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CURRENT TECHNOLOGY IN ION AND ELECTROTHERMAL PROPULSION

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Abstract

As Earth-orbital and interplanetary spacecraft missions become longer and more complex, the demands placed on spacecraft propulsion systems become much more stringent. In several instances, combinations of mission and spacecraft requirements such as weight, size, life, and total impulse have precluded the use of the more conventional chemical and cold gas propulsive systems. As a result, high-performance propulsion devices, such as electrostatic ion engines and electrothermal thrusters, are achieving wide user acceptance. This paper surveys the current technology and projected development trends in the areas of ion and electrothermal propulsion systems and components.

Introduction

The use of electrostatic thrusters for primary propulsion has demonstrated a potential for exceeding the performance of all chemical propulsion vehicles for many high-energy missions. Future missions made feasible by solar electric propulsion (SEP) technology are outer planet visits, cometary rendezvous, solar probes and geosunchronous satellite delivery and retrieval. Present mission planning assumes clusters of 30 cm, 2.75 kW, 3000 sec I_{sp} thrusters powered by light weight 6 to 22 kW solar arrays. Mission times in excess of 300 days thrusting time are presently under consideration.(1-5)

Small electrostatic and electrothermal thrusters will be used in station keeping and attitude control applications.(6-10) The economic advantage of achieving up to 30% more spacecraft payload by use of electric propulsion in place of conventional cold gas or hydrazine systems is a compelling reason to consider the inclusion of electric thruster subsystems in any new spacecraft design.(11) Options available to the spacecraft designer include mercury, cesium and xenon electron bombardment ion thrusters and electrothermal devices such as resistojets.

Mercury Bombardment Ion Thrusters

Prime Propulsion

The 30-cm electron-bombardment thruster (fig. 1) is being developed in an in-house effort at the NASA Lewis Research Center(12) coordinated with a contract effort at the Hughes Research Laboratories. This is the only electrostatic thruster in the United States that is presently being developed for primary propulsion. The major goal of this development program is a thrust of 0.135 Newtons (0.0305 lbs) at a specific impulse of 3000 seconds and a power input of 2.75 kilowatts (not including power conditioner losses). This performance corresponds to an overall thruster efficiency of 0.73 and a mercury ion-beam current of approximately 2 amperes. Measurement of electrical parameters including beam divergence and double ion content currently indicate a thruster efficiency approaching the goals (Table I). The corresponding goals for 3/4 and 1/2 maximum power input to the thruster are thruster efficiencies of 0.71 and 0.67 at specific impulses of 2970 and 2810 seconds. The lowered specific impulse during throttled operation results from lowered propellant utilization. If desired, a 2:1 range of throttling could be obtained at constant specific impulse by increasing the net acceleration voltage during throttled operation.

The specific impulse of 3000 secs was selected for the 30-cm thruster as a reasonable compromise for all of the likely early missions. The power level and size were chosen to obtain a module suitable for arrays with total powers from about 6 to 22 kW. From the viewpoint of thruster design, the 30-cm (ion-chamber diam.) size is also a reasonable compromise between the tendency of ion-chamber performance to improve with increasing size and the tendency of accelerator design problems (particularly at 3000 sec) to become more severe. The power level is, of course, chosen as the maximum consistent with the desired lifetime, which is at least 10,000 hrs for primary propulsion.

Accelerator Grids

The dished-grid approach was generated in the in-house program at the Lewis Research Center. The basic advantage of this approach is that grids in the approximate shape of spherical segments (fig. 2) are more stable under mechanical, thermal, and electrostatic forces than conventional flat grids.

This allows fabrication of grid geometries with closely spaced, high perveance optics. These optics have been the largest single factor which have led to the substantial increase in overall 30-cm thruster efficiency during the past year (fig. 3). Figure 4 shows the reduction in discharge chamber losses attributable to these new optics, while Fig. 5 illustrates the comparative extraction capabilities of the dished versus the flat optics for the 30-cm thruster. The fabrication of dished grids, though, has posed major problems that have been resolved only recently.

One problem has been the dishing of a pair of sheet-metal blanks. After many unsuccessful attempts, the approach finally used was a modification of conventional hydroforming. The modification was to clamp tightly enough at the edges of the sheets to avoid the customary edge slippage. In this manner the customary circumferential compression and associated small wrinkles were avoided.

Another problem that was overcome involved making the holes in dished grids. Because of the large number of holes (at least 14,000), acid etching is preferred over drilling. The photographic preparation for acid etching is done before dishing,

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with the actual etching taking place after dishing.

Beam divergence loss is typically 1-2 percent for conventional flat grids operated at normal conditions.(13) There would be some additional beam-divergence loss if all of the ion beam left the dished grids in a direction normal to the local grid surface. The actual beam-divergence loss, though, can be considerably greater than this latter effect.(14) It is customary to dish the two grids of an accelerator system together to minimize the variations in grid-to-grid spacing. This simultaneous dishing of two grids, followed by their installation in a thruster with a separation between the grids, leads to the additional beam-divergence losses. This is because the accelerator holes are displaced radially inwards relative to the corresponding screen holes when viewed from normal to the grid surface with the grids installed with the convex side out. The beamlets are attracted to the closer edges of the accelerator holes, leading to increased beam divergence. This effect can be reduced by increasing slightly the hole-to-hole spacing. If the amount of this change is chosen correctly, then all elements of the ion beam will be deflected to essentially a paraxial direction. Obtaining this slightly differing hole spacing has been accomplished by mechanically stretching the accelerator grid or by changing the hole pattern that is photographically put on the sheet before dishing. This technique has reduced thrust losses due to beam divergence to less than 1%.

The present status of dished grids is that 36 sets of 30-cm grids have been fabricated and run with a maximum of about 3000 hrs. accumulated on a single grid set.(15) Total running hours on dished grids is 12,000 hours. These grids have shown excellent stability and reproducibility, and are expected to have a lifetime of at least 15,000 hours with a 2 ampere ion beam at 3000 seconds. The close, stable spacings obtainable with the dished grids have permitted ion-beam currents up to about 5 amps for short times at 3000 seconds. Because of the clear present advantages of dished grids over competing accelerator approaches for the 30-cm thruster, most of the present development is concentrated on that approach.

Cathode. - Durability of the main discharge-chamber cathode is primarily determined by erosion of the orifice. Experience at the Lewis Research Center indicates that the erosion rate should be consistent with at least 20,000 hr life if the operating temperature of the tungsten tip is at, or below about 1100 C. A discharge potential difference of 40 V or less, which is desirable to keep the double-ionization rate low, is also assumed.(16)

The depletion of the oxide in the insert has also been suggested as a possible lifetime limitation.(17) Recent tests at the Lewis Research Center, though, have shown that the oxide depletion rate is acceptably low if the insert is recessed from the tip to maintain its operating temperatures at acceptable levels.(16)

The present cathode design for the 30-cm thruster uses a 6.35 mm tantalum tube with a 1.52-mm thick thoriated-tungsten tip electron-beam welded to it. The tip has a 0.76 mm chamfered hole. This design has a tip operating temperature of about 900 C. A heater is attached to the tube, but is used only during startup.

Neutralizer. - The same durability considerations apply to both the main cathode and the neutralizer cathode. The present neutralizer design for the 30-cm thruster uses the same tube diameter and tip thickness as the main cathode but the cylindrical part of the tip orifice is only 0.38 mm instead of 0.76 mm. The neutralizer cathode tip also operates without heater power at about 900 C. The neutral flow rate required for operation is about 0.04 amperes equivalent.

The present neutralizer position is 6.35 cm downstream and 12.7 radically outwards from the last row of holes in the accelerator. This position is a compromise between a large distance from the accelerator to avoid the SERT-II-type erosion grove, a large distance from the ion beam to minimize direct ion impingement on the neutralizer, and a small distance to the ion beam to keep the coupling voltage low - together with general trial-and-error for good control characteristics. The keeper voltage (at constant current) is used to control the neutralizer vaporizer.(18)

Auxiliary Propulsion

5-cm Thruster. - The 5-cm electron bombardment thruster was developed in an in-house effort at the NASA Lewis Research Center coordinated with a contract effort at the Hughes Research Laboratories. The 5-cm thruster subsystem, which includes mercury propellant tankage, is shown in Fig. 6. The total discharge-chamber propellant flow is fed from the single gas-pressurized tank through the cathode. Thruster potentials are separated from the grounded tank by a vapor-phase isolator. The neutralizer flow goes directly from the tank to the neutralizer. The 5-cm thruster, with the exception of the accelerator grid system, has operated more than 9715 hours in an endurance test at Lewis.(19) One of the accelerator grid systems used in this duration test has operated 7688 hours. In addition to the thruster duration tests, a separate cathode-isolator-vaporizer assembly has gone through over 3200 cycles.(20)

8-cm Thruster. - The trend in communication spacecraft is towards longer life and greater mass. To station keep larger spacecraft efficiently requires a system capable of adequate thrust levels. Recent study efforts at NASA Lewis(21) have confirmed the choice of a 1-2 millipound 8-cm diameter thruster as the best compromise for future spacecraft requirements. A comparison of 8-cm thruster parameters with those of the SIT-5 is made in Table II.

In parallel with Lewis in-house efforts, Hughes Research Laboratories is developing an 8-cm thruster (fig. 7) and power conditioner on contract NAS3-17791 and NAS3-17780. Preliminary data indicates that for 5.07 mN (1.14 mlb) of thrust at 2804 sec power input of 122 W (not including power conditioning losses) is required. This corresponds to a thruster efficiency of 57.5%. Anticipated optimized propellant utilization is 80.6%. This higher utilization efficiency, coupled with the greater thrust from the 8-cm (1.14 vs. 0.4 mlb produced by the 5-cm) results in a thrust subsystem that can perform N-S station keeping on larger spacecraft without excessive life time requirements.

System tradeoffs (ref. 21, Table III) indicate that when operation at higher thrust levels from an onboard battery becomes feasible[22], the use of multiple 8-cm mercury thrusters is an attractive alternative to the use of a larger thruster for this purpose. The duty cycle of the 15 cm thruster would be 1/3 of an 8 cm. But the increased reliability of the redundant smaller thrusters, which could run individually directly from the solar array bus in case of battery failure, would offset the slightly lower thruster efficiency (about 10%).

In addition, the smaller thrusters are more compatible with mounting at the ends of flexible, deployable solar array systems, a concept which is an integral portion of a proposed LeRC SERT C spacecraft. Thrust level and power demand of the 8-cm are more nearly commensurate with requirements for both N-S and E-W station keeping and attitude control operations than are those of the larger thrusters. Table IV demonstrates optimum 8 cm operating modes for various size spacecraft.

Vector grids. - Maximizing the usefulness of auxiliary propulsion systems requires some form of thrust vectoring system. Beam deflection allows the thrust to be aligned with the spacecraft center of mass thus preventing unwanted torques while allowing a single thruster to perform station keeping and attitude control functions. The conventional approach is to gimbal the entire thruster; this represents a weight and structural penalty.

The LeRC has tested several beam deflection concepts on the 5-cm thruster, three of which exhibited adequate life times to be suitable for synchronous satellite applications.

Translating grid system. - This accelerator system uses plane-parallel grids, with two-axis thrust vectoring accomplished by a translating screen grid (fig. 8). The motion of the screen grid is controlled by the electrical heating and cooling of springs. Over 5000 hours have been obtained in a duration test of this accelerator system with the screen grid locked in a position to give a 10 degree vectoring angle for 2000 of these hours. This mode was felt to be a more severe test than undeflected operation. The accelerator lifetime projected from this test is 20,000 hours or more.[23]

The 8-cm thruster will use a translating dished grid system to provide ±10 degree deflection. All preliminary indications are that lifetimes in excess of 20,000 hours are obtainable from this grid set. A further advantage resulting from employing this high perveance optics system is that near optimum thruster performance is obtained with no loss of vectoring capability.

Two-axis electrostatic system. - Here, the beam from each screen-grid hole is accelerated by orthogonal pairs of accelerator ribbons (fig. 9). Differential potentials across pairs of accelerator ribbons cause beam deflections of up to ±10 degrees in the same manner as the electrostatic deflection system used in cathode ray tubes. The small bent tabs shown at the intersection of ribbons in Fig. 9 were used to protect against high local arrival rates of charge-exchange ions. The locations of these tabs correspond to the erosion pits normally observed with flat, drilled accelerators. This grid system has operated for 7688 hours. This accelerator system offers the advantage of no moving parts together with full 2-axis thrust vectoring. The complexity of this accelerator design, however, results in small structural cross sections where the charge-exchange erosion takes place. Further development would be required to reach a lifetime of 15,000 hrs. Typical performance of the 5-cm thruster with this accelerator system is 1.82 mN (0.41 mlb) of thrust at a specific impulse of 2432 sec and a power input of 70 W (including power conditioning losses). This performance corresponds to an overall thruster efficiency of 0.35.

One-axis electrostatic system. - This grid is similar to the two-axis electrostatic system, except that ribbons run in only one direction and provide only one axis of thrust vectoring (fig. 10). Because there is no need for notches to avoid contact with a second set of ribbons, the one axis electrostatic system appears easier to design for long lifetime than the two-axis electrostatic system.

To date, only a 4000 hr duration test has been conducted with this accelerator system, but the low wear and ruggedness of the parts indicate a lifetime of at least 20,000 hrs.[24] Most of this test was conducted at a thrust of 4.4 mN (1 mlb), a specific impulse of 32 sec, and a power input of 148 W (not including power conditioning losses). The corresponding thruster efficiency is 0.49.

Cesium Bombardment Ion Thruster

An ion engine experiment is planned for the Applications Technology Satellite-F (ATS-F),[25,26] which is scheduled to be launched in 1974. The objective of this experiment is to demonstrate north-south station keeping of a geosynchronous spacecraft. There are two complete ion thruster systems aboard ATS-F, with the two thrusters pointing in opposite directions. The cesium bombardment thruster used in these systems (fig. 11) develops 4.4 mN (1 mlb) of thrust at 2600 sec specific impulse, thruster efficiency of 49% and provides thrust vectoring of ±3 degrees in two directions by displacement of the accelerator electrode. The propellant system is designed for two years of operation at a 25 percent duty cycle (4400 hr). The system input power is less than 150 W and the total system mass (including thruster, power conditioning, and propellant with tankage) is 16 kg (35 lb). Duration testing to date, which was obtained during cycle-tests, is a maximum of about 800 hr on a single thruster.

Xenon

Xenon propellant has been tested experimentally at LeRC in 8, 15, and 30-cm thrusters as an environmentally inert substitute for cesium or mercury.[27] In addition to having a high atomic mass (131.3 vs. 200.6 for Hg and 132.9 for Cg), xenon has a low critical pressure and can be stored noncryogenically for a moderate tankage fraction, approximately 5% greater total system mass than for mercury. Xenon is an expensive gas, but the propellant cost would still represent a small fraction of the spacecraft cost and during ground tests the xenon could be recovered and recycled.

Tests with xenon in 15 and 30 cm ion thrusters indicate that xenon performs similarly to mercury; however, there is a performance penalty, in the order of 10-20% thrust/power, due to its lower mass.

Electrothermal Propulsion

Introduction

The basic function of an electrothermal thruster is to increase the enthalpy of a fluid propellant by the addition of heat prior to expulsion through a propulsive nozzle. The increased energy content of the propellant results in improved performance (specific impulse) compared to that obtained by expelling the propellant at ambient temperature. The maximum performance that can be attained is limited by the thermodynamic properties of the propellant and the amount of thermal (electrical) energy available. Practical limitations such as material capabilities must also be considered. Electrothermal thrusters have been built and tested at design temperatures ranging from near ambient up to values which tax the capabilities of the best refractory alloys available. Several propellants have been used with electrothermal thrusters for flight applications; scores of others have been tested. Electrothermal thruster propellants include the conventional cold gas propulsion system working fluids, such as argon and nitrogen, anhydrous ammonia, anhydrous hydrazine, and the biowaste gases resulting from the environmental control systems of manned spacecraft.

The propellant supply systems used with electrothermal thrusters are generally not unique to the electrothermal propulsion field. The first electrothermal thrusters to fly, those aboard the Vela-series spacecraft in 1965, were simply added to an existing cold gas propulsion system to increase the available total impulse.[28] Ammonia propulsion systems have been flown with both electrothermal thrusters and ambient-temperature propulsive nozzles. One of the significant advantages of electrothermal hydrazine thrusters is the ability to share a common propellant supply system with much higher thrust catalytic engines. Electrothermal propulsion technology therefore relates primarily to thruster technology. The feed system technology is relatively more mature.

It is the intent of this section of the paper to review the current status in electrothermal propulsion. A tradeoff study is outlined whereby a spacecraft designer can compare the merits of electrothermal propulsion with those of other types of systems in a very general manner. A few representative examples of propellant supply systems and thrusters are presented. For a more comprehensive survey of available hardware, refer to References 29, 30, 31, and 32. Interest in the gaseous, inert electrothermal propellants such as nitrogen has waned in recent years since higher performing systems are available. The development of biowaste electrothermal thrusters has been slowed considerably by the reorientation of the manned space program. Ammonia propulsion, on the other hand, is a viable, expanding technology both with and without electrothermal thrusters. A significant effort is also currently being expended in the area of electrothermal hydrazine propulsion, and these systems are almost certainly destined for use on a number of missions. The primary emphasis of this section will therefore be placed on ammonia and hydrazine systems with a rather cursory review of the other types of electrothermal propulsion. Projected development trends are also discussed.

Propulsion System Selection

For many spacecraft, the selection of a propulsion system may be relatively simple. If the total impulse requirement is very small, for instance, the high weight of a nitrogen pneumatic system may be tolerated in light of cost and simplicity advantages. Over a rather wide thrust and duty cycle range, catalytic hydrazine offers very significant advantages over other types of propulsion systems and the selection becomes obvious. However, there is a broad class of missions (primarily three-axis-stabilized, earth-orbiting spacecraft) to which several types of propulsion systems are applicable and the selection process becomes more difficult.

Consider, for instance, a 1000-pound three-axis stabilized, earth-orbiting spacecraft which requires attitude control, stationkeeping, and repositioning capabilities. The total impulse requirement neglectint North-South station keeping would be of the order of 7000 lb-sec for a 5-year mission. The optimum impulse bit size for attitude control may be in the range of 10^{-3} pound-second and cross-coupling torques may limit the stationkeeping thrust level to a few millipounds. For this hypothetical case, a nitrogen pneumatic system is noncompetitive since it would weigh about 250 pounds. Catalytic hydrazine thrusters are not available at the low thrust levels required. High-performing electric propulsion systems may be used to perform the orbital maneuver functions but their obvious weight advantage may be negated at least partially by the requirement for a separate attitude control system for pulsed operation.

The remaining viable condidates include ammonia, either unheated or electrothermal, and hydrazine. The hydrazine system may be comprised of a gas generator/plenum system with or without electrothermal thrusters or a conventional liquid distribution system with electrothermal thrusters. Each of these systems will be discussed in subsequent paragraphs and comparative data will be given.

Ammonia System Description. Two types of ammonia propulsion systems have been flight qualified. In one, unheated propellant is delivered to the nozzles, while in the other, the propellant is resistively heated prior to expansion through some of the nozzles. In the latter case, the nozzles to be heated are those by which spacecraft velocity corrections are made. Except for the nozzle (thruster) configuration and the propellant and tankage volume and weight, the ammonia supply systems are identical whether heated or not.

A schematic of a fully-redundant ammonia system is shown in Figure 12. The feed system shown is based on zero-gravity ammonia heat exchanger developed by TRW.[33] Capillary tubes*, bonded to

*The term "capillary" refers to the diameter of the tubing (0.016 in. ID). Capillary forces play no role in the process.

the propellant tank outer surface, act as a heat exchanger to vaporize any liquid that exits the tank. The regulator accepts either liquid or vapor, but is controlled by the pressure at the gas (downstream) side of the capillary tubes. The plenum tank provides a reservoir to minimize transients and to limit the number of regulator cycles. The capillary tubes are sized to completely vaporize all liquid over the entire range of anticipated flow rates. Typically, a regulated pressure of from 10 to 35 psia is supplied to the nozzles.

Hydrazine System Description. Three types of hydrazine systems are applicable to the hypothetical mission mentioned earlier: the monoplenum, a hybrid monoplenum/direct decomposition design, and a hydrazine thermal decomposition thruster design. All-catalytic systems were not considered because no thrusters are currently available in the very low thrust ranges (< 30 millipounds).

Propellant feed and pressurization systems are common for all of the hydrazine systems. The preferred approach (Figure 13) is to use a positive expulsion diaphragm-type tank which contains both propellant and pressurant. As the propellant is expelled, the pressurant gas expands, reducing the feed pressure. In a system which uses propellant at a very slow rate, expansion is essentially isothermal. The decomposition device called out in the figure may be a set of thruster assemblies or a gas generator.

If the decaying thrust cannot be accommodated, then a regulated pressurant supply can be added as shown in Figure 14. However, the added complexity which is obvious in the schematic causes cost, weight and reliability penalties.

The hydrazine monoplenum concept is shown schematically in Figure 15. In this case, a single gas generator supplies gaseous hydrazine decomposition products to the thrusters at regulated pressure. The thrusters may be comprised of unheated nozzles or electrothermal devices.

System Comparison. System performance parameters have been calculated for several different configurations. These are summarized in Table V. All include redundant thrusters. All assume 1500 lb-sec of total impulse for attitude control and 5500 lb-sec for orbital maneuvers (E-W station keeping and station walking).

The first ammonia system involves the use of unheated nozzles at a thrust level of 0.010 pound. With an average specific impulse of 100 seconds, the system total impulse of 7000 pound-seconds can be achieved with 70 pounds of propellant. The total system weight will be 88.5 pounds. A power of 17 watts is required for vaporization during velocity corrections only.

The second system uses unheated nozzles for attitude control and heated thrusters for velocity corrections. All nozzles deliver 0.010 pound of thrust. The unheated nozzles operate at an average specific impulse of 100 seconds, while the electrothermal thrusters will achieve 195 seconds with a power input of about 60 watts. A power of 9 watts will be used to vaporize propellant during velocity correction maneuvers. Propellant weight for this case will be 43 pounds and total system weight will be 65.5 pounds.

The third system is similar to the second, except that the velocity control thrust level is 0.005 pound. With a power input of 45 watts, the heated thrusters will operate at a specific impulse of 222 seconds. Propellant weight will be 40 pounds and total system weight will be 62.2 pounds.

Threse three cases are representative of the flexibility of an ammonia system. The parameters of input power, thrust and specific impulse may be varied to arrive at a final configuration which is optimum for a given spacecraft.

The fourth system is a hydrazine monoplenum system. The propellant is decomposed in a gas generator and the decomposition products are stored at regulated pressure in a plenum. The thrusters expel the decomposition products (NH_3, N_2 and H_2) at ambient temperature. The delivered specific impulse is about 105 seconds. System weight is about 90 pounds.

The fifth system uses electrothermal hydrazine thrusters with a conventional blowdown-type supply system. The thrust decreases continuously throughout the mission as propellant is expended. The blowdown ratio is about 2:1. For attitude control, the delivered specific impulse is about 150 seconds. For velocity corrections, the specific impulse is about 220 seconds. Note that the 0.030-0.015 pound thrust level of the ΔV thrusters may be high for the assumed mission and pulse modulation may be necessary. Also the attitude control thrust and impulse bit will vary predictably with supply pressure.

The final system uses the electrothermal thrusters for ΔV only and uses the monoplenum system for attitude control. This system combines the high performance of the electrothermal thrusters with the excellent pulse reproducibility of the gaseous attitude control thrusters.

At this point, the spacecraft designer is still faced with selection of the optimum system in terms of weight, power, volume, and cost and their relative priorities. Other less tangible factors such as reliability, complexity, qualification status and growth potential must also be considered.

State-of-the-Art Hardware

More than twenty spacecraft have been launched with electrothermal propulsion systems. Most of these used ammonia propellant. The following paragraphs briefly describe several propellant supply systems and examples of thrusters which are representative of the current state-of-the-art.

Figure 16 shows the electrothermal ammonia system[34] built by AVCO Space Systems Division which was used for stationkeeping on the Applications Technology Satellites D and E. The first launch was in 1968. Each of two thrusters provided 50 micropound of thrust. The overall system power requirement was 10 watts. The propulsion system was tested periodically in orbit; the final test came three years after launch. No apparent degradation was observed.

AVCO also provided the ammonia system shown in Figure 17 which was successfully flown aboard LES-6[35]. TRW is currently fabricating a much larger ammonia propulsion subsystem for LES-8. This system, shown schematically in Figure 18, carries a propellant load of 70 pounds. The system is unique in that virtually all of the system components, including tanks, valves, fillers, control units, fill valves and transducers were previously qualified for hydrazine service. Most of the LES-8 ammonia system components are the same as those used in the system shown in Figure 19. This system,[36] developed for an Air Force spacecraft, uses catalytic hydrazine thrusters for velocity corrections and high level attitude control. Low level attitude control is provided by a hydrazine gas generator and electrically-heated, warm-gas nozzles.

Literally dozens of electrothermal thruster concepts have been developed. The Marquardt Corporation has developed a concentric tube design[37] (Figure 20) which was tested for 8000 hours with ammonia and hydrogen propellants at temperatures of about 3500°F. The same generic design was used with the biowaste propellants carbon dioxide, methane and water. One of AVCO's resistojet concepts is the extremely simple self-heated tubular heat exchanger and nozzle[38] shown in Figure 21. TRW's vortex design[39] is equally simple and employs a coaxial heater element located within a vortex heat exchange cavity (Fig. 22).

Electrothermal hydrazine thrusters differ somewhat from other electrothermal thrusters in that most of the heat is supplied by an exothermal decomposition process rather than electrically. A heater element supplies enough energy to trigger the decomposition reaction and the chemical energy increases the chamber temperature to about 1700°F. As a result, the power requirements for electrothermal hydrazine thrusters are quite low. The TRW thruster[40] shown in Figure 23 requires 3-5 watts for thrust levels between 0.010 and 0.1 pound. Steady state specific impulse is about 225 seconds. Performance of the AVCO hydrazine resistojet is similar.

Technology Trends

Several trends are evident in the area of earth earth orbital missions. Three-axis-stabilized spacecraft are generally supplanting the spin-stabilized design. Mission durations are becoming longer; five years is now the norm. Pointing accuracy requirements are becoming more stringent. Weight penalties for spacecraft power are decreasing. And finally, spacecraft experiments are becoming more sophisticated and more susceptible to contamination from thruster exhaust plumes. In terms of propulsion system requirements, these trends translate into: lower thrust levels, smaller impulse bits, the need for more precise impulse bit reproducibility and higher values for total impulse and thruster performance. All of these trends favor a more widespread use of electrothermal and elestrostatic propulsion.

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TABLE I. - 30-CM ION THRUSTER PARAMETERS

Thrust, mlb	28.8
Specific impulse, sec	2955
Total input power to thruster, W	2620
Ion beam current, amp	2.0
Net accelerating potential, V	1100
Output beam power, W	2200
Thruster power efficiency, %	83.5
Thruster weight, lb	16.5
Power processor weight, lb	34
Power processor electrical efficiency, % . .	92
Power processor input power, W	2850

Power processor size 22"x50"x4-1/2" (self radiating shear plate)
Vibration Qual Spec - Thor-Agena

TABLE II. - 8-CM THRUSTER PARAMETERS COMPARED WITH SIT-5 PARAMETERS

	5-cm performance	8-cm performance goals	8-cm performance	
Thrust (ideal), mlb	0.40	1.14	1.14	2.06
Specific impulse, sec	2432	2804	2530	2870
Total input power, W	70.0	122.2	154.6	232
Total efficiency, %	30.5	57.5	41.0	55.8
Power efficiency, %	46.8	71.3	56.4	67.7
Total utilization, %	65.0	80.6	72.7	82.5
Discharge utilization, %	68.8	86.4	77.5	88.5
Total neutral flow, mA	36.0	89.3	99.0	157.7
Power/thrust, W/mlb	175	107	135.5	112.5
eV/ion excluding keeper, V	607	294	567	363
eV/ion including keeper, V	825	328	608	382
Beam current, J_B, mA	23.4	72	72	130
New accelerating voltage, V_I, V	1409	1220	1220	1220
Neutralizer floating potential, V_g, V	-9	-10	-8	-10
Output beam power, W	32.7	87.1	87.1	157.5
Accelerator voltage, V_A, V	-700	-500	-500	-500
Accelerator drain current, J_A, mA	0.082	0.23	0.135	0.27
Accelerator drain power, W	0.17	0.40	0.23	0.46
Discharge voltage, V_I, V	40	40	40	33
Emission current, J_E, A	0.355	0.53	1.02	1.43
Discharge power, W	14.2	21.2	0.8	47.2

TABLE III. - FIVE YEAR STATION KEEPING OF A 1500 LB SPACECRAFT

		Thruster North	Thruster South	Thruster East	Thruster West	Totals
8-cm Thruster (1 mlb thrust 138 watts)	Propellant wt., lb*	9.8	9.8	4.3	3.4	27.3
	Tankage wt., lb	3.0	3.0	2.0	2.0	10.0
	PPU wt., lb	9.0	9.0	9.0	9.0	36.0
	Thruster wt., lb	5.1	5.1	5.1	5.1	20.4
	Total wt., lb	26.9	26.9	20.4	19.5	93.7
SERT II 15-cm Thruster (6.3 mlb thrust 850 watts)	Propellant wt., lb*	11.2	Not required	4.1	3.7	19.0
	Tankage wt., lb	2.6		2.0	2.0	6.6
	PPU wt., lb	32.0		32.0	32.0	96.0
	Thruster** wt., lb	23.6		23.6	23.6	70.0
	Total wt., lb	69.4		61.7	61.3	191.6

*Includes startup propellant consumptions.

**Includes gimbals.

TABLE IV. - TOTAL SYSTEM MASS MINIMUMS AND ASSOCIATED OPTIMUM PARAMETERS FOR VARIOUS SPACECRAFT MASSES ASSUMING USE OF TWO 8 CM THRUSTERS (D = 8 CM), I_{sp} = 2750 SEC AND Y = 5 YR

M_{sc}, kg	Independent propellant tankage			Connected propellant tankage		
	M_{sys}, kg	J_B, A	T, hr	M_{sys}, kg	J_B, A	T, hr
500	31.9	0.036	5911	26.4	0.036	5911
750	41.1	.050	6363	32.5	.050	6363
1000	49.9	.072	5911	38.4	.065	6618
1250	58.5	.093	5669	44.2	.079	6781
1500	66.9	.108	5911	49.8	.100	6363
1750	75.2	.129	5736	55.2	.115	6506
2000	83.5	.143	5911	60.7	.129	6618

TABLE V. - SYSTEM COMPARISON

	Cold NH_3	Cold & Heated NH_3	Cold & Heated NH_3	Mono/ Plenum N_2H_4	Electrothermal N_2H_4		Hybrid/ Plenum N_2H_4	
Thrust Level, pound								
Attitude Control	0.010	0.010	0.010	0.010	0.030	0.015	0.010	
Velocity Correction	0.010	0.010	0.005	0.010	0.030	0.015	0.030	0.015
Specific Impulse, seconds								
Attitude Control	100	100	100	105	150		100	
Velocity Correction	100	195	222	105	220		220	
Propellant Weight, pounds	70	43	40	67	35		40	
System Weight, total, pounds	88.5	65.5	62.2	90	58		66	
System Power								
Attitude Control Mode								
Valves, watts	2	2	2	2	2		2	
Thruster heaters, watts	---	---	---	---	20		---	
Velocity Correction Mode								
Valves, watts	2	2	2	2	2		2	
Thruster heaters, watts	---	60	45	---	5		5	
Tankage heaters, watts	17	9	---	---	---		---	
Control System Power								
(transducers, control unit, valves, etc.) watts	1	1	1	8	1		8	
Peak Power (watts)	22	74	50	12	30		17	

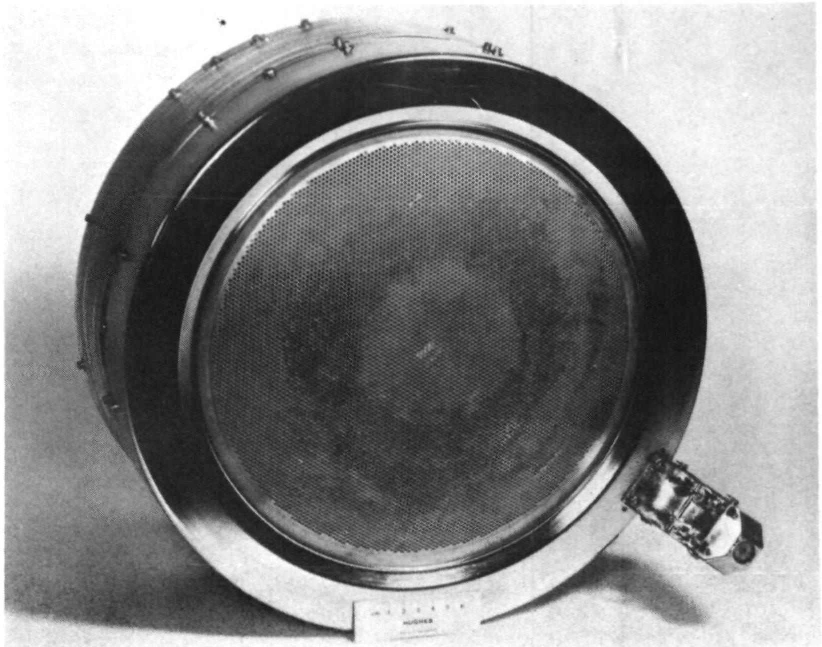

Figure 1. - 30-Cm electron bombardment thruster with dished accelerator grids.

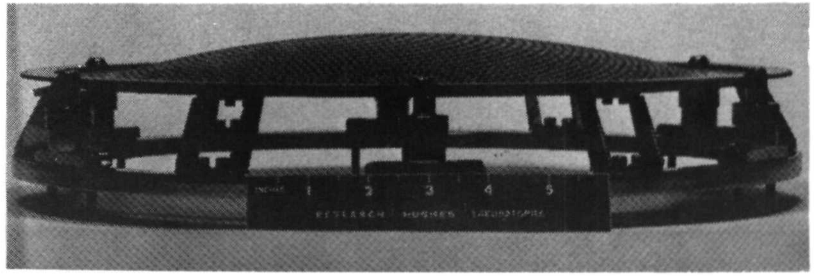

Figure 2. - 30-Cm dished grids with mounting structure.

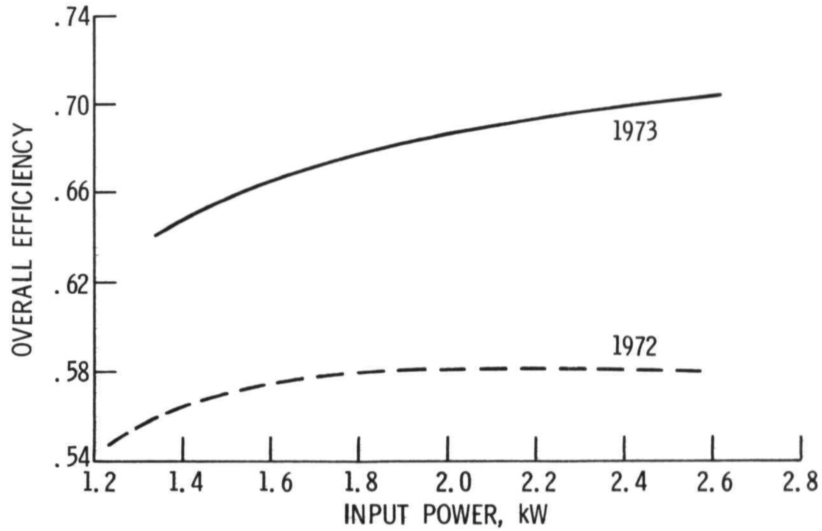

Figure 3. - Measured 30-cm thruster performance (corrected for beam divergence and double ions).

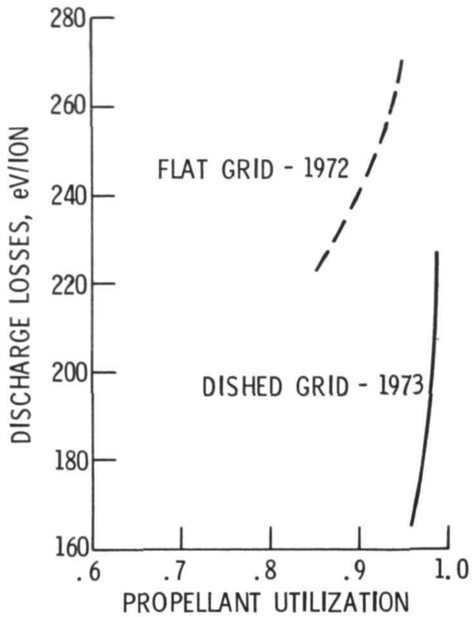

Figure 4. - Discharge chamber performance.

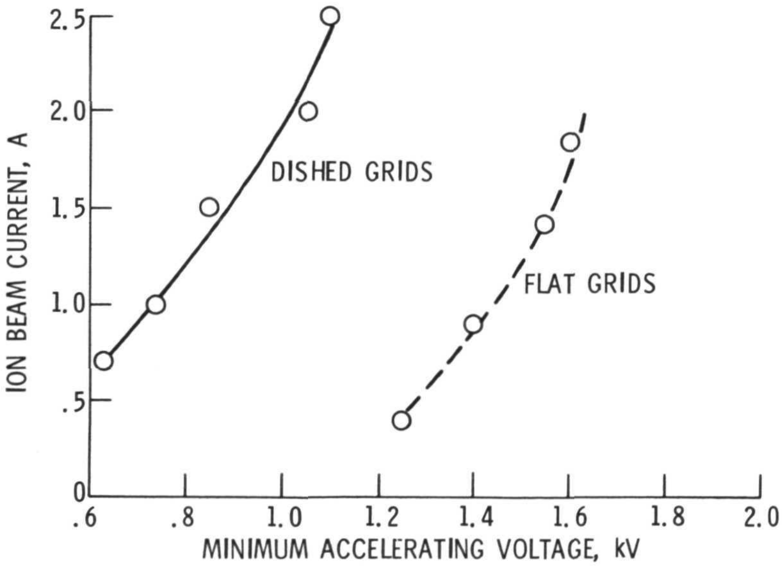

Figure 5. - Comparison of current extraction.

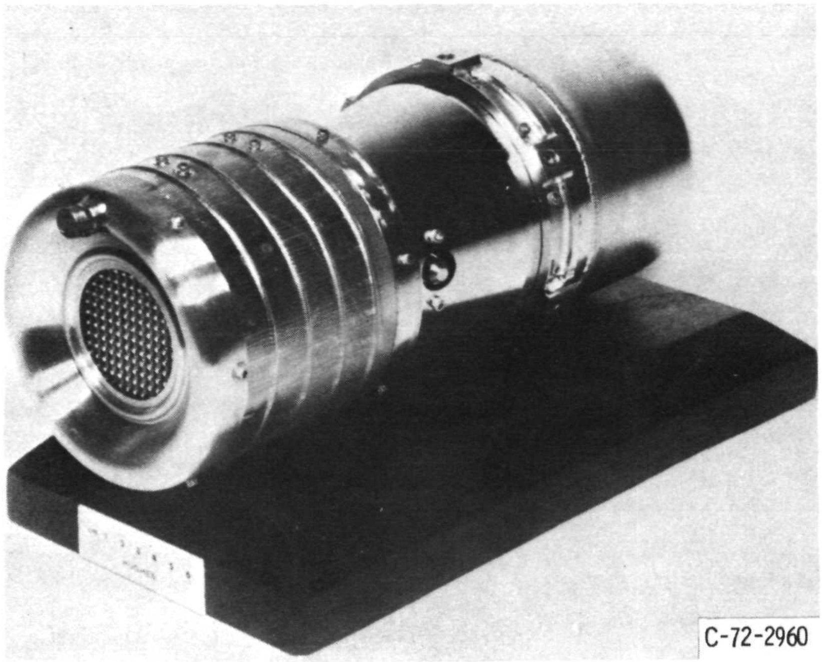

Figure 6. - 5-Cm electron-bombardment thruster.

E-7686

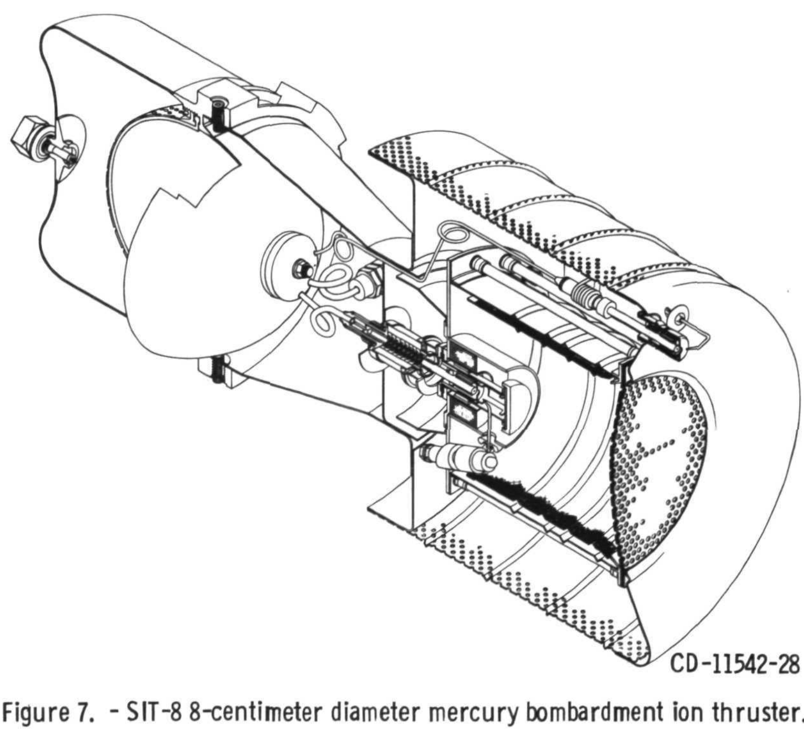

Figure 7. - SIT-8 8-centimeter diameter mercury bombardment ion thruster.

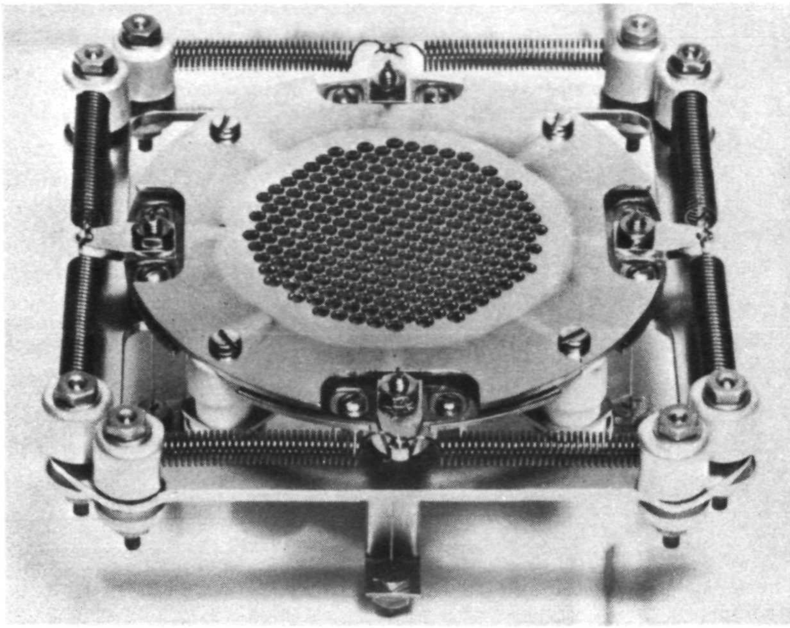

Figure 8. - 5-Cm grids with two-axis translation vectoring.

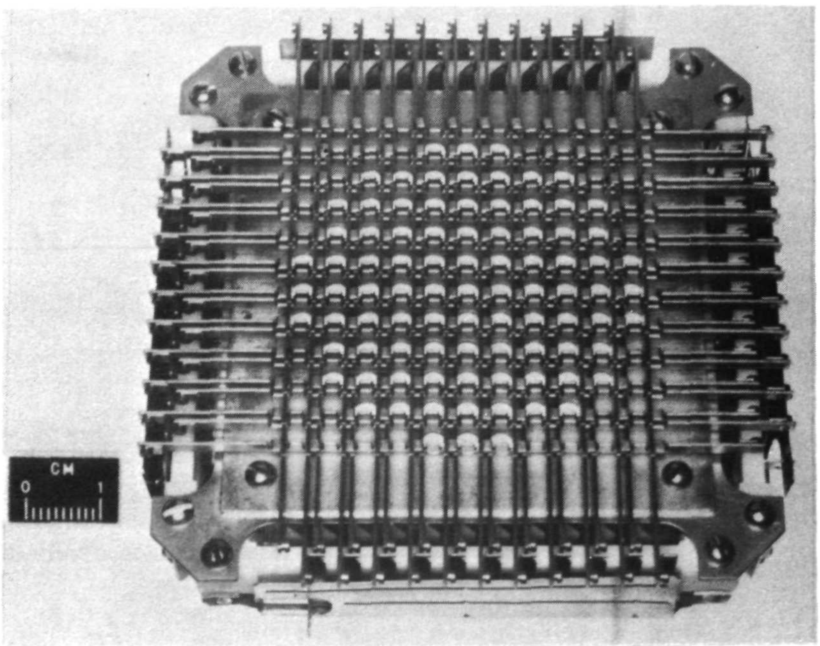

Figure 9. - 5-Cm grids with two-axis electrostatic vectoring.

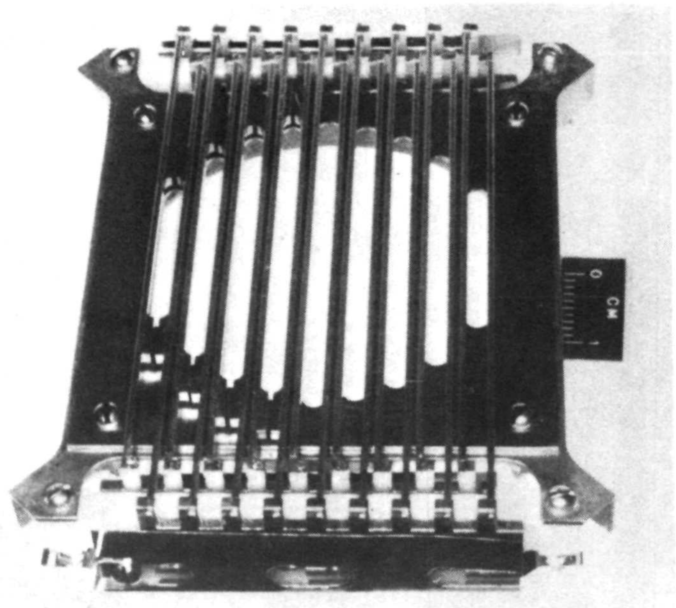

Figure 10. - 5-Cm grids with one-axis electrostatic vectoring.

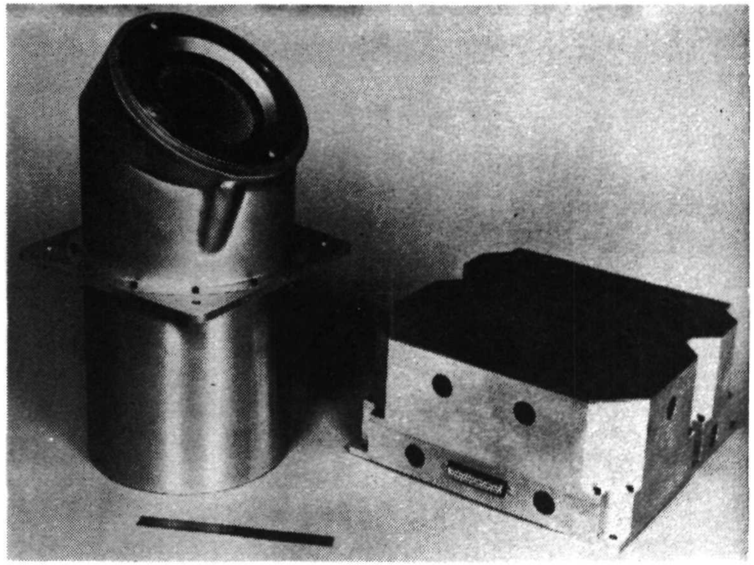

Figure 11. - Cesium bombardment thruster to be used on ATS-F.

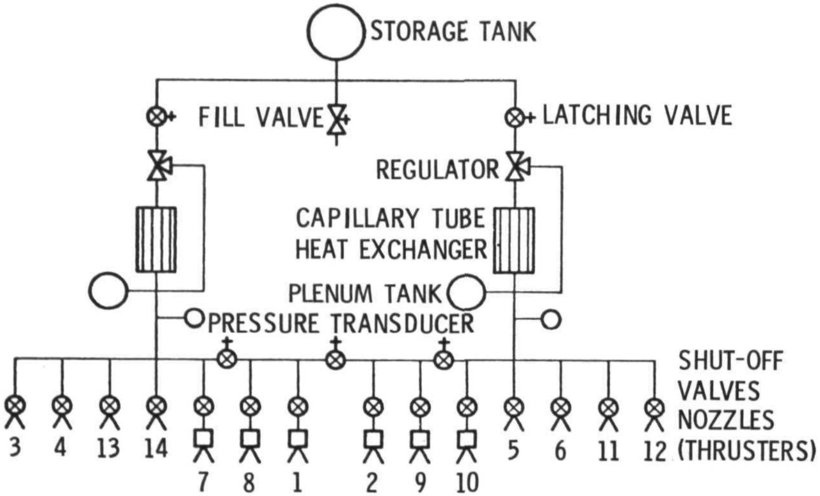

Figure 12. - Ammonia system schematic.

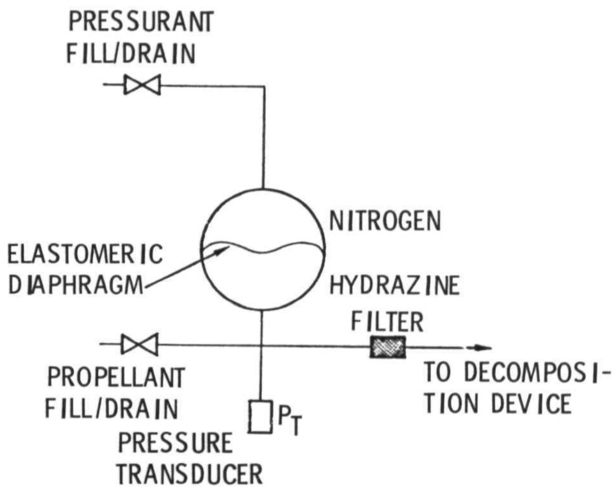

Figure 13. - Basic hydrazine feed system.

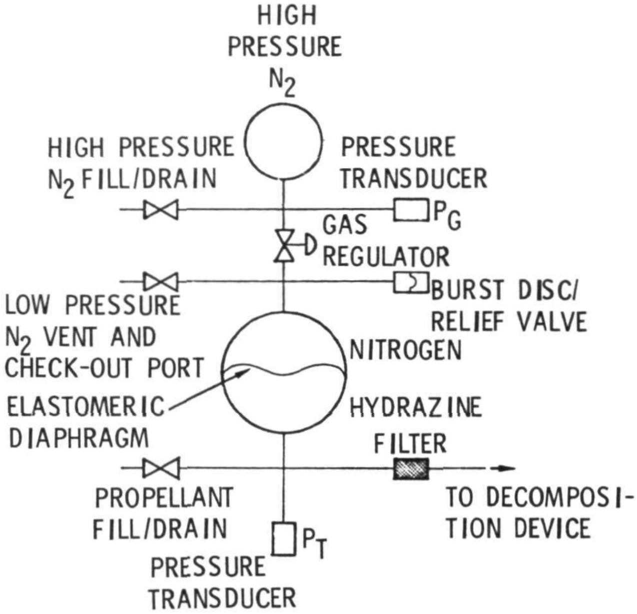

Figure 14. - Regulated pressure hydrazine feed system.

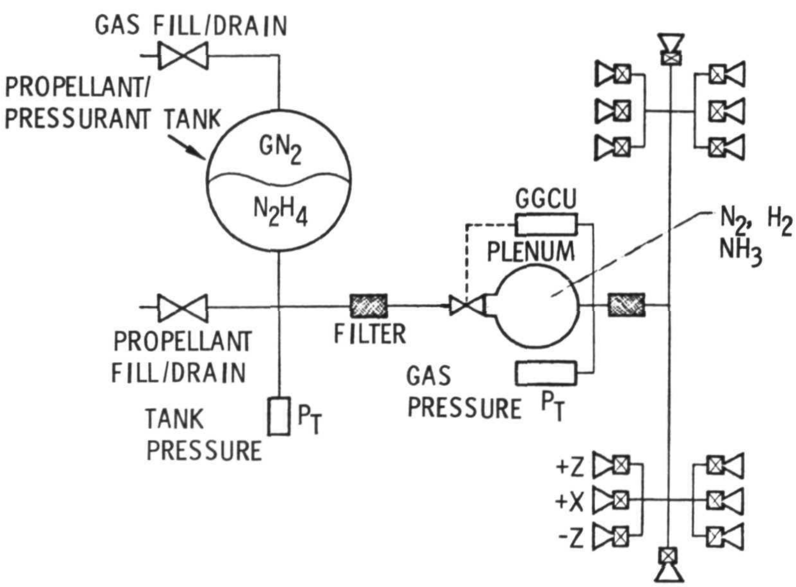

Figure 15. - Schematic of monoplenum system.

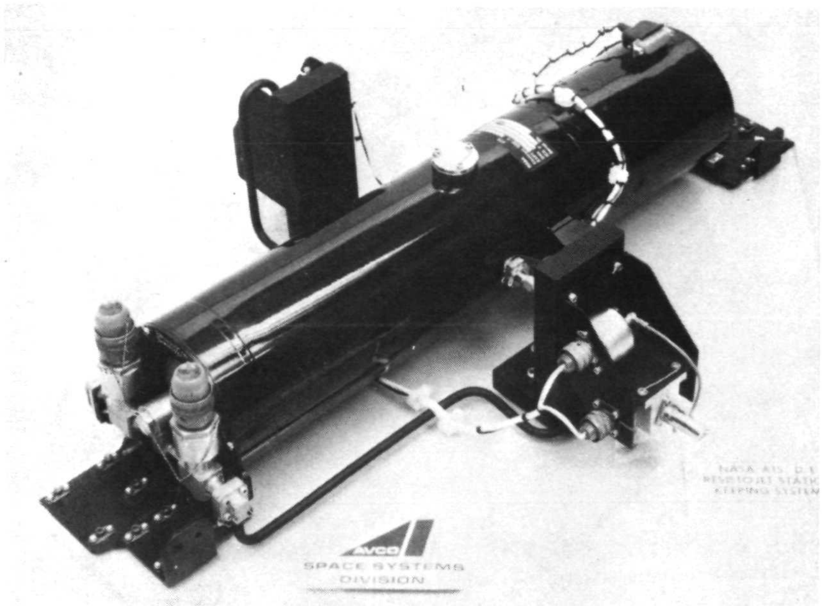

Figure 16. - Ammonia propulsion system for ATS - D/E.

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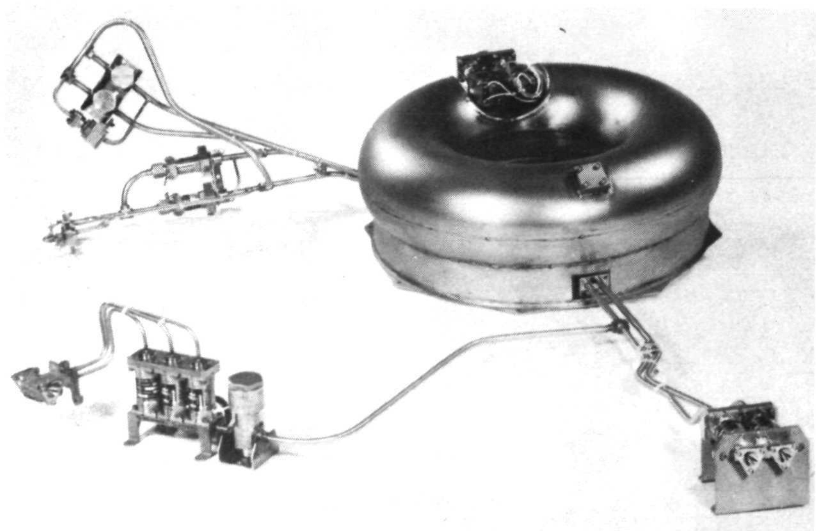

Figure 17. - LES-6 Ammonia propulsion system.

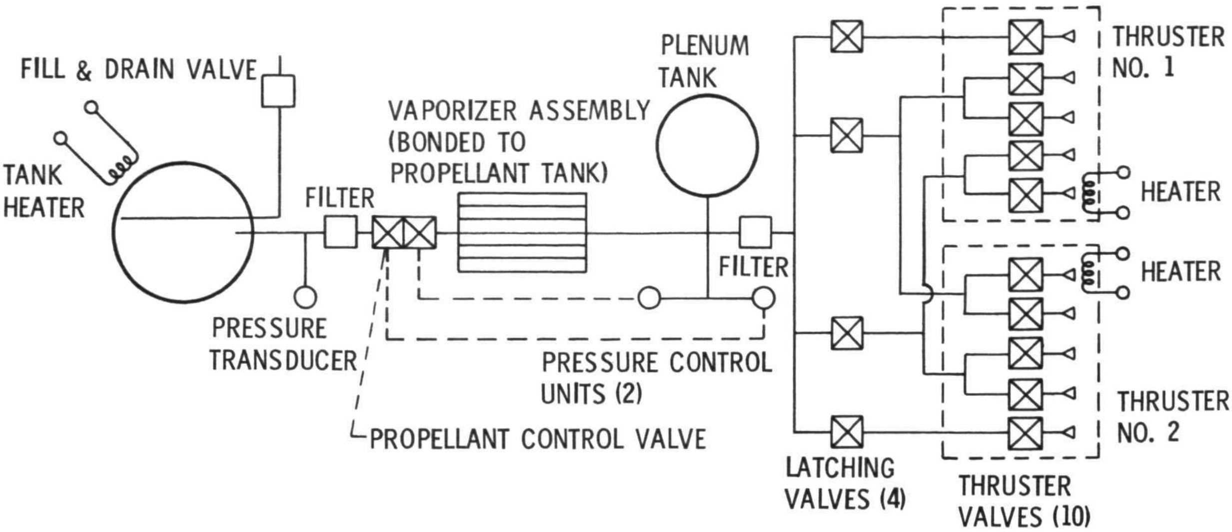

Figure 18. - LES-8 ammonia propulsion system schematic.

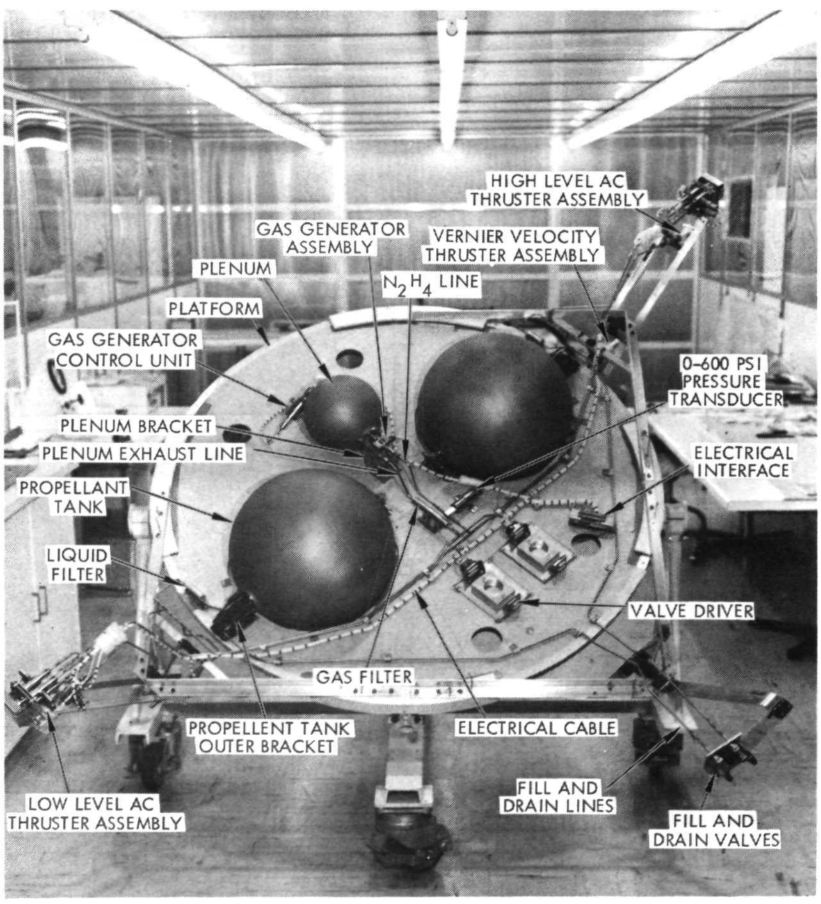

Figure 19. - Monoplenum hydrazine system.

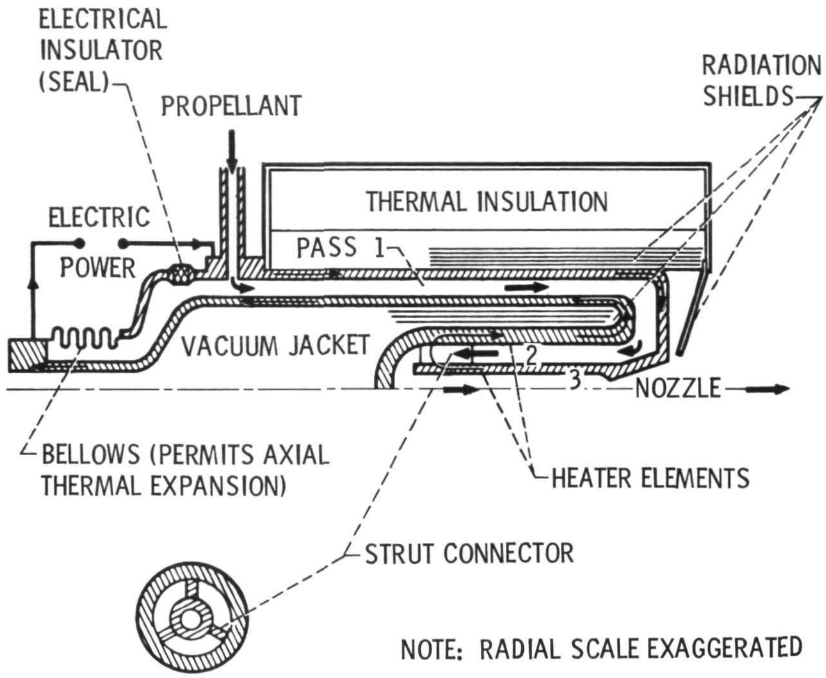

Figure 20. - Concentric tube thruster.

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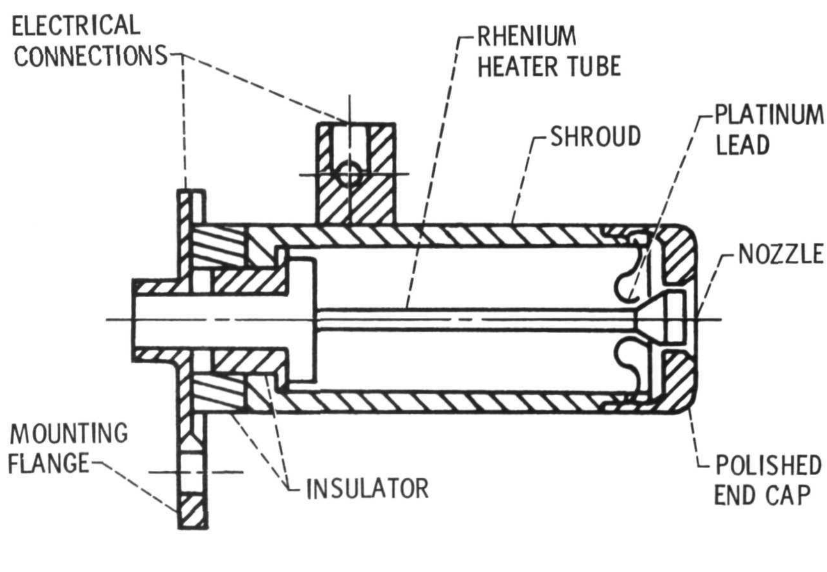

Figure 21. - Tubular heat exchanger thruster - AVCO.

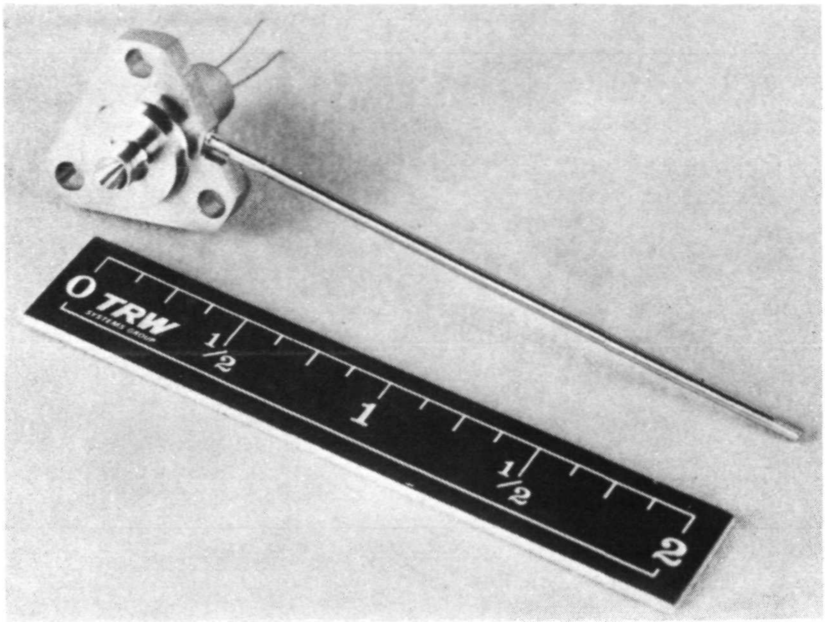

Figure 22. - TRW vortex thruster.

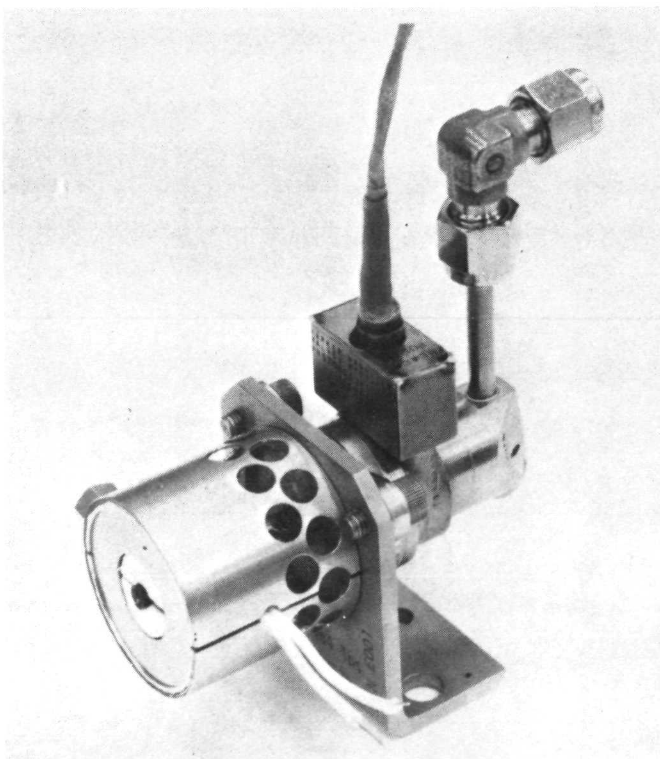

Figure 23. - Electrothermal hydrazine thruster.