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KAUFMAN THRUSTER DEVELOPMENT AT LEWIS RESEARCH CENTER (LeRC)

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Abstract

The current status of research programs on mercury electron-bombardment thrusters is reviewed. Future thruster requirements predicted from mission analysis are briefly discussed to establish the relationship with present programs. Thrusters ranging in size from 5 to 150 cm diameter are described. These thrusters have possible near to far term applications extending from station keeping to primary propulsion. Beam currents range from 10 mA to 25 A at accelerating potentials of 500 to 5000 V.

Introduction

Initial applications of electric propulsion probably will be with systems powered from photovoltaic cells. Specific masses of present day solar cell systems dictate that electrostatic thrusters will be required to operate at specific impulses generally below 4,000 sec (1600 V Hg^+ ion) in order to minimize overall propulsion system specific mass, estimated to range from 30 to 50 kg/kW. Several prospective solar electric missions assume the use of mercury-electron bombardment thrusters.[1-4] The references in this area are not complete but are only intended to be representative of trends in mission studies reported in the past year or so. The studies detail the requirements for thruster operation at less than 4,000 sec specific impulse. Because of inherent ion production losses and some unavoidable fixed losses, the efficiency of a bombardment thruster decreases with specific impulse. Continued research is being conducted to ensure optimum performance at the required specific impulse.

General Thruster Description

Kaufman thrusters have been tested over a size (discharge chamber diameter) range of 2.5 to 150 cm. Fig. 1 shows a typical thruster system used in the 1970 orbital flight test (SERT II).[5] Liquid mercury propellant is pressure fed to a heated, porous-tungsten plug which vaporizes the mercury. The vapor flows into the discharge chamber where it is electron-bombardment ionized by a 2 A, 40 V discharge. A relatively weak (35×10^{-4} T) magnetic field helps contain the discharge electrons and reduces discharge losses. Mercury ions diffuse to a parallel plate grid system which extracts, focuses and accelerates them into an exhaust beam. A neutralizer cathode emits electrons into the beam to maintain overall system neutrality. Depending on thruster size and accelerator grid design, single charged mercury ion beams from 10 mA to 25 A are produced at energies between 500 and 10,000 eV.

A heavy ion, such as mercury, was chosen as a propellant because it produces a maximum thrust per unit area of accelerator grid and a low value of power to thrust ratio. Maximum thrust is limited by the electric field gradiant in the accelerator grid system. Minimum power is achieved by ionization of relatively few heavy ions. Mercury was chosen over cesium primarily for simplicity of propellant feed systems. Heavy molecules or colloids were not chosen because they fragment when ionized by electron bombardment.[6]

The discharge chamber has been developed primarily by experimental testing.[7,8] Permanent magnets may be replaced by electromagnets if a variable field is desired. The shape of the magnetic field is produced by soft iron pole pieces around the cathode and near the screen grid. The shape and strength of the magnetic field was experimentally varied until low values of discharge power and loss of ionized propellants were achieved. The discharge chamber has been probed and found to contain maxwelliam electrons at several volts energy plus a small fraction (less than 0.1) of primary energy electrons.[8] The electron density is of the order 10^{11} per cm^3. Spectroscopic observations indicate less than 10 percent (by volume) of the propellant atoms are ionized.[9] Because of relatively high ion temperatures and small favorable electric field gradients, only 5 to 15 percent of the total propellant escapes through the grid opening as unionized mercury. The plasma potential is near that of the anode, with most of the discharge drop occurring at a cathode sheath or across a discharge baffle. The neutral density in the discharge chamber is estimated to be 10^{12} per cm^3.

The discharge cathode and the neutralizer cathode are similar and are called hollow cathodes. Figure 2 shows a cross section of a typical hollow cathode. A tantalum tube is capped by a thoriated-tungsten disc containing a small orifice. A coiled tantalum foil insert containing barium oxide is placed inside the tube and assures a small thermionic electron emission to start the hollow cathode discharge. A starting-keeper electrode produces a local electric field for starting and maintaining a discharge. Mercury vapor at an estimated pressure of 3000 to 6000 N/m^2 (25 to 50 torr) flows through the tube, and a discharge to the starting-keeper electrode occurs when sufficient density has been reached. The cathode flow of mercury enters the main discharge chamber where it mixes with the main propellant flow. The neutralizer cathode flow escapes into the vacuum downstream of a thruster and is lost for producing thrust.

The accelerator grid system consists of two parallel molybdenum plates with matching holes.

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The first grid (screen) is at the common high positive potential of the thruster body. The second grid (accelerator) is at a negative potential for two reasons. One is to prevent downstream electrons from backstreaming through the accelerator grid system. The other is to provide a higher total accelerating voltage which permits a larger, space charge limited, beam current to be extracted. The entire thruster (except the accelerator grid) is enclosed in a ground screen which prevents electron-backstreaming to the positive potential thruster.

SERT II Flight

The Space Electric Rocket Test II was launched in February 1970 for the purpose of demonstrating long-life space operation of either one of two 15-cm diameter ion thrusters. A cut-away flight model is shown in Fig. 1. The results of the space testing[5] and correlary ground tests indicated excellent agreement of thruster performance in space compared to the same thruster pretested before launch. One flight thruster operated for 5 months, the other for 3 months. The nominal operation of the solar-cell powered thruster was: 0.25 A beam current; 4150 sec specific impulse; total power input, 0.85 kW; overall thruster efficiency, 0.68; thrust, 28 mN (6.3 mlb).

Each thruster operated in space with less degradation than corresponding ground tests until a sudden permanent electrical shorting occurred between high positive and negative voltages. All cathodes and feed systems continued to relight and function normally.

After study of post flight failure data, ground endurance test observations, and a number of possible causes, it was concluded that both flight thrusters failed (shorted) for the same reason. Localized wear of the accelerator grid, due to neutralizer ions, caused a few small, metal grid fragments to detach. During ground tests such fragments could fall (by gravity) away from the thruster, but in space a fragment would be drawn by electrostatic force and short between the closed space thruster grids.

There are two possible solutions to the SERT II thruster problem. One is to move the neutralizer to a new location which would reduce the localized accelerator grid wear to a level at which fragments would not form. The other is to provide a high current source to "burn out" such fragments if they should form. This capability, unfortunately, was not incorporated in the SERT II power conditioner. The remaining parts of the accelerator grid are capable of useful life to 10,000 hr or more in the absence of the fragment problem.

30-cm Thruster

Future multi-kilowatt electric propulsion spacecraft, such as the Solar Electric Multiple Missions Spacecraft (SEMMS), requiring on the order of 10 kW power will probably not use the SERT II size (15-cm) thruster system. Instead a larger diameter thruster, such as 30-cm will be used to reduce the total number of thrusters required.

A 30-cm diameter thruster (Fig. 3) has been experimentally tested both at Lewis Research Center and Hughes Research Laboratory the past several years.[7,8] These programs defined critical design problems with increasing thruster size, optimized the discharge chamber geometry and magnetic field shape, and experimentally documented higher propellant utilization efficiencies. The following paragraphs discuss the present critical design areas and current research programs.

Accelerator Grid System

The design beam current, 2.0 A, is eight times that of SERT II, while the accelerator area is only increased four times. The accelerator grid is the present limiting thruster component for maximum beam current and low specific impulse. More beam current and lower specific impulse may be obtained by a two-grid system by decreasing the grid spacing and using smaller holes. Decreasing grid spacing is limited by thermal buckling of the grid system and electrostatic attraction of the two grids. The use of small holes will eventually result in web-thickness-between-holes being too thin compared to tolerances of fabrication. Present programs at Lewis Research Center are emphasizing the mechanical design and support of the grids to minimize the effects of thermal buckling.

Three approaches are under consideration. The first approach is to design a radius of curvature (dish-shape) into the previously flat grids. Such curvature reduces the total movement due to thermal expansion, controls the direction of expansion and because both grids are the same approximate temperature, results in low relative change in grid separation. Grid systems with radius of curvature of 67 cm have been experimentally tested for short (20 hr) periods and found to be capable of extracting a 2.0 A beam at a total accelerating voltage of 1300 V. Grid separations of 0.06 cm (cold) have been used, but the design of a grid mounting assembly to reliably hold this spacing on a flight thruster for operating times of the order of 10,000 hr has not been achieved.

A second approach is to design flat grids with enough radial tension to exceed compressive forces (thermal buckling) produced by radial temperature gradients in the grids. Such an approach has been previously suggested.[11] Grids are presently being made that use a heavy circumferential ring of stainless steel or titanium attached to the molybdenum grid. As the grid system heats to operating temperature (300° to 400° C), the ring having a higher thermal expansion coefficient expands faster and stretches the grid flat.

A third approach is to use multiple points of support. Grids have shown adequate support if the aspect ratio (distance between supports divided by grid separation) is kept below the range of 50 to 75. For the grid separation of 0.06 cm, the maximum distance between supports may be only 3 to 4 cm.

In the first two approaches a center support may be necessary, if for no other reason, to limit electrostatic movement of the grids. An alternate approach to center supports is the use of support rods between the screen grid and thruster back plate for a 20-cm diameter thruster.[12] Other methods besides closer-spaced grids are also being pursued to increase the maximum beam current. These are: (1) reduce the radial variation in the discharge-chamber plasma density to more effectively use the outer diameter areas of the grid system; (2) increase the percent open area of the screen grid; and (3) use thinner screen grids. The results of these research areas will be incorporated into flight prototype thrusters when they become proven.

The present state-of-the-art fabrication for flight thrusters is represented by a 30-cm thruster tested for 450 hr (December 1970) at the Hughes Research Laboratory. (The promising dished-grid results noted previously, are not yet developed for flight use.) Using the 450-hr test grid data at a beam current of 1.87 A, an extrapolation to 2.0 A with Child's Law scaling, gives a total accelerating voltage of 2080 V. Use of a minimum value of 0.5 for net-to-total voltage ratio, limits the specific impulse to values greater than about 2940 seconds for the 2.0 a beam current. The 0.5 value was assumed to prevent excessive thrust loss due to beam divergence and possible direct beam erosion of the grid. Two-grid lifetime due to sputtering by charge exchange ions will be in excess of 20,000 hr. The adverse effects of thinner grid material and high beam current density are more than offset by lower grid voltages and high propellant utilization efficiencies. A major lifetest (6000 hr) is contracted to begin in early 1972 at the Hughes Research Laboratory. A two-grid, 30-cm thruster will be tested in a vacuum facility containing a frozen mercury beam target.

Main Cathode

Accumulated short term endurance tests (up to a total of 1200 hr for the same cathode) have indicated acceptable or negligible wear rates for 30 cm type cathodes.[10] These low wear rates result from keeping the hollow cathode tip temperature low. Good designs include larger tip diameter and larger hollow cathode orifice diameter. The most promising design to date contains a chamfered or diverging nozzle shape in the orifice.[8]

Neutralizer

Hollow cathode neutralizers have been successfully used in 30-cm tests but areas still exist for improvement and are the subject of present research programs. One area is the position and angle (to the beam) of the neutralizer cathode to reduce the localized sputtering of the accelerator grid by neutralizer ions. Preliminary investigations of position and angle variables have indicated an order of magnitude reduction in local neutralizer ion impingement.[13] Another improvement area is reduction of required neutralizer flow. Bell jar neutralizer tests have demonstrated up to 12,000 hr durations.

Throttling and Control

The 30-cm thruster has been throttled over a 2 to 1 range of beam current and experimentally controlled well enough to permit unattended endurance operation over weekends.[7] But the loss of thruster efficiency has been somewhat excessive (a drop from 0.60 to 0.45 with 2:1 throttle range) and the full range of possible flight control has not been verified. Present research programs are studying main thruster control loops and ways to improve throttled thruster efficiency. One possibility is a variable magnetic baffle, such as used in Ref. 14. Recently a control system using a magnetic baffle has been developed at the Hughes Research Laboratory that throttles a 30-cm thruster over a 5 to 1 range by varying a single input to the power conditioning system. The loss of thruster efficiency was excessive, however, when the throttling range exceeded 3 to 1. Additional information regarding throttling loss predictions can be found in the section Supporting Research.

Near-Term Flight Thruster Performance

If the best of the endurance-proven research data is taken as a starting point for a 2-year development program, and if minor thruster development improvements offset unplanned minor development losses, the following predictions of 30-cm thruster performance could be met for near-term flights. (The 2-year development program is assumed to produce a prototype test model (PTM). It should be a flight qualified thruster that has been array tested and is ready to go into the final qualification program of a flight spacecraft.)

Figure 4 is a predicted performance plot of a 30-cm PTM thruster efficiency versus effective specific impulse for a 3 to 1 throttling range of beam current. The following assumptions have been used in preparing Fig. 4. (1) Current state-of-the-art grid designs limit the effective specific impulse to the solid portion of the curves. (The dashed portion is beyond present technology, but perhaps feasible in the future with closer-spaced accelerator systems. Note that the range of usable effective impulse opens up at lower beam currents where lower accelerating voltages are possible.) (2) The chamber propellant utilization (not including neutralizer flow) is 0.93 at maximum beam current of 2.0 A and is reduced to 0.90 and 0.81 at beam current levels of 1.33 A and 0.67 A respectively. (3) The neutralizer flow is constant at 50 mA equivalent flow. (4) A constant value of 220 eV/ion discharge loss for all beam currents is used. (5) A neutralizer coupling loss of 20 V and a 0.6 percent accelerator grid impingement is assumed. (6) The fixed power losses (heaters, keepers, etc.) are 42 W.

The suggested design point is 2.0 A beam current at an effective specific impulse of 3000 sec. The thruster efficiency is 0.73 at this point. The best manner to throttle to lower power levels is undecided at this time, but will probably include a significant throttling in beam current level and may include some decrease in effective specific impulse. The performance of the SERT II flight thruster is compared with the 30-cm 1973 PTM thruster and with a far term (1980) thruster in Table I.

The grid lifetime of the PTM thruster should not be limited by ion erosion at maximum beam current. The grid spacing, however, does limit the maximum beam current and the effective specific impulse. The design PTM thruster grid lifetime is well over 10,000 hr. Based on a relatively few short time cathode tests, the cathode lifetimes should be similarly long. The problems of the development program should not be those associated with lifetime, but rather those of maintaining thruster efficiency and high beam currents without grid warping or control stability problems occurring.

5-cm Thruster

Spacecraft with design lifetimes of several years place severe requirements on attitude control and station-keeping subsystems. The long life and high specific impulse available from low thrust, electrostatic thruster subsystems makes them increasingly competitive for these functions. Thrust levels in the millinewton range, for example, are of particular interest for spacecraft in the 500-1000 kg class.[15]

The 5-cm thruster program at LeRC is aimed at both providing an efficient, lightweight and durable thruster for the above applications and serving as a test article upon which new component concepts can be demonstrated. The inhouse work is coordinated with a contract effort being conducted at Hughes Research Laboratory. The physical characteristics of the 5-cm thruster are given in table II. Photographs of the Hughes-developed subsystem are shown in Fig. 5.

The total discharge chamber propellant flow is fed from the single gas pressurized tank through the cathode. The thruster potentials are separated from the grounded tank by a vapor phase electrical isolator. The neutralizer flow goes direct from the tank to the neutralizer.

Table III lists the performance characteristics of the 5-cm thruster. The best experiment data taken to date with a complete thruster subsystem has been obtained at the Hughes Research Laboratories under Contract NAS3-14129. These data are presented in the first column of table III. Similar component efficiencies have been obtained at LeRC in separate tests. The second column of the table is a reasonable goal for any continued effort on this thruster and was generated by combining the best experimental component performances as the total subsystem goal. The final column of the table is the projected possible performance level achievable by a 5-cm thruster after extended development. Lifetimes greater than 10,000 hr are consistent with all of these performance parameters.

In 1970 the results of a 5-cm thruster performance improvement program were reported.[16] In that study it was found that cathode pole piece and baffle position and geometry significantly influenced ion chamber performance and could be used to tailor the discharge characteristics to obtain efficient operational modes (e.g. to change current-voltage characteristics). Enclosed hollow cathodes were chosen for both chamber and neutralizer emitters based on discharge stability, operational range, durability and structural design. Recent results of 5-cm thruster component tests indicate that acceptable lifetime cycling (2800 on-off cycles) can be achieved for neutralizer cathodes.[17] Also, neutralizer positioning can be accomplished which is consistent with low coupling voltages, and long accelerator grid lifetime, and does not result in direct erosion from beam ions.

Thrust Deflection

Thrust deflection can reduce the number of thrusters required on a spacecraft by combining the function of station keeping and attitude control into a single thruster. Several thrust deflection methods have been demonstrated both at Hughes Research Laboratories under Contract NAS3-14058, at LeRC and Electro Optical Systems, Inc.[18,19]

A thermomechanical system, in which the screen grid is moved with respect to the accelerator by heating opposite pairs of actuating springs is shown in Fig. 6. The relative movement of the screen and accelerator causes misalinement of the optics and the beam deflects toward the nearer wall of each accelerator hole. Beam deflections of $\pm 15^{\circ}$ have been made in two orthogonal directions without significant increase in accelerator impingement.

A more elegant solution to the beam deflection problem is shown in Fig. 7. This figure is a photograph of a two-axis electrostatic deflecting grid. The beam from each screen hole is accelerated and focused by orthogonal sets of accelerator ribbons. Differential potentials applied across pairs of ribbons causes beam deflection in the same manner as the electrostatic scan systems on cathode ray tubes. Beam deflection of $\pm 9^{\circ}$ have been demonstrated without significant increase in accelerator erosion. A 100-hr test with beam deflection has been conducted under contract.[20] A 1000 hr test has been completed at LeRC.[21] Continuing life tests at LeRC are in progress and are aimed at demonstrating the suitability of this beam deflection concept to mission applications.

150-cm Thruster

The 150-cm thruster is designed for power levels in excess of 100 kW, which places its potential applications in the category of primary propulsion for large space vehicles, probably using nuclear-electric power conversion systems. Present investigations of thrusters of this size are exploratory in nature, aimed primarily at seeking general information. This thruster is included herein mainly for completeness since no additional data has been obtained beyond that reported in Ref. 22.

A cutaway view of the 150-cm thruster is shown in Fig. 8. Propellant flows from a distributor manifold into the ion chamber via perforated radial channels located on the chamber rear wall. Ten cathodes are equally spaced on the rear wall and their radial positions may be varied. The ion chamber L/D (length to diameter ratio) is 0.15 and a conventional two-grid accelerator system is used with the exception that it is slightly dished and spacers are used to help maintain uniform grid spacing. Performance highlights are listed in table IV.

Supporting Research

The energy required in the discharge chamber to produce a beam ion is about 200 eV/ion in present day optimized thruster. High percentage open area grids and divergent magnetic fields are the two major factors contributing to progress in reducing discharge losses. No doubt additional gains will be made in the future, although an increasing degree of sophistication may be required. As an example, one program recently investigated offers additional insight into present limitations in the discharge chamber.

In this program chamber performance parameters were monitored to determine possible limitations on the propellant utilization of the mercury electron bombardment thruster.[23] The results of this analytical and experimental study show that the loss rate of un-ionized propellant at maximum utilization is nearly a constant over a wide range of propellant flow rate. This constant loss rate strongly effects thruster performance during throttling.

Spectroscopic diagnostic techniques have been perfected for viewing the discharge chamber and the exhaust ion beam.[9] These techniques quantatively measure relative density of ions (singly or doubly ionized) and neutrals as well as electron temperature and distribution. One immediate application of such work is estimating[24] any interference of a spacecraft star tracker signal from radiation produced by the ion exhaust beam.[24]

To permit correct design of flight discharge power supplies and to furnish a basic understanding the discharge type, volt-ampere curves have been taken by R. C. Finke using battery banks and pure resistive loading. Data as shown in Fig. 9 indicate a classical form of dark emission, glow and abnormal glow and finally arc discharge. Many of these discharge details are lost if a commercial power supply is used due to the high reactive impedance associated with these supplies.[25]

In another program a 15-cm SERT II thruster was operated on various gases.[26] Xenon, krypton, argon, neon, nitrogen, carbon dioxide and helium were tested. These materials are less efficient than mercury for propulsion but have possible ground based applications. Changing the propellant atomic mass while holding geometric and electrical parameters fixed allows a broad view of thruster operation. The correlation of thruster data with a wide range of atomic characteristics has resulted in greater design confidence for mercury thrusters.

Program Summary

The status of the various thruster programs can be conveniently summarized and discussed within the framework of a thruster efficiency vs specific impulse plot as shown in Fig. 10. Shown for comparison are data points from the thrusters and an "ideal" curve in which all system losses (including un-ionized propellant) are assumed equal to 200 eV/beam ion. The open symbols represent data points while the solid symbols represent design goals.

The 150-cm data point falls significantly below the ideal curve. Problems specifically associated with large thrusters have been investigated and there appear to be no fundamental limits on this size thruster. A decreased emphasis will probably continue until a more specific requirement for this power level thruster becomes apparent.

The increased emphasis placed on the 5-cm thruster over the past few years has resulted in an auxiliary propulsion subsystem with a highly desirable demonstrated power-to-thrust ratio of 31 W/mN (138 W/mlb). Values under 22 W/mN (100 W/mlb) are expected with improved subsystem. The information gained in the 5-cm component and thruster subsystem optimization program with small, relatively inexpensive pieces of hardware has had a significant benefit on the efficient design and rapid optimization of larger components and thrusters. The demonstration of acceptable lifetime for space application is in progress for the 5-cm thruster subsystem with electrostatic beam deflection capability.

Present 30-cm research thrusters using conventional grids have shown single point performance, ability to operate over a 3:1 throttling range, and good lifetime of all components. Such thrusters, developed through a 2-year program would be capable of performing a variety of near-term space missions.

In the context of the needs of many anticipated missions, mercury electron bombardment thrusters capable of fulfilling a wide variety of applications are already available. Present developments aimed at specific applications should result in increased performance for these applications perhaps without any loss of flexibility.

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TABLE I. Thruster Comparison

	15-cm SERT II flight	30-cm 1973 PTM*	30-cm 1980 thruster
Input power, kW	0.85	2.7	2.3
Net voltage, V	2900	1090	720
Beam current, A	.25	2.0	2.5
Utilization efficiency	.77	0.91	0.94
Thruster efficiency	.68	.73	.74
Thrust, mN	28	135	135
Thrust, mlb	6.2	30	30
Effective spec. imp., sec	4152	3000	2500

*Prototype test model

TABLE II. 5-cm Thruster Subsystem Specifications

7.5-cm diameter, 30-cm long
2.1 kg empty weight
6.2 kg propellant weight
Synchronous orbit thermal design

TABLE III. 5-cm Thruster Performance Values

	Data	Goal	Possible
Input power, W	56.4	59.5	49.5
Net voltage, V	650	1000	750
Beam current, mA	35	35	40
Utilization efficiency	0.73	0.78	0.80
Overall efficiency	.29	.46	.48
Thrust, mN	1.8	2.2	2.2
Thrust, mlb	.41	.5	.5
Effective spec. imp., sec	1840	2460	2180
Power to thrust ratio, W/mlb	138	119	99

TABLE IV. 150-cm Thruster Performance Values

Input power, kW	177
Net voltage, V	6000
Beam current, A	25
Utilization efficiency	0.90
Overall efficiency	.76
Thrust, N	4.0 (0.89 lb)
Effective spec. imp., sec	7000

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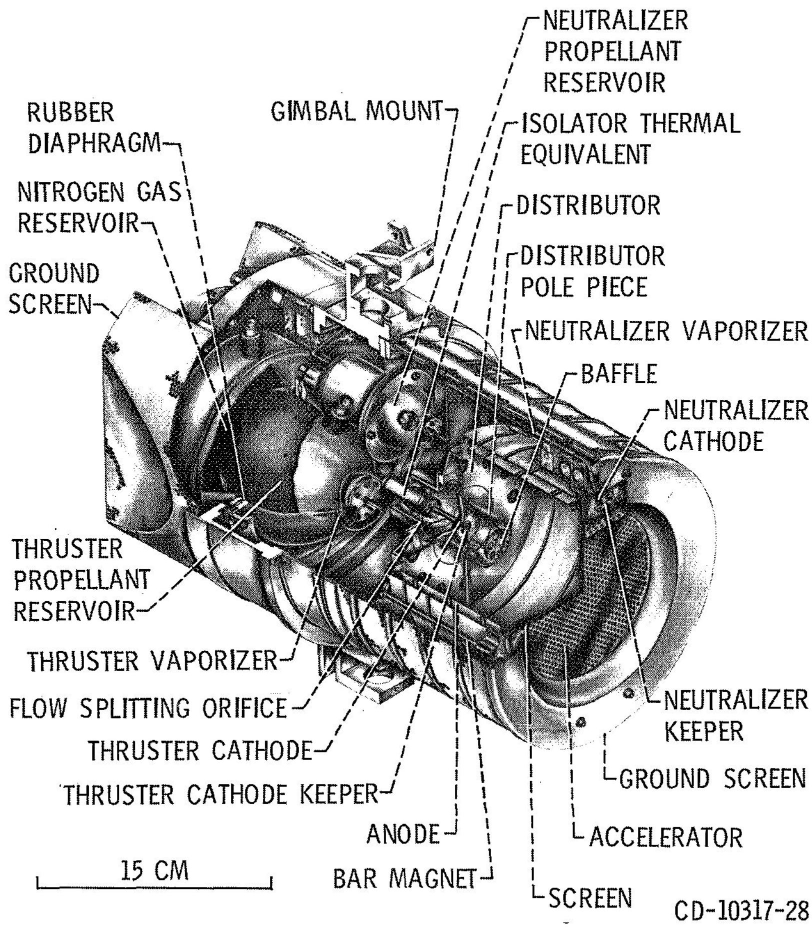

Figure 1. - Cutaway sketch of 15 cm SERT II thruster.

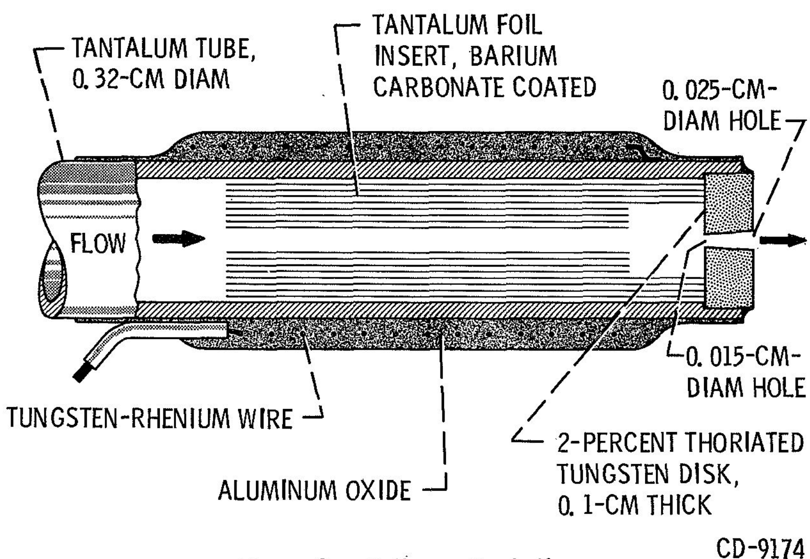

Figure 2. - Hollow-cathode tip.

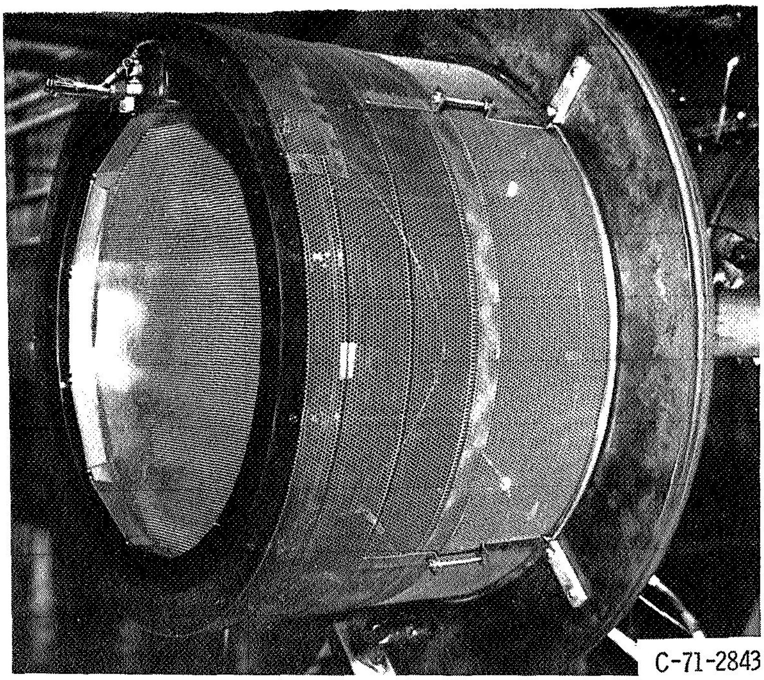

Figure 3. - 30-cm dished two-grid thruster.

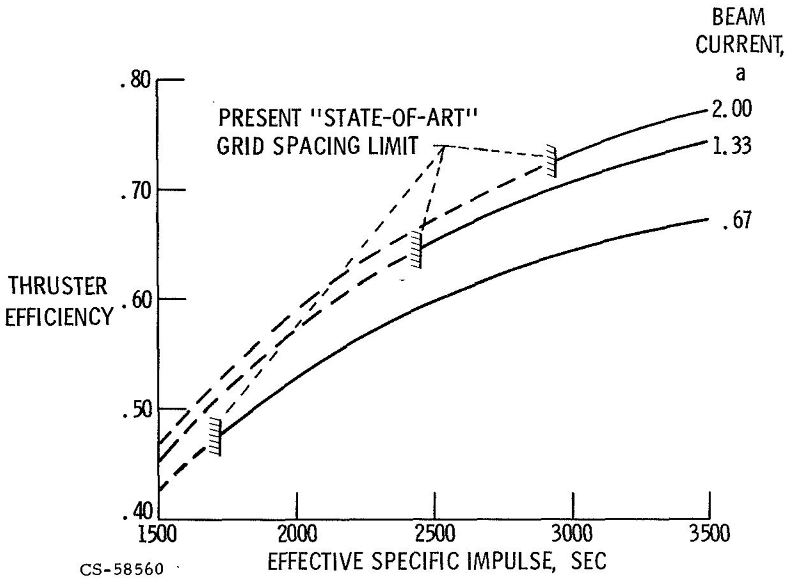

Figure 4. - 30-cm thruster efficiency (1973 PTM) versus specific impulse.

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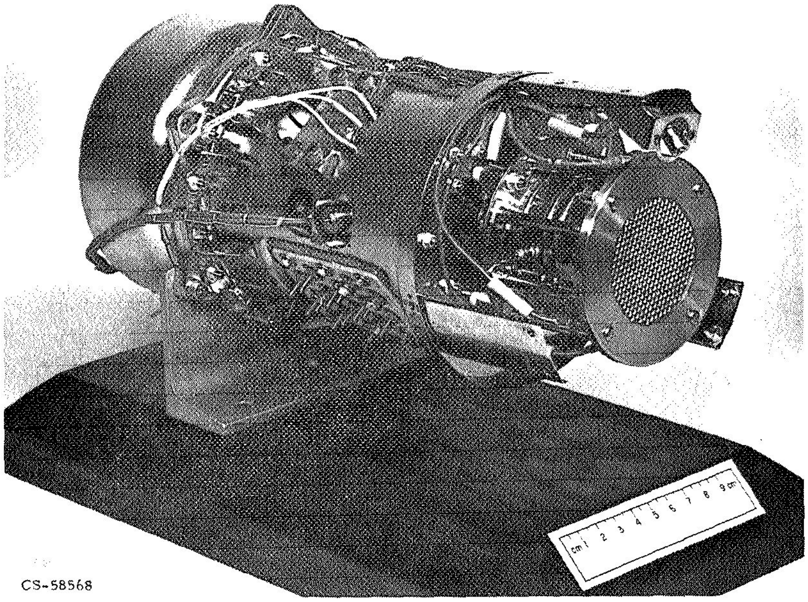

Hughes prototype.

Figure 5. - 5-cm thruster.

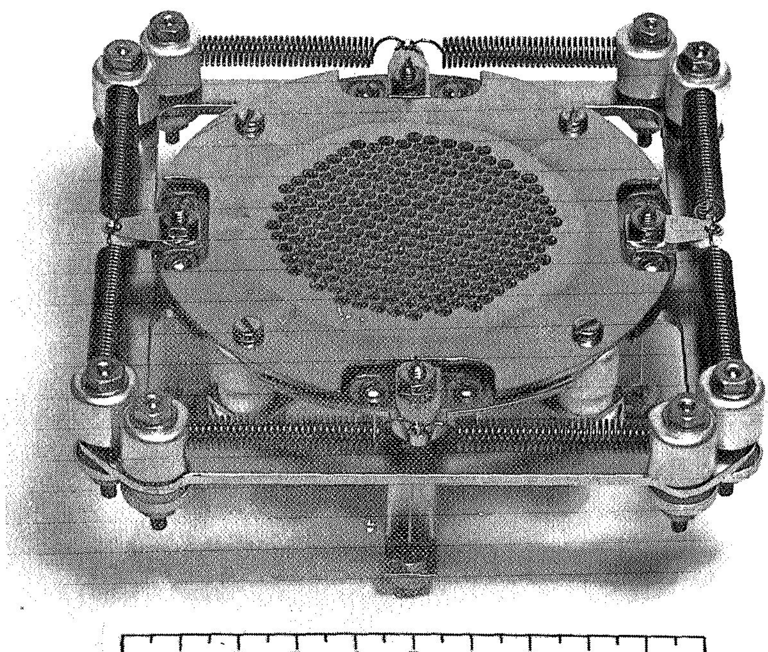

Figure 6. - Thermomechanical 5-cm vectorable grid. (Contract NAS 3-14058, Hughes Research Lab.)

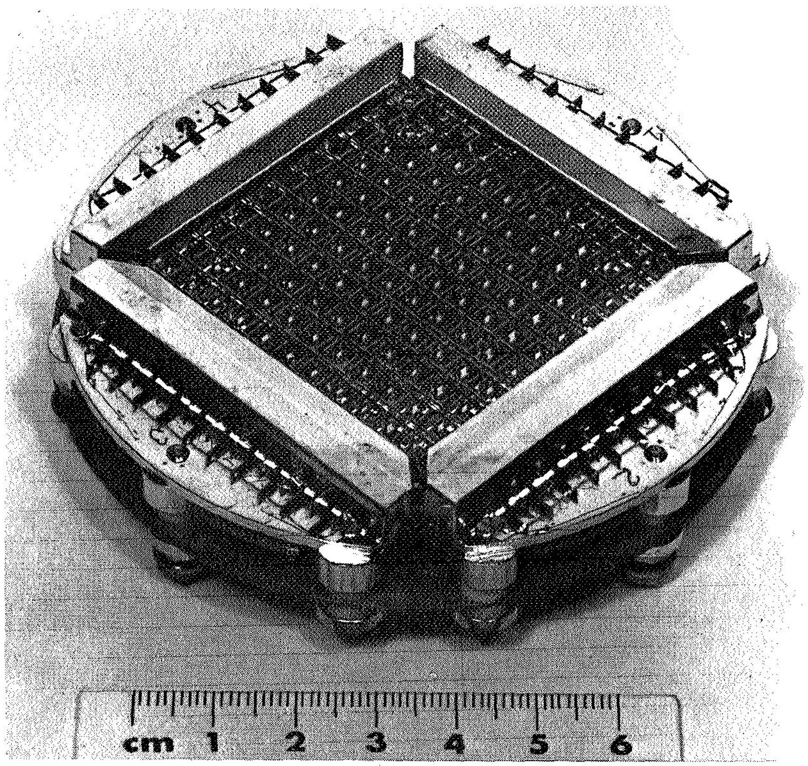

Figure 7. - Electrostatic deflection 5-cm vectorable grid. (Contract NAS 3-14058, Hughes Research Lab.)

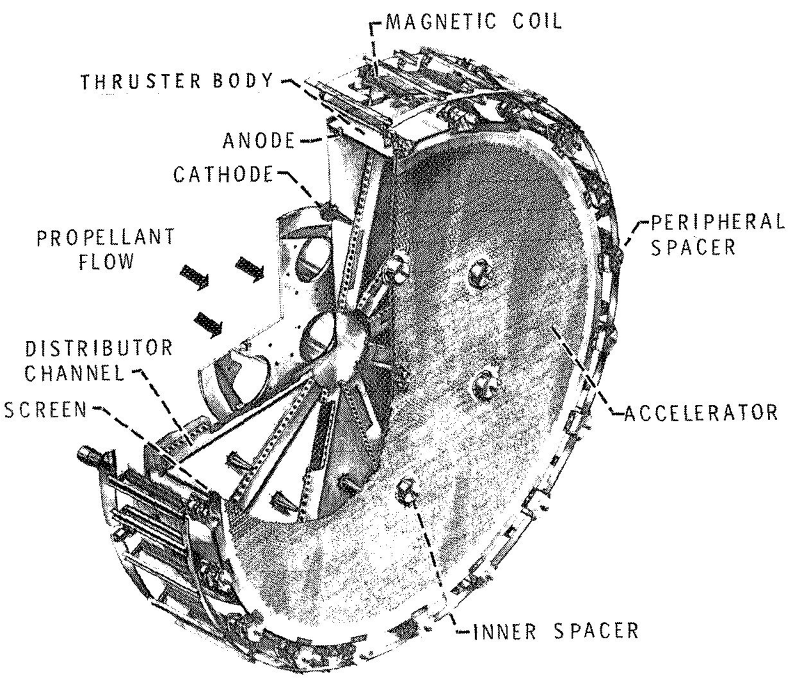

Figure 8. - Cutaway view of 1.5 meter diameter Kaufman thruster.

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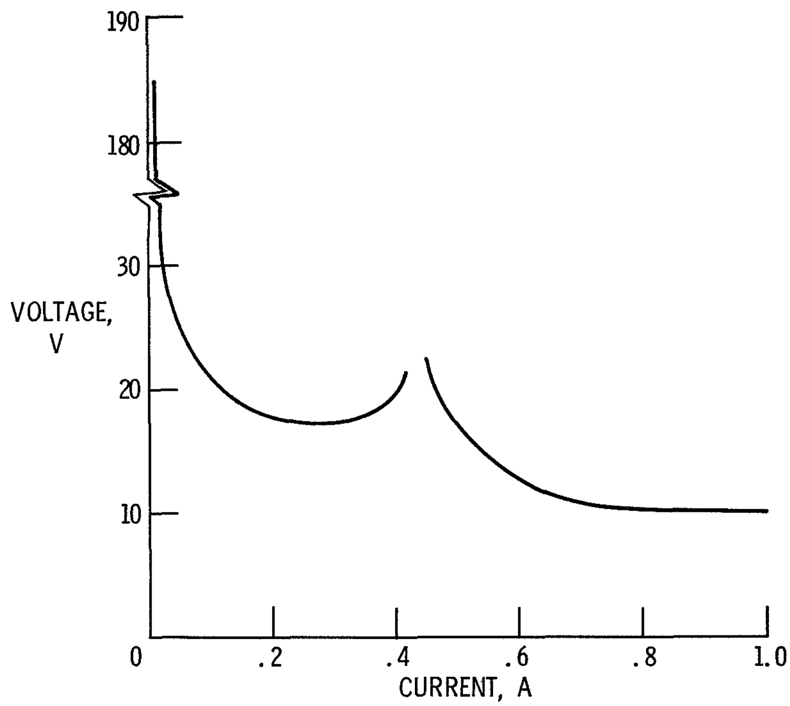

Figure 9. - Characteristic of hollow cathode discharge. Hg flow, 40 mA.

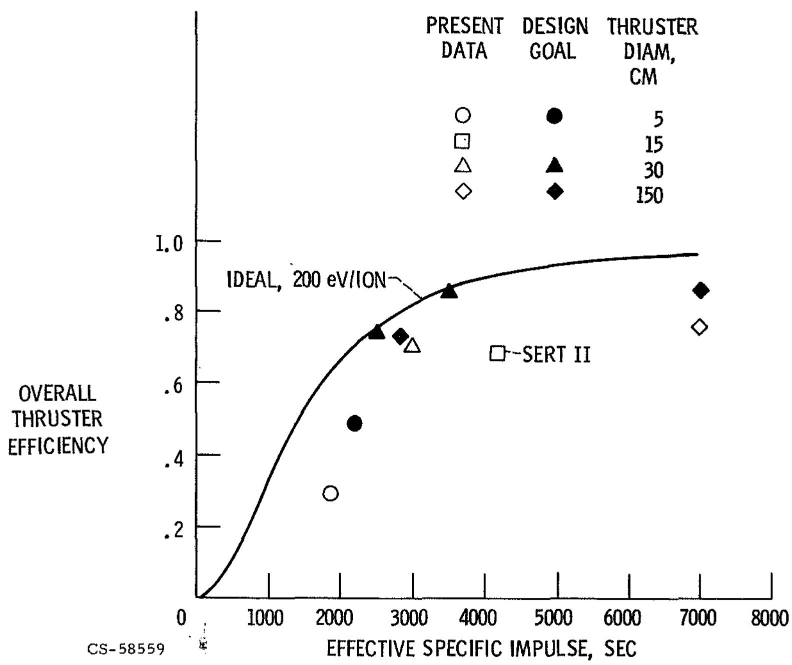

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Figure 10. - Comparison of design goals with present data of various size mercury bombardment thrusters at maximum beam currents.