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Electric Propulsion—Characteristics, Applications, and Status

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ELECTRIC PROPULSION - CHARACTERISTICS, APPLICATIONS, AND STATUS

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Abstract

As chemical propulsion systems were achieving their ultimate capability for planetary exploration, space scientists were developing solar electric propulsion as the propulsion system need for future missions. This paper provides a comparative review of the principles of ion thruster and chemical rocket operations and discusses the current status of the 30-cm mercury ion thruster development and the specifications imposed on the 30-cm thruster by the Solar Electric Propulsion System program. The 30-cm thruster operating range, efficiency, wear out lifetime, and interface requirements are described. Finally, the areas of 30-cm thruster technology that remain to be refined are discussed.

Introduction

Data recently obtained from the Voyager I and II missions produced a substantial amount of new information and led to new understanding of Jupiter and the self-contained "solar system" it forms with its moons. These missions were performed with chemical propulsion systems and were representative of the ultimate capability for planetary exploration that is achievable using conventional, proven technology. Future missions will become more extensive and difficult to accomplish with chemical propulsion alone. In chemically propelled vehicles, all of the energy required for propulsion is stored in the propellant and must be carried (and launched) on board the space vehicle.

During the past 20 years, space scientists have assessed the expanded capabilities that are available when solar energy is used for propulsion of space vehicles. The approach that appears most promising at this time is referred to as solar electric propulsion and involves photovoltaic conversion to solar energy (solar cells) to produce electric power and subsequent conversion of electric power to produce thrust (ion thrusters). The key technologies which form the basis for a solar electric propulsion system (SEPS) are therefore solar cells and ion thrusters. In order to increase the propulsion capability over that of a chemical system, the total energy and thrust conversion capability (total impulse) of the SEPS must exceed that of the chemical system (per unit of mass). This requirement dictates low mass, high power solar arrays and high specific impulse, high efficiency conversion of electrical power to thrust.

To meet the propulsion requirements that have been projected for future missions, the NASA Office of Aeronautics and Space Technology (OAST) has sponsored a technology program at the Lewis Research Center (LeRC) for development of an ion thruster to provide prime propulsion in a SEPS. In support of this technology program, Hughes Aircraft Company has participated in the development, fabrication, and testing of 30-cm mercury ion thrusters under contract to NASA's LeRC. The latest version of this thruster is known as the J-series thruster. Current development is concentrating on verification of design modifications that have been incorporated in the J-series to eliminate real and potential design deficiencies that were uncovered in testing of the previous series thruster designs.

This paper presents a general comparison between chemical and electrical propulsion. Also presented is a brief description of the principles of operation of an ion thruster and discusses recent progress and status of the development of the 30-cm diameter mercury ion thruster that has been developed for conversion of electric power to thrust.

Background

Chemical Rockets

To date man has relied on chemical rockets to enable him to explore space. A chemical rocket gets the energy it needs to perform a mission from the energy bound in the molecules of the propellants (fuel and oxidizer) which is subsequently released upon combustion of the propellants. A chemical rocket is comprised of a combustion chamber, one or more storage tanks for the propellants, and a supply system for feeding the propellants to the combustion chamber as shown in figure 1.

The propellants are injected at high pressure into the combustion chamber, where they are atomized, mixed, and burned. Hot gases under high pressure are produced from burning of the propellants (fig. 2(a)). These gases except equal pressure in all directions on the walls of the chamber (fig. 2(b)), but in the rocket these gases are allowed to escape through a nozzle. From Newton's laws (force = mass times acceleration, i.e., action = reaction) we get a reaction force (referred to as thrust) that propels the rocket forward (see fig. 2(c)). Total impulse (thrust x time = total impulse) and specific impulse (total impulse ÷ propellant mass = specific impulse) are standard measures for comparing a rocket's performance. These terms will be used for comparing chemical and ion propulsion.

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Solar Electric Propulsion

In a solar electric propulsion system (SEPS), the energy required to produce the propulsive force, or thrust, is provided by the sun. Solar energy is converted into electrical energy by photovoltaic solar cells and electrical energy‡ is converted into thrust by the ion thruster. In an ion thruster the propellant is ionized in a discharge chamber (penning-type discharge) and the ions are then accelerated electrostatically, and expelled from the thruster at high velocity (see fig. 3). Electrons are injected into the thrust beam to maintain space charge neutrality. As will be discussed later, the choice of propellant depends on several system parameters but in general, it can be stated that materials with high atomic mass provide the best system performance. Therefore, most ion thruster development has been concentrated on operation with mercury as the propellant. Consequently, the propellant is stored as a liquid, and must be vaporized before ionization. Electrical inputs for vaporization and ionization do not contribute the production of thrust and therefore are the principjal inefficiencies of the ion thruster. The exhaust velocity of the ions is a design variable based on the accelerating voltages V^+ and V^- (shown in fig. 3) and therefore specific impulses in the 2000- to 5000-second range are readily available with the same generic thruster design. The construction details of an ion thruster are shown in figure 4.

Performance Comparison

Equations

The thrust which propels a vehicle is equal to the rate of change of vehicle momentum. As mentioned previously this thrust is the reaction to the action of the propellant flow from the vehicle and can be expressed as:

$$T = M_s \frac{dv_s}{dt} = \dot{m}_p v_p \qquad (1)$$

where

- T thrust
- M_s instantaneous mass of spacecraft
- v_s instantaneous velocity of spacecraft
- $\dot{m}_p$ propellant flow rate from spacecraft
- v_p exhaust velocity of the propellants (ion)

If the thrust produced by a vehicle is constant over a period of time, the total impulse is then given by:

$$T\,\Delta t = \dot{m}_p v_p \,\Delta t = I_t \qquad (2)$$

where

- Δt time interval over which thrust is being produced
- I_t total impulse

‡A power processor is required to convert the unregulated power from the solar arrays into the currents and voltages (electrical energy) needed by the ion thruster.

§Effective exhaust velocity is equivalent to specific impulse; $v_p'/g = I_{sp}$, where g is the acceleration of gravity.

The specific impulse imparted by a rocket is given by:

$$\frac{I_t}{M_p} = \frac{\dot{m}_p v_p \,\Delta t}{M_p} = I_{sp} \qquad (3)$$

where

- M_p mass of propellant used
- I_{sp} specific impulse

Comparison

A chemical rocket develops a large thrust (needed to lift a payload off the ground) with high propellant flow rates and a low effective exhaust velocity§ (v_p') for a short time using large amounts of propellants. On the other side of the spectrum, an ion thruster develops a low thrust (not sufficient for lifting a payload off the ground, but capable of propelling a payload in space) with low propellant flow rates and high exhaust velocity for periods up to years using a relative small amount of peopellant. Figure 5(a) illustrates the difference in performance between a chemical rocket and an ion thruster. Figure 5(b) makes a comparison between a solar electric stage and the Centaur stage. From equation (2) we know that the total impulse produce is equal to thrust x time.

The performance value presented in figure 5(b) are presented go give the reader an appreciation for the difference between chemical propulsion and electrical propulsion. The Centaur (used as a second stage) delivers a thrust of 133,440 Newtons (30,000 lb) from two $H_2 + O_2$ engines for approximately 440 seconds giving a total impulse of 58.7×10^6 N-sec (13.2×10^6 lb-sec). In comparison eight (8) 30-cm ion mercury thrusters each producing a thrust of 0.135 N (0.03 lb) for 617 days (5.33×10^7 sec) will deliver a total impulse of 56.9×10^6 N-sec (12.8×10^6 lb-sec). Using equation (3), the specific impulse for the Centaur stage is 443.5 seconds, while that for the solar electric stage as shown in figure 5(b) would be 2903 seconds. From this comparison of the total impulse and specific impulse of the Centaur stage and the solar electric stage (fig. 5(b)), it can be seen that a large percentage of the energy spent in a chemical rocket is for carrying along the weight of propellant. The savings in propellant weight between a chemical rocket and an ion propulsion system would go into additional payload capability.

Ultimately both chemical and electrical propulsion will be needed for future missions. To escape the earth's gravitational field and put a payload into space we need a high thrust (a big push) for a short duration. Thus we have the need for chemical propulsion. Because of its combination of high performance and low thrust, electric propulsion is ideal for transferring large space delicate structures from low earth orbit to geosynchronous orbit. Electric propulsion is also suited for interplanetary travel in the low gravity field and vacuum of free space.

30-cm Mercury Ion Thruster Description

The 30-cm mercury ion thruster has been developed to provide the low thrust, high specific impulse engine needed to satisfy the requirements of future planetary and earth orbital missions. A cross section of the 30-cm thruster is shown in figure 6. The main subassemblies identified in figure 6 are:

(1) The discharge chamber where the neutral propellant is ionized.

(2) The main vaporizer-isolator (MV-I) assembly where liquid mercury is vaporized and the neutral mercury vapor is conducted from the propellant storage system potential to the discharge chamber potential without ionization (maximum of about 1100 V). A sintered porous tungsten plug is employed in the MV-I as a phase separator between the liquid mercury and its vapor.

(3) Cathode-vaporizer-isolator (CV-I) assembly where mercury is vaporized, as in the case of the MIV, and then supplied to the hollow cathode which emits electrones (aided by thernionic emission from a low work function thermionic emitter) into a low-voltage discharge. These electrons from the hollow cathode discharge are then used to create mercury ions from the vaporized mercury in the discharge chamber.

(4) The magnetic pole pieces and baffle which shape the magnetic lines of force generated by the axial and radial permanent magnets to form a "magnetic confinement bottle" that "contans" the discharge electrons. This "containment" forces the electrons that are injected from the hollow cathode discharge into the discharge chamber to travel a longer (spiral) path and encounter more collisions with the propellant gas atoms, thus improving the ionization efficiency.

(5) An ion extraction system which extracts and accelerates the propellant ions from the discharge chamber to form a collimated ion beam (thereby producing thrust). The ion extraction system consists of two grids that have approximately 15,000 pairs of matched apertures that are accurately positioned to focus the ion trajectories into parallel "beamlets." The grid adjacent to the discharge plasma is commonly referred to as the screen grid and operates at the discharge chamber potential (approximately 1100 V). The grid adjacent to the extracted ion beam is called the accelerator grid and operates at a voltage sufficiently negative (approximately -300 V) to prevent backstreaming of the neutralizing electrons (injected into the extaracted ion beam) into the acceleration region between the extraction system grids.

(6) Neutralizer Vaporizer-Isolator (NV-I) where mercury is vaporized and supplied to the neutralizer hollow cathode. The neutralizer hollow cathode generates as low-voltage discharge plasma (aided by thermionic emission from a low work function surface) from which electrons are drawn into the extracted ion beam to equal, in number, the ions leaving the thruster.

Mission Requirements Effects

Thruster Versus Impulse

Planetary exploration, near earth orbit, and manned space station missions call for the development of a high energy propulsion system with a maximum of fuel efficiency with a minimal propulsion stage weight. Ion thrusters have been shown to be capable of delivering high energy propulsion while requiring small amounts of propellant. In designing an ion-thruster propulsion system one must address the need of thrust versus high specific impulse (high energy). The thrust produced by an ion thruster can be expressed in electrical terms as shown below.

From equation (1), $T = \dot{m}_p v_p$; m_p can be expressed in terms of the beam current

$$\left[\dot{m}_p = J_b \cdot (m/e)\right] \tag{4}$$

amd v_p can be expressed in terms of the net accelerating potential

$$\left(1/2\ m v_p^2 = eV_b\right) \tag{5}$$

where:

J_b beam current (amperes

(m/e) mass to change ratio ($\approx 2.07\times10^{-6}$ kg/coulomb for mercury)

V_b net acceleration voltage for beam ions (Volts)

From the above relationships it can be shown that the equation for thrust reduces to

$$T = 2.04\ J_b \sqrt{V_b}\ \text{(mN)} \tag{6}$$

From equations (2), (3), and (6), the specific impulse

$$(I_{sp}) = \frac{T x \Delta t}{M_p g} \tag{7}$$

and

$$M_p = \dot{m}_p x \Delta t = J_b (m/e) \Delta t \tag{8}$$

Using the relationships above and equations (4) and (5) we can derive at the following:

$$I_{sp} = \left[\sqrt{2(m/e)}\ J_b \sqrt{V_b}\right]\left[\frac{1}{J_b}\ (e/m)\right] \div g \tag{9}$$

$$= \frac{\sqrt{2(e/m)}}{g} \times \sqrt{V_b} \tag{10}$$

$$= 100\sqrt{V_b}\ \text{(seconds)} \tag{11}$$

The typical 30-cm mercury ion thruster operating profile is shown in figure 7. From the above deviations the specific impulse is a function only of screen voltage (since $V_b = V_{screen} + V_{discharge} - V_{neutralizer}$ and $V_{discharge}$ and $V_{neutralizer}$ are negligible when compared to V_{screen}). The thrust is a function of both beam current and screen voltage. Using equations (6) and (11) and referring to figure 7, it can be seen that throttling a thruster near the top of the operating profile will provide a high specific impulse at the expense of thrust. But moving along a constant power line (indicated by wattage) to a throttle point in the lower half of the envelope will increase the thrust at the expense of specific impulse. Mission considerations will determine the most favorable region within the operating profile of figure 7 for operation of the thruster. Also, the throttle point at which the thruster is operated will determine the power requirements (see fig. 7) and thereby it impacts the power system weight.

Propellant

The propellant weight and the storage tanks to hold the propellant should both be minimized in order to increase payload capability. The specific impulse (exhaust velocity) (has been shown to be a function of the propellant mass (eq. (7)).

Figure 8 illustrates the relationship between specific impulse and total propellant mass (at constant total impulse).

Figure 8 shows that increasing the specific impulse reduces the total propellant mass. Consequently, the system designer would like to operate the thrusterse at as high a specific impulse as possible from the standpoint of propellant mass requirement.

Power System Mass

The power required to produce a given thrust (or total impulse) is given by:

$$P_o = \frac{\dot{m}_p (I_{sp} \cdot g)^2}{2} \qquad (12)$$

Since the power system mass is directly proportional to the power output required (solar array area + power processing), operation at high specific impulse increases the power system mass. Figure 9 illustrates the effect of specific impulse on the power system mass.

A propulsion system should be optimized to deliver a maximum payload (or minimize propulsion system mass). The two largest contributions to the mass of an ion propylsion system are the propellant mass (incluiding the mass of the tankage) and power system mass. Figure 10 illustrates how consideration of the propellant mass and power systems mass should be employed to optimize the propulsion system mass and determine the optimum specific impulse.

Unique Requirements on Thruster Hardware Wearout Mechanisms

Missions currently under consideration for electric propulsion require a useful propulsion system lifetime of 15,000 hours or more. Also, during a given mission, an ion thruster must be capable of surviving numerous startups and shutdowns and be capable of operation at different throttle points (fig. 7). In designing a thruster to meet these requirements, special attention has been given to satisfy conditions for preventing early failure of the thruster because of the dominant wearout or failure mechanisms., For 15,000 hours of operation, the dominant mechanisms that could result in thruster failure are:

(1) Erosion of the internal surfaces of the discharge chamber by ion sputtering (particularly the ion extraction system screen grid.

(2) Spalling of deposited material that is generated by ion sputtering. This produces "flakes" of conductive material that can bridge insulators or gaps between thruster elements that operate at different potentials, thereby short-circuiting the power supply.

(3) Failure to obtain ignition of thruster discharge or neutralizer hollow cathodes because of depletion of low work function material or heater failure.

(4) Degradation of insulators that have to withstand the high voltage required for acceleration of beam ions (because of deposition of metallic coatings of operation at elevated temperatures that results in impurities "activating" the insulator material).

Designs Against Wearout Mechanisms

Some basic features of the thruster design that protect against these wearout or failure mechanisms are:

(1) Operation of the discharge chamber with low discharge voltage to reduce multiple-charged ion production and sputtering rates.

(2) Covering of the portion of the internal surface of the discharge chambaer with low sputter yield material such as tantalum (where ion impact is predominant).

(3) Covering of portions of the internal surface of the discharge chamber with wire mesh to inhibit spalling of formations of large flakes (in locations where there is little or no ion sputtering).

(4) Development of a hollow cathode that operates at low enough temperatures to reduce loss of the barium aluminate impregnant.

(5) Development of heater design and fabrication criteria that produce highly reliable heaters.

(6) Shielding of sensitive ceramic insulators (fig. 11) to prevent deposition of conductive coatings.

(7) Development of ceramic insulator materials and fabrication procedures that can operate for long periods of time at elevated temperatures without degradation of the insulating properties.

Status of 30-cm Thruster Development

During the last 5 years, NASA-Lewis Research Center and Hughes Aircraft Company have been engaged in completing the development of the 30-cm mercury ion thruister and demonstrating the readiness of ion thruster technology for flight application. During this period, seven thrusters have been fabricated, tested, and modified to obtain the current J-seires design. Currently, three new thrusters are being produced, which will contain the latest designed components. A standardized set of test procedures has been used to evaluate the seven thrusters and a substantial data base has been established for comparing thruster-to-thruster characteristics. Results indicate that the goal of 15,000 hours operating lifetime with greater than 70 percent overall thruster efficiency have been met. Endurance testing is now being performed to substantiate the extrapolations that have been made on the basis of a 4000- to 5000-hour test. Separate component test of cathodes have exceeded 32,000 hours.

Documentation Requirements

Development of the 30-cm thruster to achieve thruster performance and lifetime goals, the documentation of thruster properties, and the definition of interface specifications for incorporating ion thrusters into propulsion systems has been completed. Some of the properties of interest that have been documented are:

- Particles and fields testing at Lewis Research Center has identified the distribution of neutral and ionized particles.
- The thruster magnetic moments have been measured.
- Thruster to thruster performance variations have been measured and are being analyzed.
- The simultaneous operation of two and three thrusters have occurred at Lewis Research Center and -XEOS and the interaction between thrusters have been evaluated.
- Variation of performance with time for 4000 to 5000 hours is negligible.
- Component testing has shown that the major thruster subassemblies meet and exceed performance and lifetime requirements.

Several aspects of thruster fabrication have been identified that have to be tightly controlled to insure reliability and reproducibility in the manufacturing of thrusters. Principle areas identified are:

(1) Swaged, coaxial heaters

(2) Porous tungsten vaporizers

(3) Alignment of ion extraction grids

Areas Requiring Manufacturing Attention

The fabrication of the 30-cm ion thruster have revealed several areas requiring extra ordinary attention in manufacturing. The most critical of these is the area of heaters. Specifications (acceptance criteria) for heaters were progressively made more stringent as the 30-cm thruster has approached flight readiness. The yield of heaters in manufacturing has been low.

Heaters

To increase the yield of the heaters requires control of manufacturing techniques that ensures heaters with (1) uniformly compacted (compaction accomplished without damaging the heater element) insulation within the heater sheath and (2) welding of heater element to the outer coaxial sheath without excessive melting of the center heater element. It is imperative that these heaters have high reliability and uniformity.

Porous Tungsten Vaporizers

Another area of concern is the porous tungsten used for the vaporizer. Although the material that has been used has been successful, the uniformity of the porous tungsten properties between different batches requires good control. Improvements in process specifications and requirements with manufacturers of this material are being continued to obtain reproducably uniform material that exhibits the required properties for fabricating vaporizers.

Other Areas

Other areas of technology that require continued attention pertain to techniques for accurate, reproducible measurement of thruster performance such as direct measurements of thrust and determination of the thrust vector. These measurements, coupled with the present acceptance testing procedures, provide the information required for evaluatin of the thruster performance. This information is critical for mission planning.

Conclusion

The 30-cm J-series thruster has been developed to a flight-ready status. The presently-projected lifetimes of 15,000 hours for the thrusters are sufficient for planning missions that could be launched in the near future. The use of ion thrusters will permit larger payloads to be launched.

The extrapolation of life time from the 4000-hour, 2a-amp beam life test not only indicated that the 15,000-hour design goal had been achieved, but that the components of the thruster had lifetimes in excess of 15,000 hours. Component testing has also shown that the thruster subassemblies meet and exceed performance and lifetime requirements. Advanced technology to increase the capabilities of the 30-cm thruster is continuing at Lewis Research Center.

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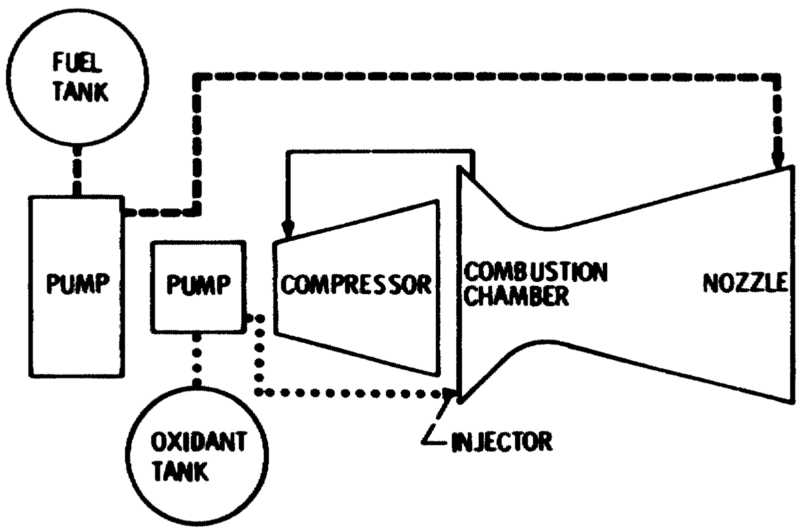


Figure 1. - Schematic of a chemical rocket.

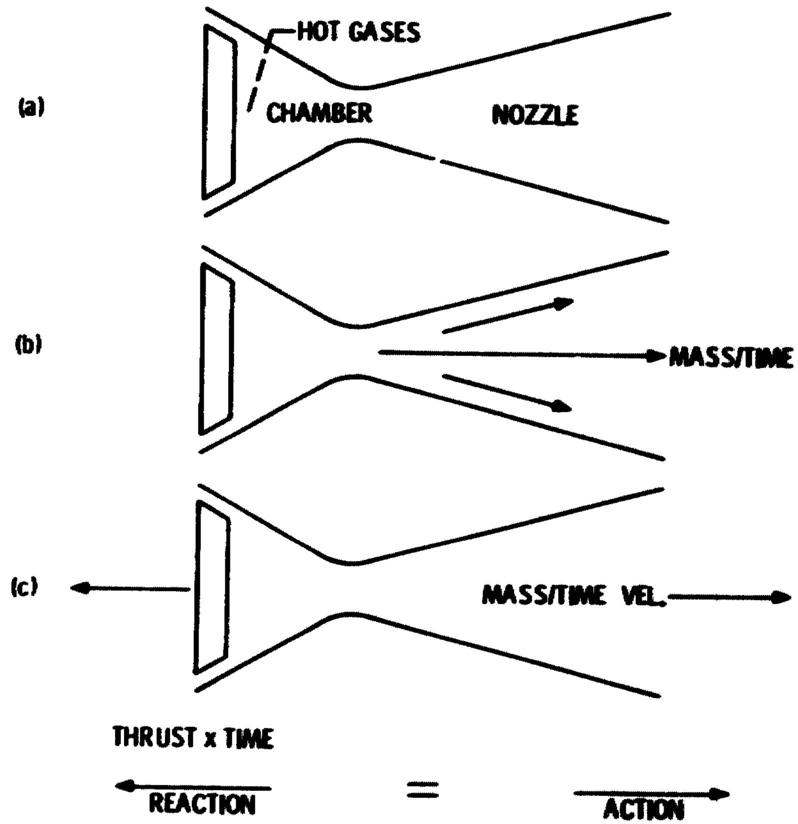


Figure 2. - Schematic of how a chemical rocket works.

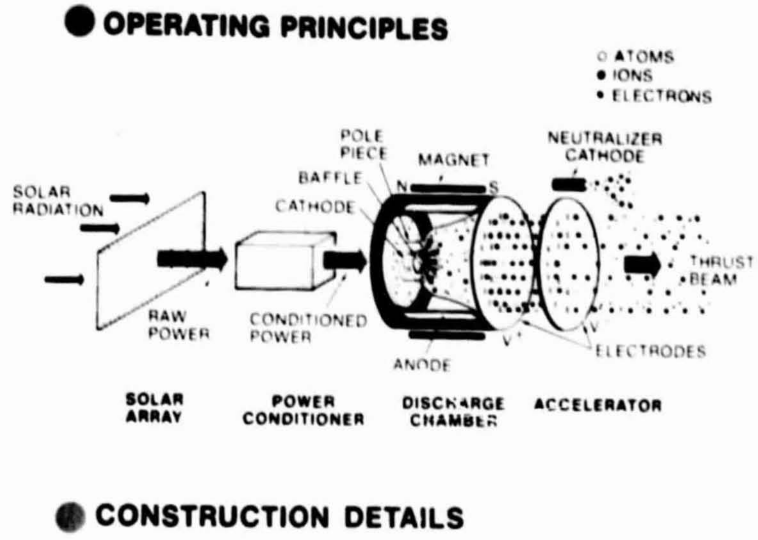


Figure 3. - 30-Cm thruster.

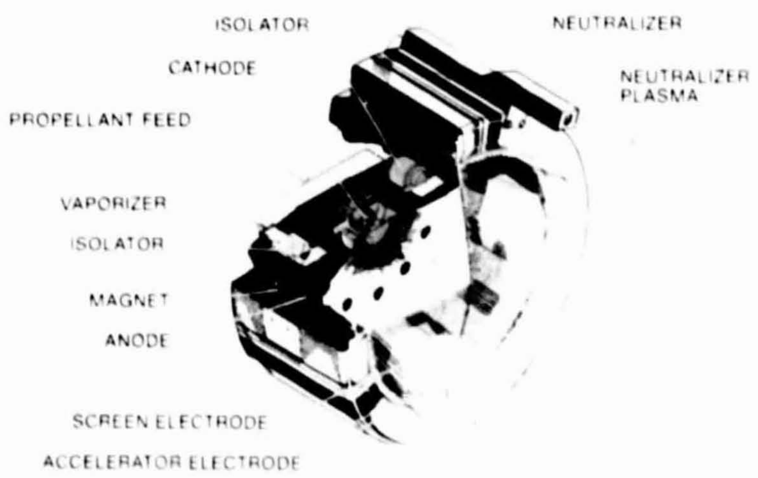
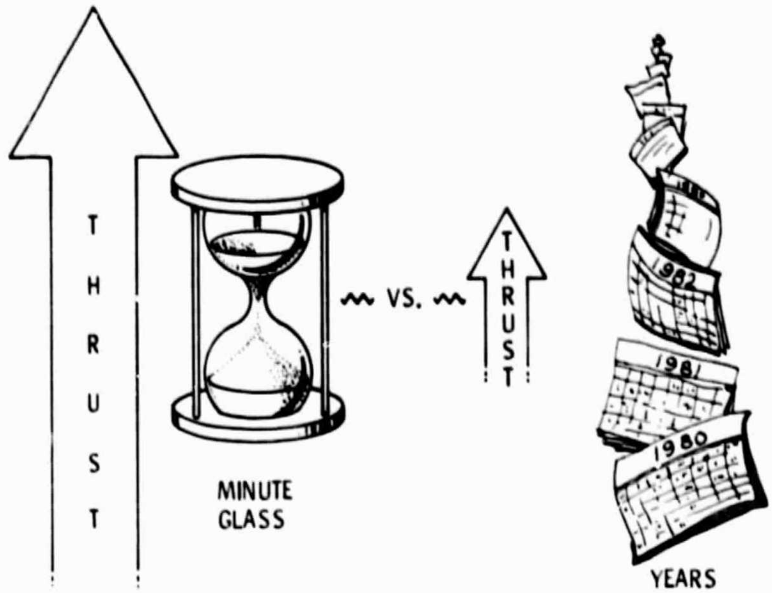


Figure 4. - Construction details of 30-cm thruster.



(a) COMPARISON OF TOTAL IMPULSE - ILLUSTRATIVE.

ELECTRICAL VS CHEMICAL PROPULSION

SOLAR-ELECTRIC

IMPULSE 12.8×10^6 lb-sec	
DRY MASS	1,450 kg
PROPELLANT (Hg)	2,000 kg
TOTAL MASS	3,450 kg

CENTAUR

IMPULSE 13.2×10^6 lb-sec	
DRY MASS	2,000 kg
PROPELLANTS ($H_2 + O_2$)	13,500 kg
TOTAL MASS	15,500 kg

(b) COMPARISON OF TOTAL IMPULSE AND PROPELLANT MASS.

Figure 5. - Performance comparison between chemical rocket and ion thruster.

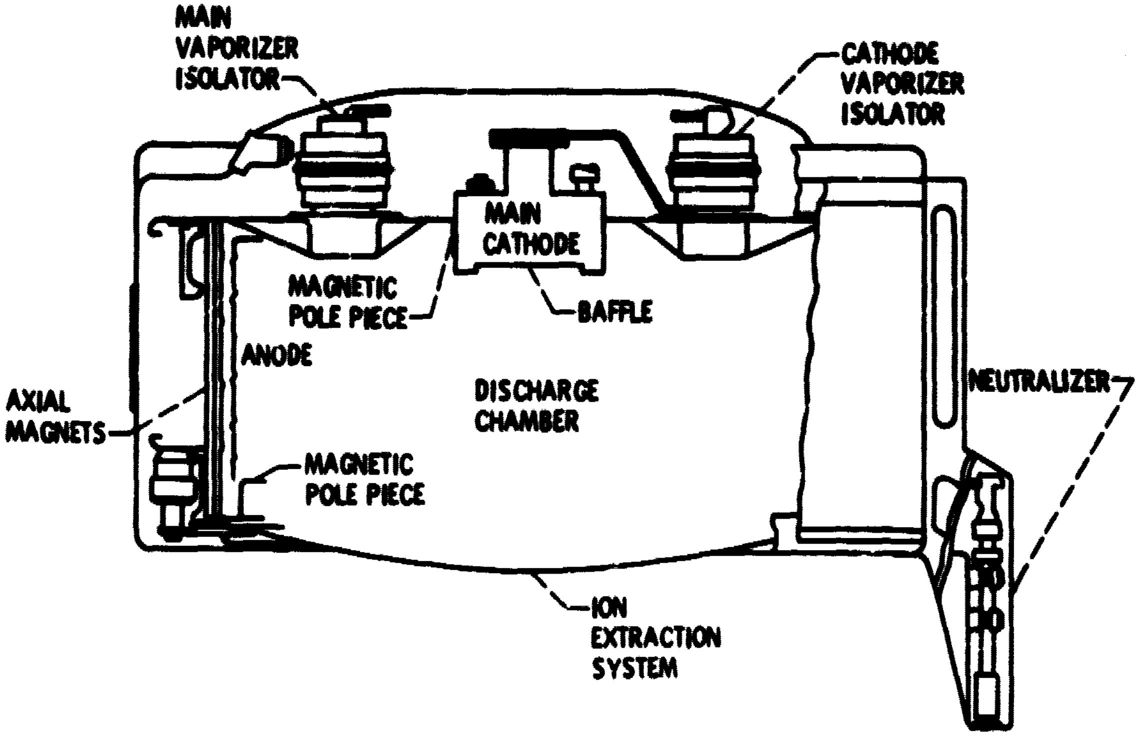


Figure 6. - Thruster section.

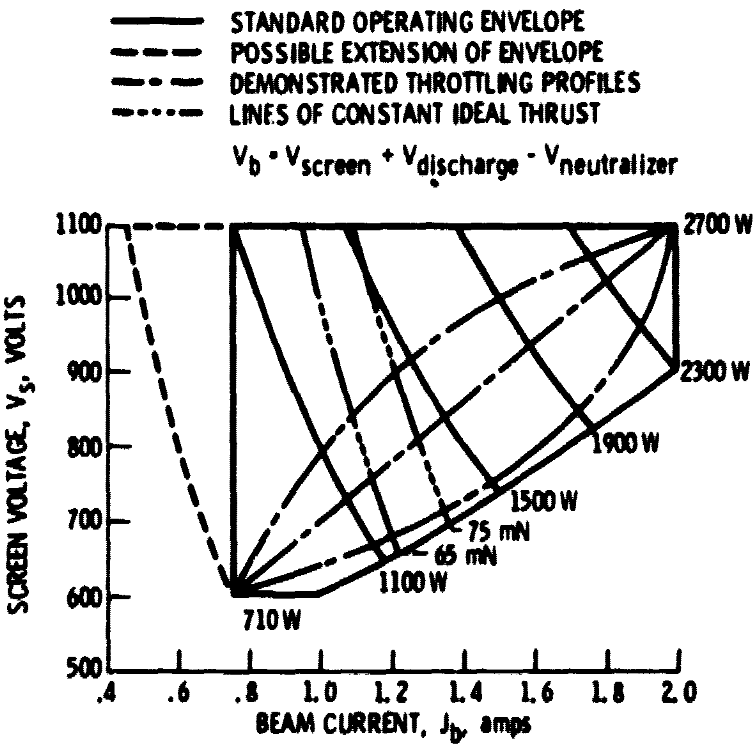


Figure 7. - Thruster operating envelope.

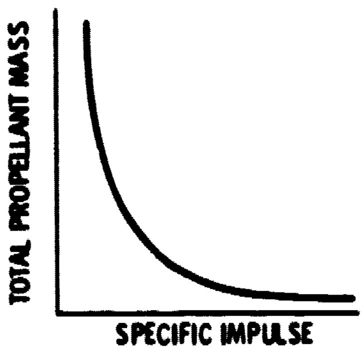


Figure 8. - Total propellant mass as a function specific impulse.

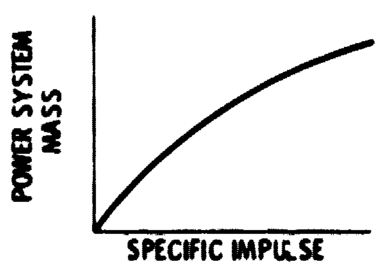


Figure 9. - Power system mass as a function of specific impulse.

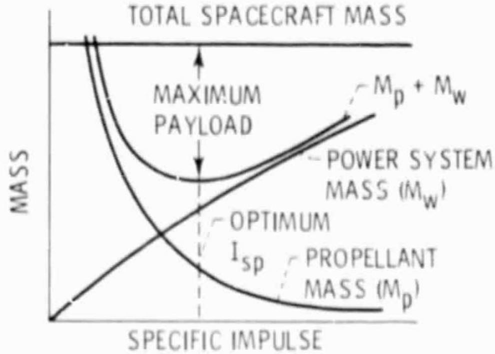


Figure 10. - Optimization of propulsion system mass and specific impulse.

Figure 11. - Ceramic insulator without and with shielding.