %	ε_B, eV/ion
SERT I	1964	10	Mercury	~5000	46	~80	~700
SERT II	1970	15	Mercury	4200	37	80	220
XIPS-13	1997	13	Xenon	2565	25–28	90	~220

Table 2 Characteristics of flight-qualified thrusters

Thruster	Circa	Beam diameter, cm	Propellant	Specific impulse, s	V_D, V	η_u, %	ε_B, eV/ion
SEPS	1981	30	Mercury	3100	32	88	170
IAPS	1983	8	Mercury	2550	37	84	258
NSTAR	1998	30	Xenon	3100	24	90	185
XIPS-25	1998	25	Xenon	3800	25–28	96	115

perience in space.[3] Results from these tests gave no indication that significant noise or material contamination had occurred in spacecraft systems or on either the solar array or thermal control surfaces, even though grid sputtering was abnormally high, and mercury, which could condense on these surfaces, was being used as the propellant.[3,33]

Conclusions

The application of ion thrusters for near-Earth missions that are presently occurring suggests that a steady growth in ion thruster use can be expected. Such growth is considered likely over the wide range of specific impulses associated with missions in which either payload fraction or propulsion system thrust-to-power ratio are maximized. The trend should accelerate as spacecraft bus power levels increase and as missions become more ambitious. Missions can be expected to become more ambitious, for example, as 1) the number of satellites in busy orbits increase and tighter position tolerances are required, 2) stationkeeping lifetimes increase in response to demands for lower satellite life-cycle costs, and 3) missions to other planets and beyond become more appealing. Other natural trends will be toward power sources with lower specific masses, greater thruster and power conditioner efficiencies, and greater nonpropulsion power demands. All of these trends point toward increases in the optimal specific impulse and this in turn favors the application of ion thrusters that can not only deliver these specific impulses but that become more efficient as operational specific impulses increase.

Because the processes of energetic electron production, ion production, ion extraction and acceleration, and ion beam neutralization are all separable in ion thrusters, and because each can be controlled independently, these devices afford designers and users great operational flexibility. To use this flexibility, however, complexity may have to be incorporated into the design. The user apprehension that accompanied ion thruster development after the SERT II and ATS-6 missions should be a constant reminder to avoid the temptation to provide unnecessary flexibility. Thruster designs should be as simple as possible, consistent with essential mission goals.

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