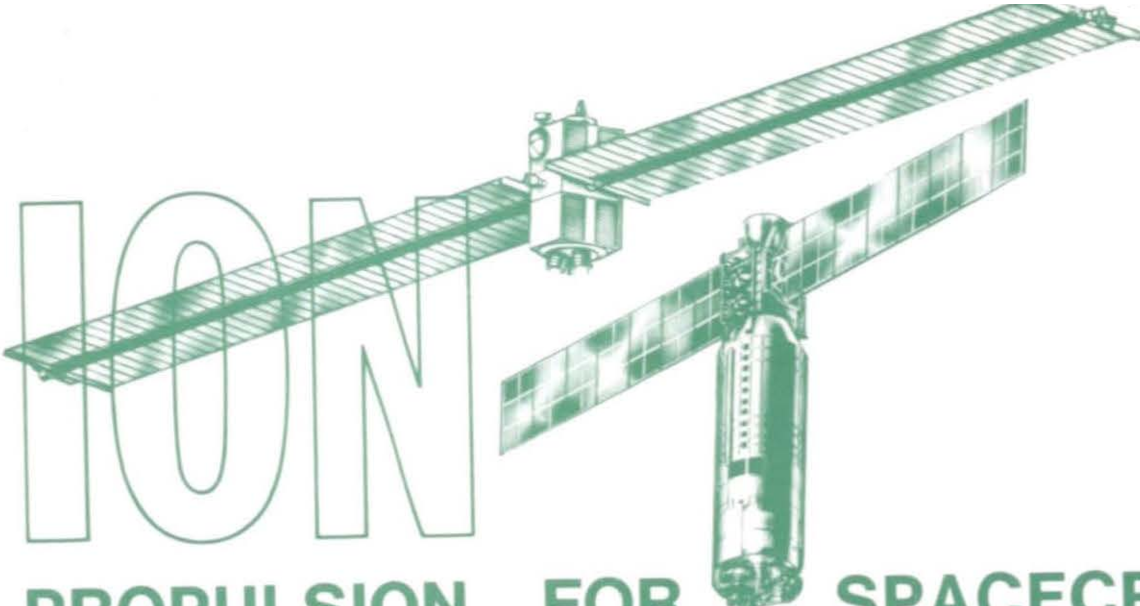
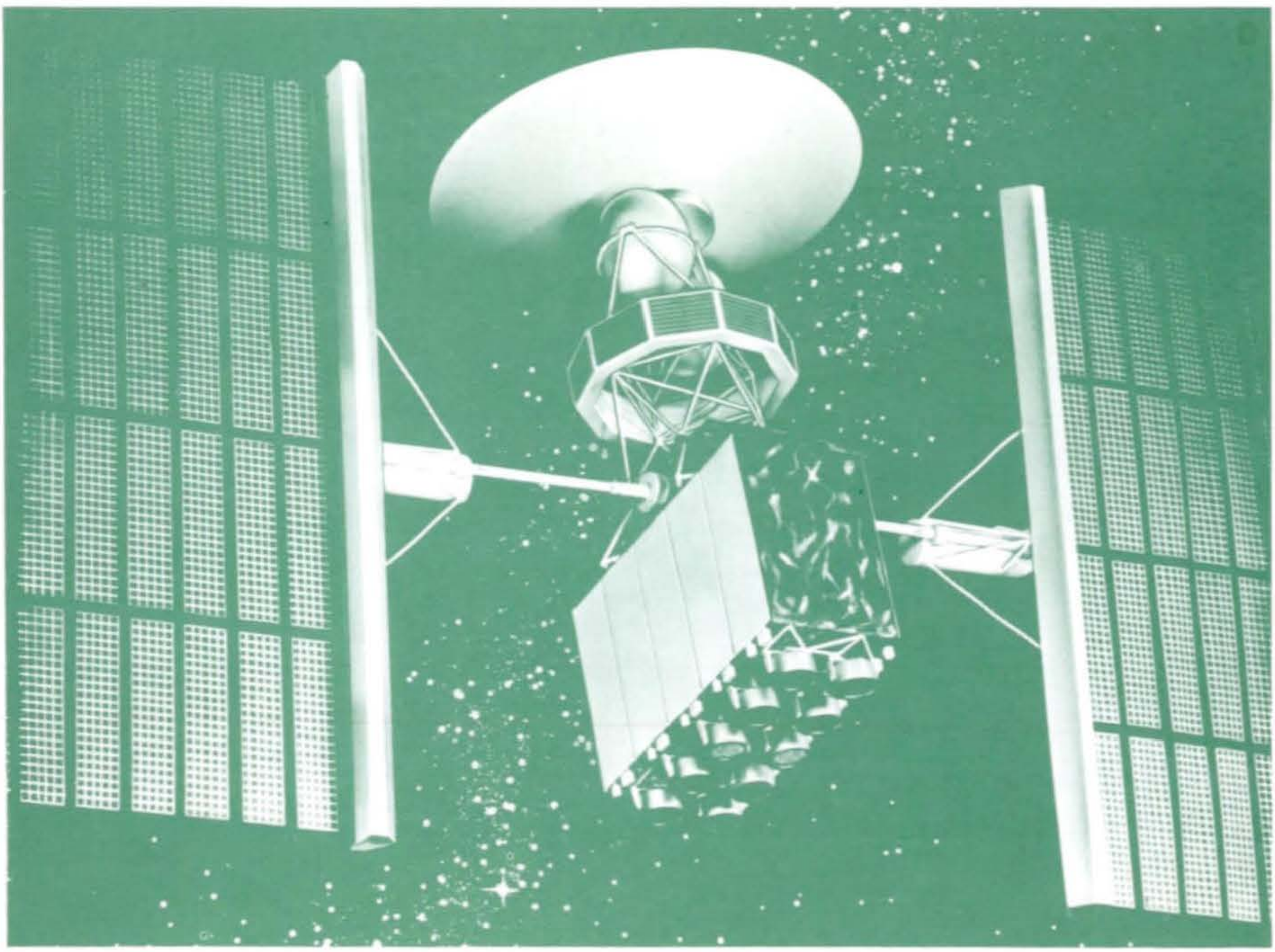


ION PROPULSION FOR SPACECRAFT



National Aeronautics and Space Administration

Lewis Research Center

CLEVELAND, OHIO 1977

ion
propulsion
for
spacecraft

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ion propulsion for spacecraft

Spaceflight has been a reality for only a score of years, yet in that time enormous technological strides have been made. Unmanned exploration of the inner and some outer planets has begun. The Mariner, Explorer, and Pioneer series spacecraft are yielding a wealth of knowledge about the nature of the solar system. Satellites in geosynchronous orbits have become a practical reality, providing physical data about the Earth and worldwide services in communication and navigation.

To date, virtually all propulsion systems for planetary or Earth orbital applications have been chemical devices. As man progresses in space, however, the missions will become more extensive and difficult to accomplish with only chemical propulsion.

Because of the range of propulsion needs, scientists and engineers continually evaluate improved propulsion system concepts. This booklet describes one type of propulsion system that offers promise of filling a wide range of propulsion needs – an ion-thruster propulsion system. The mercury-bombardment ion thruster of this system has evolved to the point of flight readiness with a propellant specific impulse of 3000 seconds. This impulse is approximately six to eight times that of the best chemical rocket systems. As is shown later, the mass of the required electric powerplant does not permit a corresponding six- to eight-fold increase in the velocity increment imparted to the spacecraft, but the gains are substantial. Thus, the ion-thruster propulsion system opens new vistas to the mission planner and enlarges the total spectrum of achievable missions.

ion propulsion theory

The principal elements of the ion-thruster propulsion system are shown in figure 1. The Sun provides energy, which is converted into electric power by solar cells. The power is then conditioned to the current and voltage needed by the ion thruster. Propellant is ionized in the thruster and electrically exhausted to produce thrust. For many missions, the power source can serve the dual roles of providing both thruster

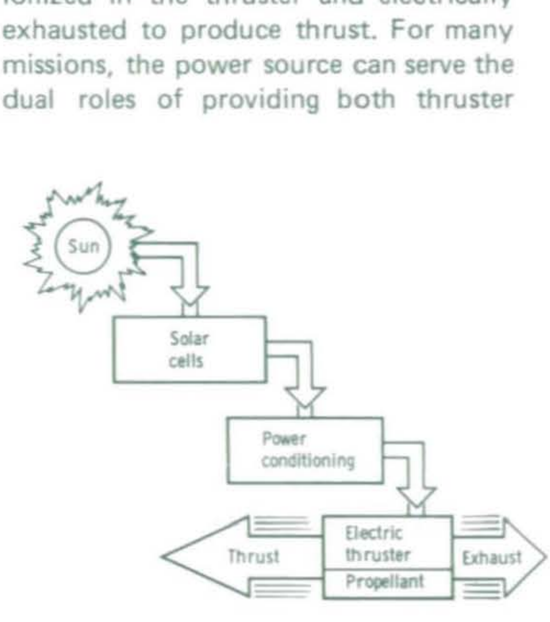

Figure 1.

power and power for mission objectives subsequent to the thrusting period. The thruster will be of appropriate size to satisfy the thrust requirements for the particular propulsion task.

The main advantage of using electric propulsion is that the electric energy added to the exhaust propellant greatly increases its velocity, or specific impulse; and hence more thrust is produced with the same propellant flow rate. The mass of propellant required to produce a given thrust decreases with increasing specific impulse, as shown in figure 2. The saving in propellant mass, however, is offset by the increasingly massive powerplant required to accelerate the exhaust to higher velocities. This increase in powerplant mass is shown in figure 3. The maximum payload of a spacecraft is achieved at the optimum specific impulse, where the sum of the propellant and powerplant masses is a minimum, as shown in figure 4.

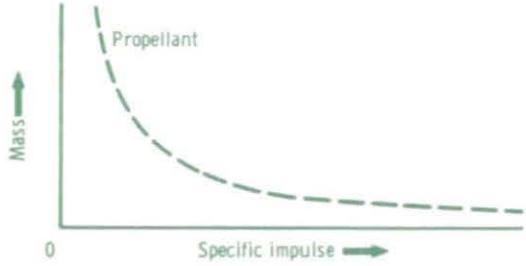

Figure 2.

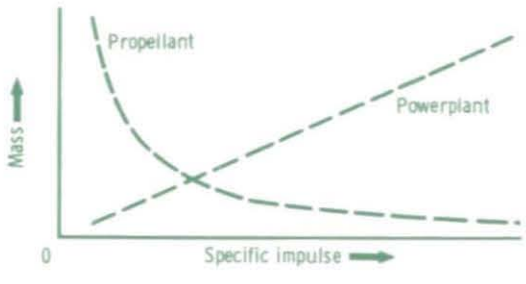

Figure 3.

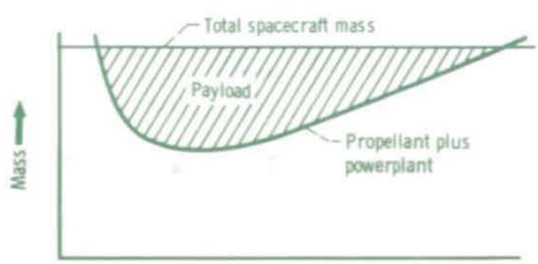

Figure 4.

Figure 4 indicates that at low specific impulse the propellant mass can be excessively large, while at high specific impulse the powerplant mass becomes excessive. Between these two extremes is a broad useful range where sufficient payload remains for design of a practical spacecraft. As defined by figure 4, payload includes the mass of the spacecraft itself and the useful payload. The optimum value of specific impulse to maximize payload usually is between 2000 and 5000 seconds, and thus the optimum value of exhaust velocity is between 20 000 and 50 000 meters per second. This range of exhaust velocity is easily achieved with ion thrusters and, as is discussed later, results in large increases in spacecraft payload over a large variety of missions.

ion-thruster operation

The first electron-bombardment thruster was conceived and tested by Dr. Harold R. Kaufman in 1959 at the NASA Lewis Research Center (ref. 1). This thruster operates by flowing a gaseous propellant into a discharge chamber. The propellant may be any gas,

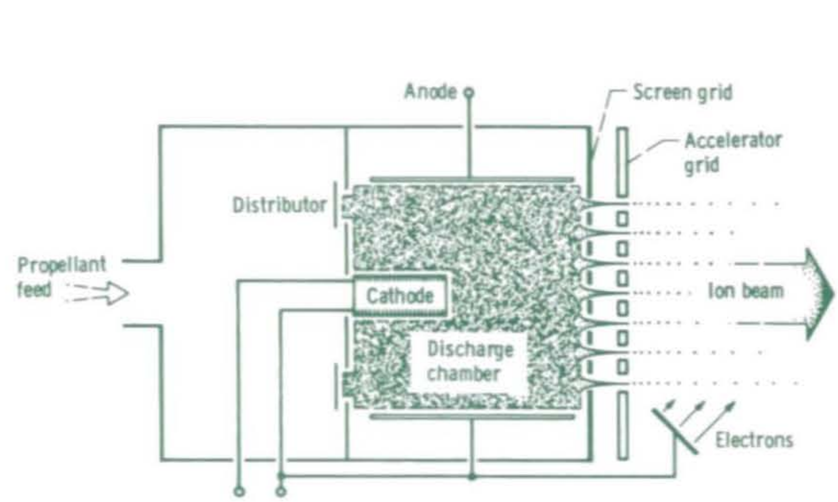

Figure 5.

but mercury, cesium, and the noble gases are the most efficient for propulsion applications. Propellant atoms are ionized in the discharge chamber by electron bombardment in a process similar to that in a mercury arc sunlamp. This ionization occurs when an atom in the discharge loses an electron after bombardment by an energetic (40-eV) discharge electron. The electrons and the ions form a plasma in the ionization chamber. The electric field between the screen and the accelerator draws ions from the plasma. These ions are then accelerated out through many small holes in the screen and accelerator electrode to form an ion beam, as shown in figure 5. A neutralizer injects an equal number of electrons into the ion beam. This beam of electrons allows the spacecraft to remain electrically neutral and is a requirement for successful thruster operation. A more complete description of the mercury-bombardment ion thruster is given in the appendix.

Laboratory testing of thrusters must be done in a moderately large vacuum facility in order to simulate the environment of space. Facilities such as the one shown in figures 6 and 7 are thus required for laboratory testing. Typically, these facilities are capable of simulating altitudes of more than 300 kilometers, where the background air pressure is less than 1/100 000 000 of sea-level pressure.

The development of the mercury-bombardment thruster has continued through the 1960's to the present time. Thrusters 2.5 to 150 centimeters in diameter have been successfully tested. These thrusters require power of 50 watts to 200 kilowatts and produce thrust of 0.4×10^{-3} to 4 newtons (0.1×10^{-3} to 1 lb). Two of the most advanced bombardment thrusters, the 8- and 30-centimeter-diameter thrusters, are described in the sections AUXILIARY PROPULSION and PRIMARY PROPULSION, respectively. Thrusters of these two sizes fulfill the requirements of present-day missions.

Many laboratories in this country, Europe, and Japan have worked on a wide variety of electric thrusters. These include colloid thrusters using a dopedglycerine propellant, a pulsed-plasma thruster using ablation of a Teflon propellant block (ref. 2), and a bombardment thruster using cesium propellant. In Germany, France, and England, numerous laboratories and universities are at work on electric thrusters for both auxiliary and primary propulsion. The electric propulsion effort by the Soviet Union includes flights of Zond, Meteor, and Yantar spacecraft with ion thruster experiments onboard.

The mercury-bombardment thruster technology developed at the NASA Lewis Research Center has been used worldwide. England has developed the T-4 thruster based on this technology (ref. 3). The T-4 thruster is a 10-millinewton (2.2-mlb) thruster proposed as one of two possible ion thrusters to be flight tested by the European Space Agency in late 1980. The other thruster, the RIT-10, is a radiofrequency mercury-bombardment ion thruster developed by Germany. It has a similar thrust level of 10 millinewtons (2.2 mlb) (ref. 4). The Lewis technology has also been used by Japan. That country has built and tested a 5-centimeter-diameter, 5-millinewton-thrust, mercury-bombardment thruster for possible flight qualification in 1982 (ref. 5). Both the European and Japanese ion thrusters are proposed for auxiliary electric propulsion applications.

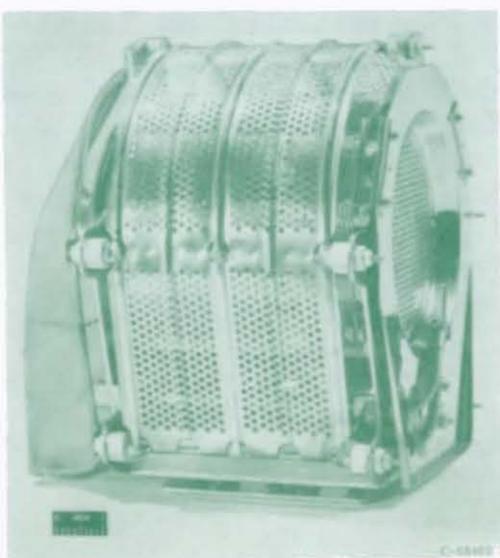

Figure 8.

Two spacecraft have been flown by the United States specifically for the purpose of testing ion thrusters in space. These tests, SERT I and SERT II, are described in the next two sections.

SERT I

Even though large vacuum tanks such as that shown in figures 6 and 7 provided an excellent simulation of an environment for testing ion thrusters, some questions could be answered only by operating ion thrusters in space. One such question was how would the ions and electrons exhausting from the thruster interact with space plasma, when the walls of a vacuum tank would no longer surround the exhaust.

On July 20, 1964, two ion thrusters were briefly tested in space by NASA. One was a mercury-electron-bombardment thruster (fig. 8) developed by the NASA Lewis Research Center; the other was a cesium-contact-ionization thruster developed by the Hughes Research Laboratories under NASA contract. This test was known as SERT I (Space Electric Rocket Test I). The battery-powered thrusters were

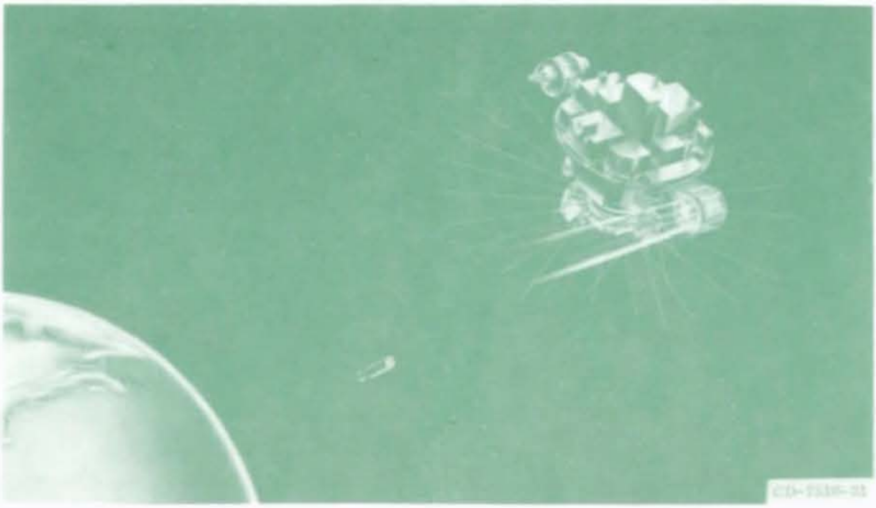

Figure 9.

mounted on a capsule that was launched with a Scout solid-propellant rocket into a ballistic trajectory (fig. 9).

These thrusters had been operated on the ground for hundreds of hours in vacuum tanks to measure their performance and qualify them for the space experiment. During these tests the cesium or mercury ions from the thrusters struck the walls of the vacuum tank and knocked many electrons loose from the walls (fig. 10). These electrons could have entered the exhaust beam and neutralized the ion space charge described previously. With the possibility of this neutralization occurring, it was difficult to tell whether the electrons from the thruster neutralizer were doing their job of neutralizing the ion beam. The primary purpose of SERT I, therefore, was to demonstrate neutralization in space and to measure any differences between ground and space operation. The direct evidence of incorrect neutralization would be a decrease in thrust from the predicted values.

Because electric rocket thrusters have only small thrust, the 50-minute SERT I ballistic flight would not have been long enough for the thruster to change the spacecraft trajectory enough to obtain an accurate thrust measurement. Therefore, the thrusters were mounted on arms so that their thrust would change the spin rate of the spin-stabilized spacecraft. The mercury-electron-bombardment thruster (fig. 8) onboard SERT I operated as predicted from vacuum chamber tests and produced thrust. Thus, the very important process of neutralization in space was proved possible.

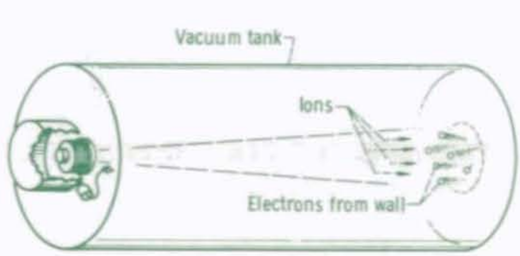

Figure 10.

The cesium thruster onboard SERT I did not operate, but on August 29, 1964, a cesium thruster produced thrust in a similar space test conducted by the U.S. Air Force. The Air Force test thruster was developed by Electro-Optical Systems of Pasadena, California. Small electrothermal, resistojet thrusters were successfully tested in space in the late 1960's. Small pulsed-plasma thrusters had also been tested in space (ref. 4). Thus, it had been established that electric thrusters would work in space and that their performance in space could be predicted from ground-based tests in vacuum facilities. The next step was to determine how durable the thrusters were. As mentioned previously, electric thrusters produce only a small amount of thrust; therefore, they must be capable of operating for long periods – several months to years in most applications.

SERT II

The SERT I flight verified the neutralization of an ion beam in space by showing the production of thrust, but it was a short flight using batteries for power. The purpose of the SERT II flight was to demonstrate long-term operation of an ion thruster in space with a flight-type power source.

On February 3, 1970, the SERT II spacecraft was launched by a Thorad/Agena launch vehicle into a circular polar orbit, as shown in figure 11. The polar orbit permitted the solar panels, which powered the thruster, to remain in continuous sunlight throughout an entire orbit. The solar panels, each 1.5 by 5.8 meters, are shown in figure 12 together with the complete spacecraft built onto the Agena stage.

Figure 13 shows one of the two SERT II thrusters. Each thruster was a 15-centimeter-diameter mercury-bombardment ion thruster and at full power used 850 watts to produce 28 milli-

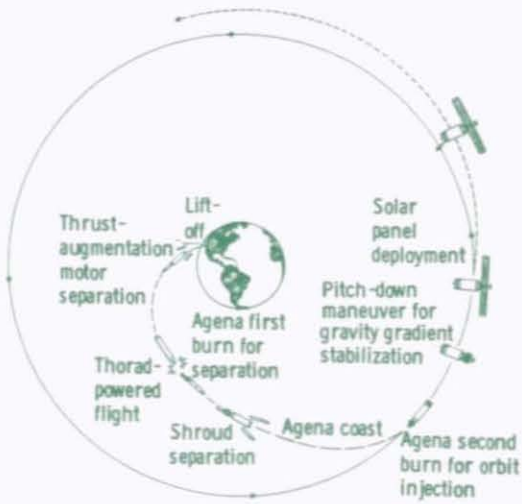

Figure 11.

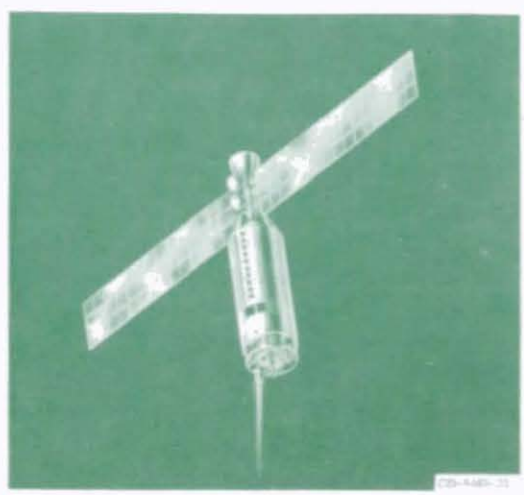

Figure 12.

Figure 13.

newtons of thrust. The thruster was also able to operate at 40 and 80 percent of full power.

Results of the SERT II flight include 5 months of successul operation with one thruster and 3 months with the other thruster (ref. 6). A minor thruster redesign as a result of follow-on ground tests (ref. 7) has provided future thrusters with design lifetimes of 15 to 30 months. Ground life tests of identical thruster power-processor systems were stopped without failure after 9 and 8 months, respectively, of continuous running. The thrust of the SERT II ion thruster was measured by an onboard accelerometer and by the change in the spacecraft's orbit. These two measurements of thrust and a thrust calculated from measured beam current and voltage produced identical values of thrust within experimental error.

Other experiments successfully performed were (1) a neutralizer cathode bias experiment in which the entire spacecraft was biased from 50 volts negative to 8 volts positive of space plasma; (2) probe sweeping of the ion beam, which showed a beam profile in space similar to those measured in ground tests (small beam divergence loss); and (3) a radiofrequency-interference measurement, which indicated no radiofrequency-interference ion beam noise detectable above the background noise level of Earth and thus gave evidence that stations on Earth will be able to communicate with future spacecraft without interference from thruster ion beam plasma noise.

The SERT II spacecraft and thruster system components have been operated periodically for over 7 years in space. One thruster system is presently fully operational (ref. 8), and most subsystems continue to function on the other thruster system. In 1979-1980 the spacecraft will again enter an orbit of continuous sunlight, and an opportunity will exist to continuously test the operational thruster.

Meanwhile, short (< 1 hr) periods of sunlight on the SERT II solar arrays permit brief testing of the thruster system and components. The thruster system was turned on in 1974, 1975, and 1976 and operated without problem. Two propellant flow systems and two hollow cathodes on each of the two flight thrusters have been started successfully over 200 times in space. Storage periods between starts ranged from 10 minutes to 350 days. Each thruster power processor has operated without incident for 4000 hours in space during this 7-year period. The spacecraft solar arrays and thermal control surfaces have shown no abnormal degradation due to contamination from thruster operation. The spacecraft attitude-control system and thruster gimbal actuators also continue to function correctly.

The SERT II flight has provided mission planners with important data needed to design space electric propulsion systems for long-duration missions. These data, combined with data from over 200 000 hours of thruster ground tests, provide a confident basis for evaluating thruster operating lifetime and thrust performance level and for designing future spacecraft so as to avoid contamination.

auxiliary propulsion

thruster system

Geosynchronous spacecraft often require careful control of their position and orientation over long periods of time. This is especially true of the highly directional antennas presently being used. Electric propulsion is ideal for the stationkeeping and attitude-control functions of geosynchronous satellites because of its high specific impulse, which results in considerable propellant mass savings over chemical propulsion systems, and because of its low thrust, which causes minimum perturbation or spacecraft "jitter."

An extensive research and development program to develop a mercury ion thruster specifically for auxiliary propulsion has been in progress for a number of years. The first-generation result of this effort, shown in figure 14, was a structurally integrated ion thruster (SIT-5) 5 centimeters in diameter. Durability tests of this thruster, which operated at 1.8 millinewtons (0.4 mlb), demonstrated an operational life of 9715 hours. At the conclusion of this test, a program was created to lengthen further the life of the auxiliary-propulsion mercury ion thruster. As a result of an

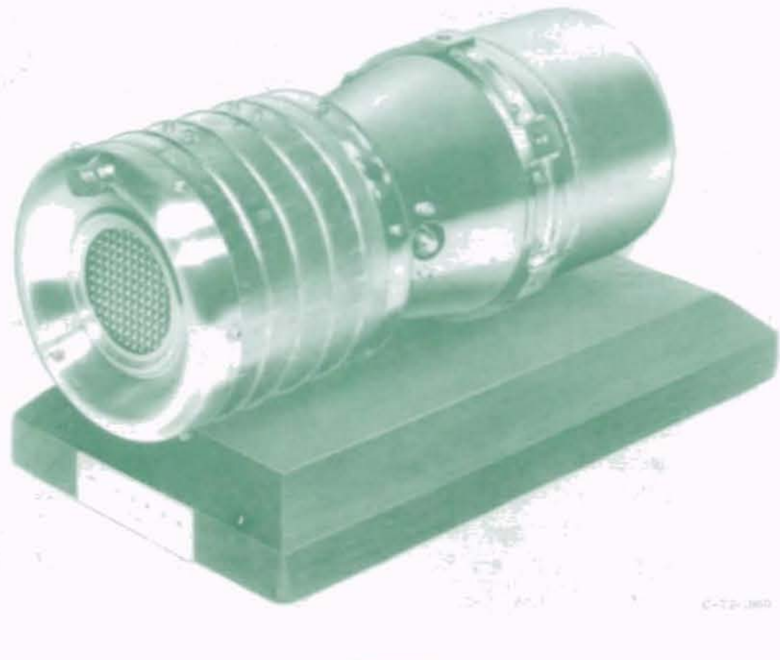

Figure 14.

analysis of mission requirements, an 8-centimeter-diameter ion thruster with a thrust output of 5 millinewtons (1.1 mlb) was chosen to be the most suitable for projected spacecraft masses and mission lifetimes.

The SIT-8 thruster system, which adapted and improved much of the technology of the SIT-5 thruster, is shown in figure 15. The thruster is mounted on a two-axis gimbal system to allow it to perform attitude-control functions such as momentum wheel dumping. The propellant reservoir is a separate component and may be varied in size or location as is most suitable to a particular spacecraft and propulsion requirement. The mass and performance of this 8-centimeter-diameter thruster system are given in table I.

A cutaway view of the SIT-8 thruster is shown in figure 16. Minimization of weight and power is accomplished through the use of only two vaporizers and a common potential propellant feed system, which is at spacecraft potential. The thruster employs a high-efficiency dished grid system and reliable high-voltage-pulse starting of both the cathode and the neutralizer. Sputter-resistant and long-life materials within the discharge chamber ensure the capability of a long lifetime.

A 15 000-hour test of an 8-centimeter-diameter thruster was completed successfully in 1976. This thruster was turned off and on approximately once per day, accumulating 460 cycles. A main cathode-vaporizer-isolator subsystem was successfully operated for 33 000 hours. The test was stopped for inspection. In an on-going test, an 8-centimeter-diameter thruster has been cycled 4500 times. At the time of this writing (April 1977) the thruster is undergoing 24 cycles per day. In other on-going tests, main and neutralizer cathode-vaporizer-isolator assemblies have been cycled 5700 times to date, and 12 additional cycles are added daily. These tests indicate a SIT-8 thruster system design lifetime in excess of 20 000 hours and 10 000 cycles.

The SIT-8 thruster, designed to operate at 5-millinewton (1.1-mlb) thrust, can easily be configured to operate at between one-half and two times this thrust level, depending on user requirements. Thrusters were successfully developed that demonstrated this range of thrust. This was accomplished by maintaining the basic propellant flow system and cathodes of the SIT-8 thruster and merely changing the diameter of the accelerator grids and thruster shell to increase or decrease the thrust level. The easy scalability of the thruster allows the spacecraft designer a range of options. These include the use of an efficient lightweight nickel-hydrogen battery for thruster power to permit higher-thrust-level operation in order to decrease the peak-power loading of solar array panels.

applications

The most important application of auxiliary-propulsion ion thrusters is north-south stationkeeping. Gravitational forces of the Sun and Moon tend to increase the inclination of the geosynchronous orbit, as shown in figure 17. Through the use of proper thrusting, centered about the nodal crossings, as indicated in figure 17, the geosynchronous orbit will not incline and will remain in the equatorial plane. The

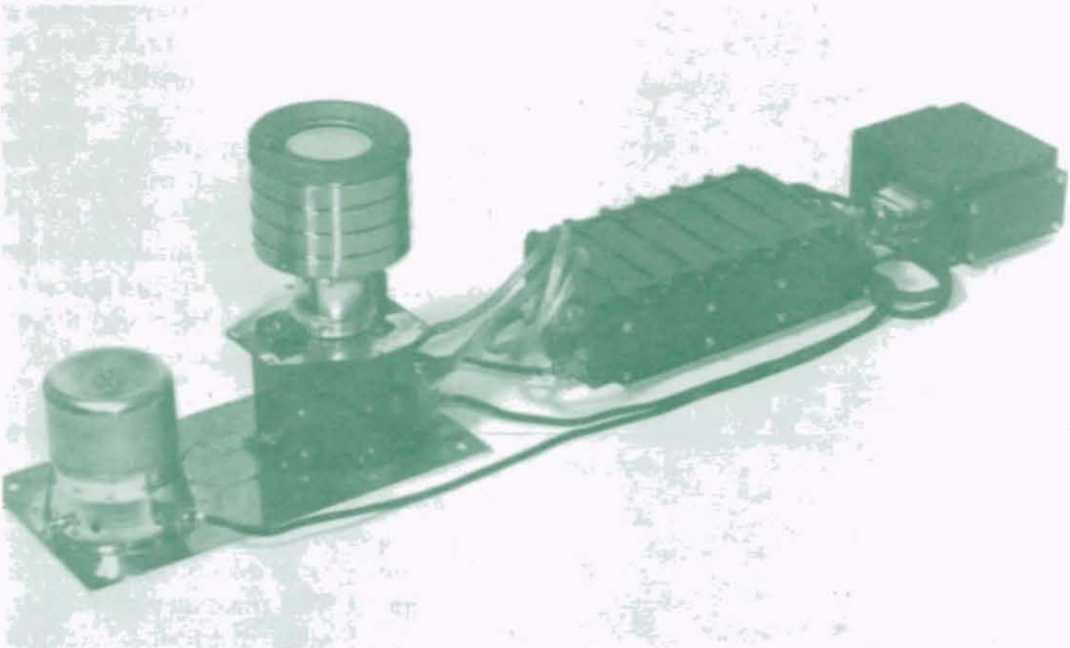

Figure 15.

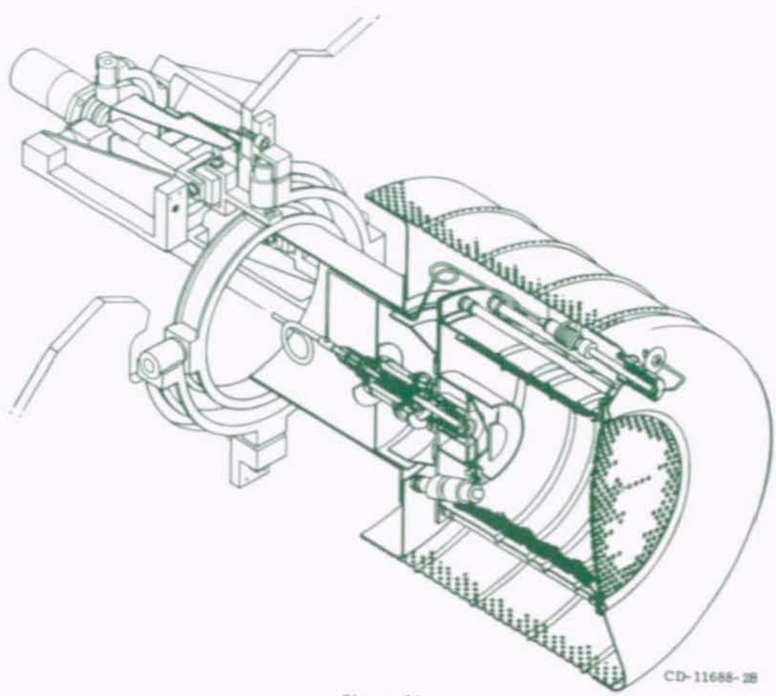

Figure 16.

duration of thrust each day depends on the spacecraft mass and thrust level and on how closely the thrusters can be alined with respect to a north-south line and still have a thrust direction through the center of mass of the spacecraft. Typical thruster orientations on a spinner spacecraft are shown in figure 18. Three-axis stable-platform spacecraft allow the thrusters to be alined directly north-south if they are mounted as shown in figure 19.

Effects of solar pressure and the triaxiality (gravitational nonuniformity) of the Earth will cause the east-west position of a geosynchronous spacecraft to vary unless thrust corrections are properly applied. The triaxiality of the Earth is a much weaker disturbing force than

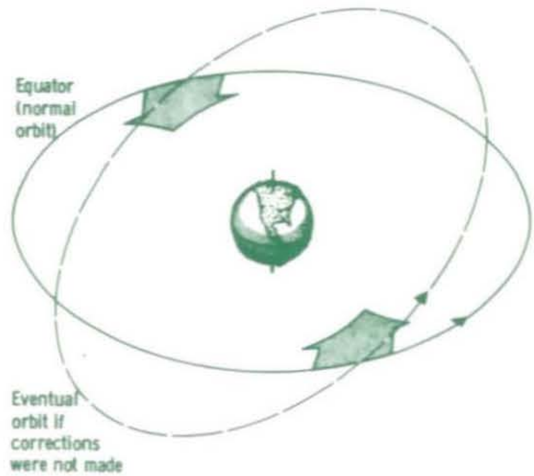

Figure 17.

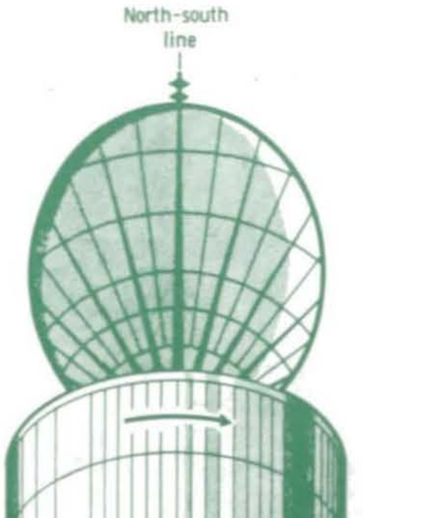

Figure 18.

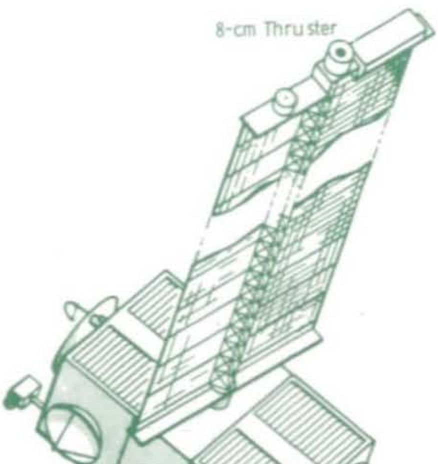

Figure 19.

that caused by the Sun and Moon and requires only 1/26 of the total impulse for proper correction. The thrusters shown in figure 19 can easily perform east-west stationkeeping through the use of the thruster gimbal system; their thrust is deflected in the correct direction and they are started simultaneously at specific times throughout the orbit. Station walking, or changing the east-west location of the geosynchronous spacecraft, can also be accomplished.

Because the center of solar pressure and the center of mass of the spacecraft are rarely the same point, the speed of flywheels, or momentum wheels, must be gradually increased in order to hold the spacecraft in the proper orientation. Electric thrusters can provide a counteracting disturbing torque, which can be used to dump or reduce the velocity of the momentum wheels. This reduction can help lower excessive momentum-wheel spin rates. Figure 20 shows how the thrusters can help despin the momentum wheel for one of the three axes of a spacecraft.

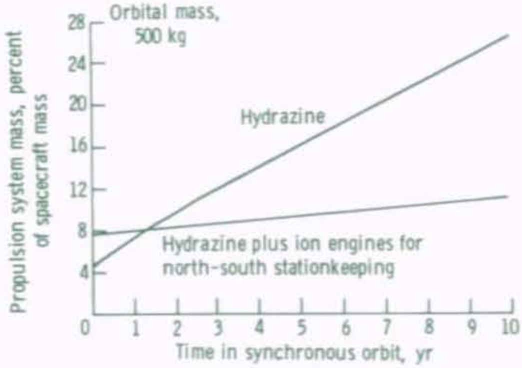

Figure 21.

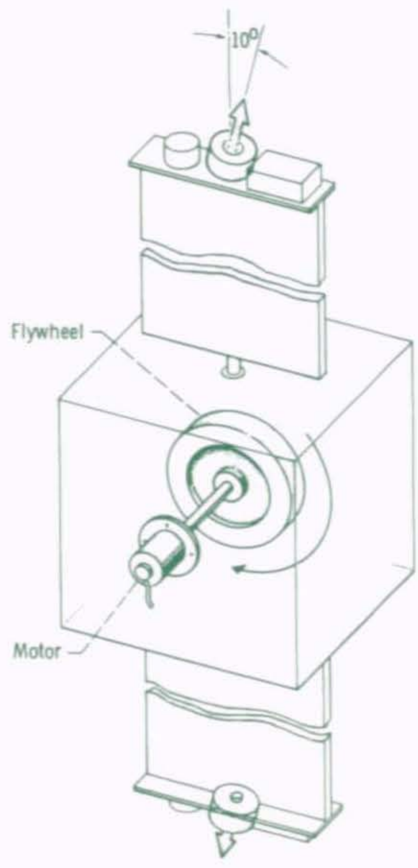

Figure 20.

benefits

Auxiliary electric propulsion can lower spacecraft weight when used in place of other types of auxiliary propulsion for geosynchronous spacecraft. Figure 21 compares weights of a mercury ion-thruster system for north-south stationkeeping with those for a hydrazine system. As can be readily seen, using electric propulsion can provide a significant weight saving on a long-mission-life spacecraft. A typical 347-kilogram (765-lb), geosynchronous, 7-year, spinner spacecraft would be 51 kilograms (112 lb) less if mercury ion thrusters were substituted for a hydrazine north-south stationkeeping system.

flight proof test

Two SIT-8 thruster systems are planned for flight test in 1980 by the U.S. Air Force. The thruster systems will be operated in a representative mode consistent with user needs and potential applications. A variety of diagnostic tests, including orbital change of the spacecraft for direct measurement of thrust magnitude, will be performed. Various probes will be used to precisely measure the low level of neutral and ion efflux expected from the thruster.

The spacecraft, containing other U.S. Air Force experiments in addition to the two ion thrusters, will be carried into low Earth orbit by the sixth orbital test of the space shuttle. A solid-stage booster rocket will place the spacecraft into an 830-kilometer (500-mile) circular orbit with an inclination in excess of 70°. The spacecraft will be three-axis stabilized and have sufficient solar array and battery capability for full-orbit testing of the thrusters.

primary propulsion

Electric propulsion for primary spacecraft thrust is of interest for both near-Earth and interplanetary missions. Near-Earth applications include the spiral-out maneuvers from low to high orbit and Earth escape. Once at high orbit the thruster may also be used for stationkeeping. Interplanetary missions include flights out of the ecliptic and flybys past, or rendezous with, asteroids, comets, and planets. A primary electric propulsion stage could offer large payload advantages as a commercial tug in conjunction with the space shuttle.

The interest in electric propulsion derives mainly from the reduction in propellant requirements relative to chemical propulsion due to operation at increased specific impulse. One way to compare the capability of an electric propulsion spacecraft with that of a chemical system is to consider the total impulse delivered by two such systems. A 1500-kilogram electric propulsion spacecraft with 500 kilograms of propellant can deliver slightly more total impulse than a 2914 Delta rocket stage, which has a mass of 5500 kilograms including 4500 kilograms of propellant. A typical electric propulsion spacecraft is shown in figure 19. Although exact comparisons are subject to details of the propulsion system configuration, the comparison just given is illustrative of the propellant savings, and, hence, overall mission performance increases, achievable with electric propulsion. To explain more fully the characteristics of a primary electric propulsion system, the major elements are discussed in this section.

thruster

Intensive tests and development are under way at the NASA Lewis Research Center and the Hughes Research Laboratory in the area of primary-propulsion thrusters. The candidate thruster for use on proposed missions is the 30-centimeter-diameter thruster (fig. 22). This thruster operates at a nominal input power of 2.75 kilowatts at a thrust of 0.135 newton and a specific impulse of 3000 seconds. Thruster efficiencies in excess of 0.71 are achieved at full thrust, with some decrease at throttle conditions. The thruster has been designed to throttle over more than a 4:1 range of input power and has a design goal lifetime in excess of 15 000 hours.

One of the first engineering model 30-centimeter-diameter thrusters was successfully endurance tested for 10 000 hours. Following the completion of this test in May 1975, small design changes were made to reduce the erosion of discharge chamber components, and the design life was increased to 15 000 hours. Over 25 000 hours of additional testing (longest test, 4000 hr) have been performed with the 30-centimeter-diameter thruster since 1972 to evaluate endurance capabilities. Endurance testing of an array of 30-centimeter-diameter engineering model thrusters is planned for 1978.

The thruster has been qualified both mechanically and thermally and is compatible with the launch environment of boosters ranging from Thor/Delta to Titan with a variety of upper stages. It is

Figure 22.

able to operate in the thermal environment associated with inbound and outbound interplanetary missions ranging from about 0.5 to 5 astronomical units. As verified in the SERT II flight, this thruster is capable of very long-term space storage and has highly reliable multiple-restart capability.

In the development of the 30-centimeter-diameter thruster, attention has concentrated on performance, reliability, and lifetime. The technology from the SERT II program has been used as a guideline. Several aspects of 30-centimeter-diameter-thruster design have, however, required technological advances in order to meet the demands of primary-propulsion missions. Improvements in thruster technology also have been added to the 30-centimeter-diameter thruster to extend its propulsion capability beyond that of the SERT II thruster. The basic hardware and operating characteristics are described in the appendix.

The accelerator grids of the 30-centimeter-diameter thruster are dished, or curved, as shown in figure 22, instead of flat as for the SERT II thruster. Using the dished shape has controlled thermal warping in the very closely spaced grids and has resulted in ion beams with higher thrust density at lower power. Adjustment of the hole patterns on both grids through new manufacturing techniques has reduced the thrust losses resulting from ion beam divergence to about 1 percent. The closer grid spacing of dished grids has strongly increased the thrust density (or current-carrying capacity) of the accelerator system. (The ion current densities achievable with dished thruster grids are significantly higher than any obtainable from large-area ion beam accelerators.) The broad current range of the accelerator system can be traded off against accelerating voltage to permit thruster operation over a wide range of specific impulse density.

Both the main cathode and the neutralizer cathodes are of the hollow cathode design developed for SERT II and are sized to accommodate the electron emission requirements of the 30-centimeter-diameter thruster (ref. 9). Many cathode life tests have been made, with test durations ranging from 5000 to over 21 000 hours and continuing. These tests, conducted at normal or higher than normal operation, have shown that the cathode lifetime and restart capability can meet the thruster design goals.

One arrangement generally useful on a primary-propulsion subsystem is the use of common tankage for all thrusters. This arrangement requires a flow-line device which can electrically isolate the thruster from the common propellant tankage system. Such an isolator has been developed, and many component and thruster life tests have indicated lifetimes well in excess of the 15 000-hour design goal. (This isolator is described in the appendix.)

power processing

Power processors for primary-propulsion thrusters have been under development for several years. Early tests defined the power supply characteristics and control logic systems required to operate over large ranges of power and thrust. Processors capable of operating in vacuum have been developed, and an example is shown in figure 23. These processors, which include 12 separate power supplies and associated control logic, have electrical efficiencies from 0.8 to 0.92 over the thruster operating range. Although these efficiencies are high, the large power requirement of the thruster (2.75 kW)

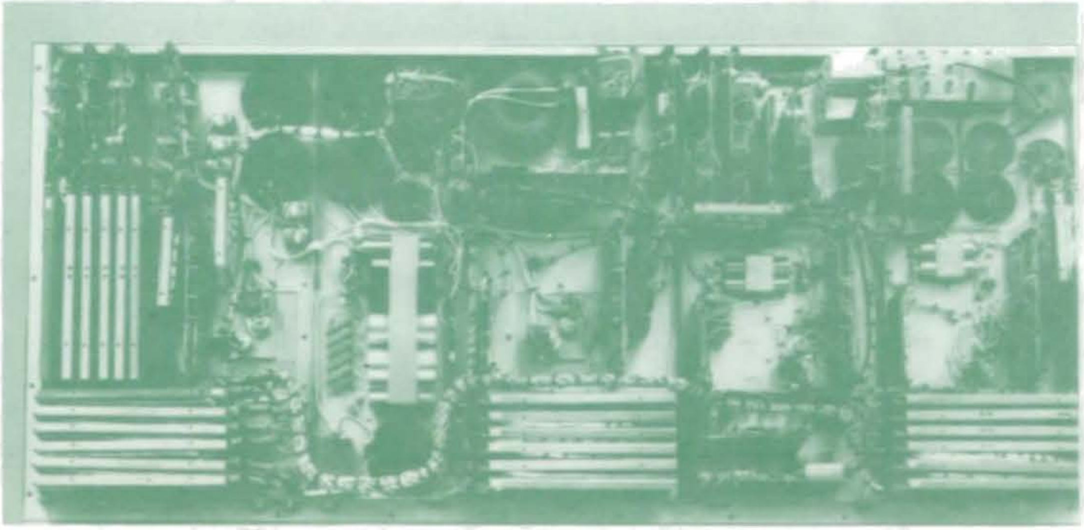

Figure 23

implies that significant amounts of heat must be removed from each power processor. Meeting the thermal control requirements of the power processor challenges the designer, and a number of studies have been made to formulate systems approaches. One candidate concept uses variable-conductance heat pipes similar to those used on the Communications Technology Satellite (CTS), launched in 1976. The power processors in the concept are buried within the spacecraft body, and their waste heat is transported by heat pipes to radiators for rejection to space (ref. 10). Figure 24 shows this concept, which among other features, such as eliminating the need for thermal-control louvers, provides excellent isolation of the power processors from the environment. And, thus, the impact of the multiple-mission capability requirement on spacecraft design is minimized.

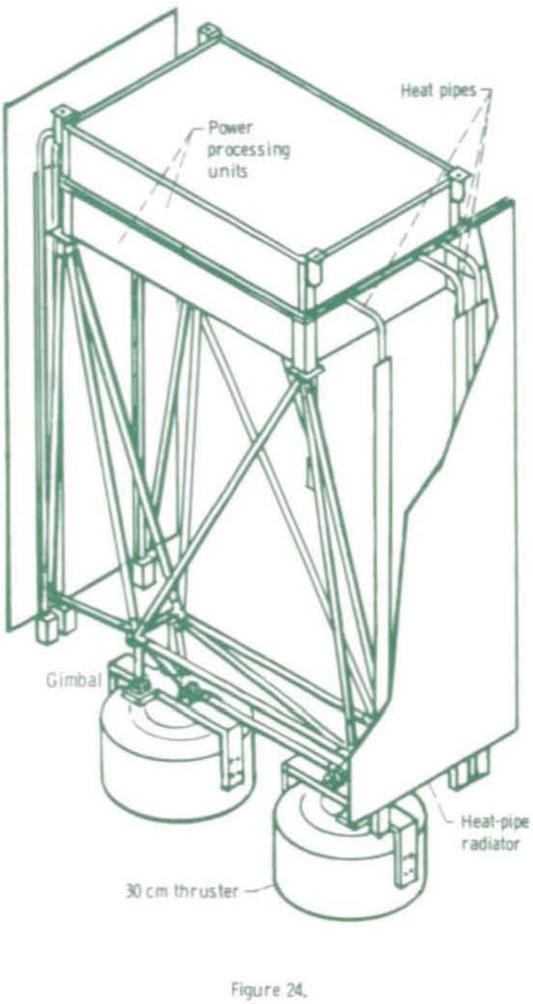

Figure 24.

thrust vector control

Thrust vector control of the 30-centimeter-diameter thruster will be available to provide spacecraft attitude-control and stationkeeping functions. This control allows development of the thruster as a module to be incorporated into a specifically designed spacecraft attitude-control system. A two-axis thruster gimbaling system is used in each module (fig. 24). Other concepts, such as gimbaling sets of thrusters, can also be used.

propellant supply and distribution system

The propellant supply and distribution system proposed for all missions is quite similar to that successfully flown on the SERT II mission. A common propellant supply, in one or more tanks, provides the propellant to all thrusters. Each thruster is turned on or off as required during the mission. A blowdown system provides the required pressure. Use of appropriate internal liners allows multiple-mission use for a single qualified tank design.

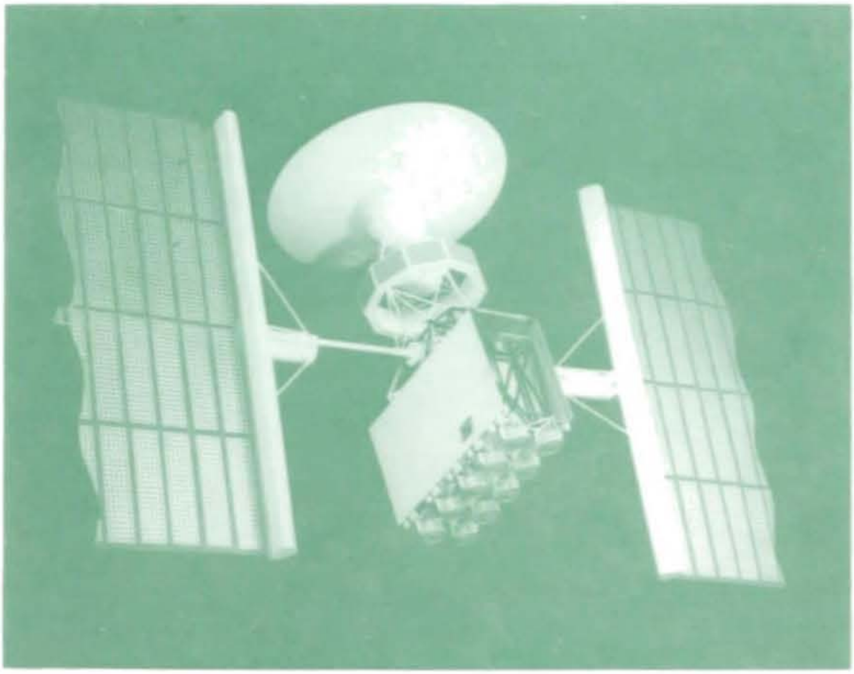

Figure 25.

comparison of thrusters

Table I lists the mass and performance of three thruster systems developed for flight. The 15-centimeter-diameter SERT II thruster system represents the state of technology in 1967. The 8- and 30-centimeter-diameter thruster designs have been effectively frozen since 1975, with only minor changes resulting from system life testing.

The 8-centimeter-diameter thruster system of table I is identical to that planned for flight testing in 1980, with the exception of the propellant tank size. A smaller tank is planned for the flight because the thruster on-time is less than the design life. The flight tank capacity is 8.75 kilograms of mercury (13 000 hr of operation), and the dry tank system mass is 1.16 kilograms. Further details of the developed thruster system are given in reference 11.

The 15-centimeter-diameter SERT II thruster system is the flight thruster, and its performance is based on flight data (ref. 6) . The thruster size, power, and specific impulse were designed to be representative of thrusters required for projected missions as conceived in 1966. In addition to the full-power operating point of table I, the thruster is capable of operating at 80 and 40 percent of full thrust. Complete performance data are given in reference 6.

The 30-centimeter-diameter thruster system has been designed with a 4:1 throttling capability. The two values listed in table I for the 30-centimeter-diameter thruster represent approximately half and full throttle. The design life and propellant mass are for the thruster operating continuously at full throttle. Further design details of the 30-centimeter-diameter thruster are given in reference 12. The total impulse values for each thruster system of table I are calculated by assuming continuous operation for a time equal to the design life.

TABLE I. — THRUSTER SYSTEMS PERFORMANCE SPECIFICATIONS

Parameter	Thruster diameter, cm		
	8	15	30
Thruster, mN(mlb)	4.9 (1.1)	28 (6.4)	65 to 129 (14.7 to 29)
Specific impulse, sec	2800	4200	2560 to 2900
Power (to PPU), W	160	1000	1600 to 3000
Thruster mass, kg	2.0	3.0	8.8
Power processor mass, kg	9.2	16.7	43
Gimbal mass, kg	1.4	7.7	3
Design life, hr (cycles)	20 000 (10 000)	6 000 (200)	15 000 (3 000)
Propellant mass (to operate for design life), kg	12.8	15.0	240
Propellant tank mass, kg	1.7	4.3	12
Total dry mass, kg	14.3	32	67
Total impulse of single thruster (design), N-sec (lb-sec)	360 000 (80 000)	610 000 (140 000)	7 100 000 (1 600 000)

appendix— thruster configuration and operating characteristics

A general description of a 30-centimeter-diameter thruster is presented. Because many characteristics of hollow-cathode electron-bombardment thruster geometry and operation are independent of thruster size, the description can apply to bombardment thrusters of any size.

Figure 26 shows a cutaway drawing of a 30-centimeter-diameter thruster. (Thruster size is specified by anode diameter.) Figure 27 shows a schematic drawing of the thruster and the major power supplies.

The mercury is introduced into the thruster discharge chamber and the neutralizer as a vapor. For a thruster using mercury as the propellant, the phase separator between the liquid propellant and the vapor is a vaporizer made of porous tungsten (fig. 27). This vaporizer blocks the liquid mercury and passes the vapor at a rate dependent on the vapor pressure (temperature) of the liquid mer-

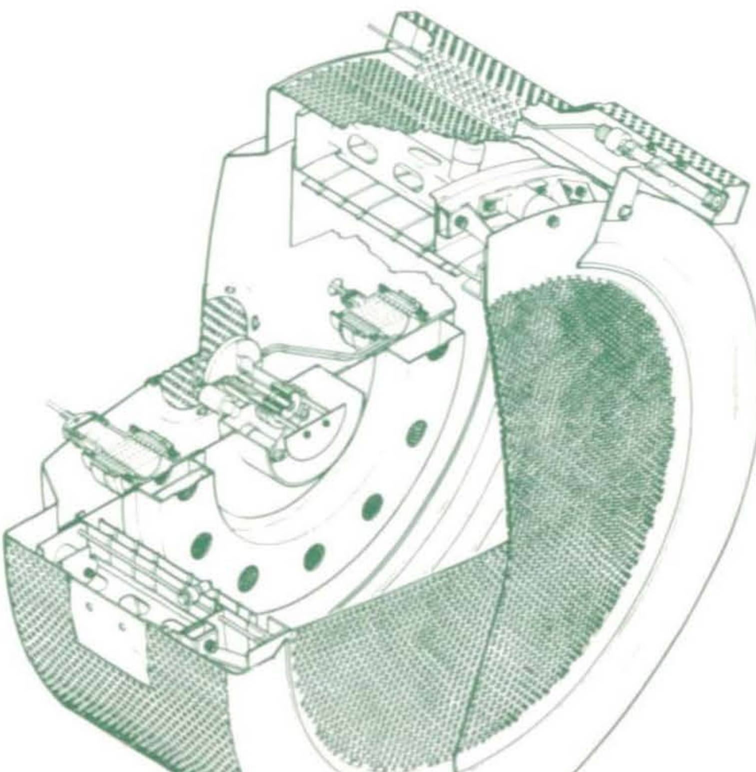

Figure 26.

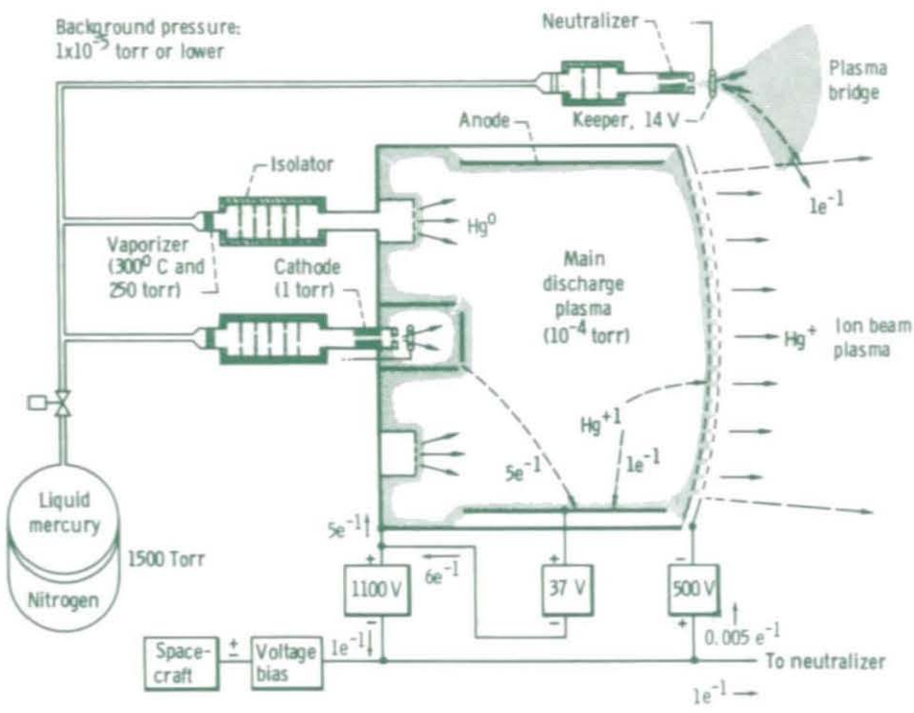

Figure 27.

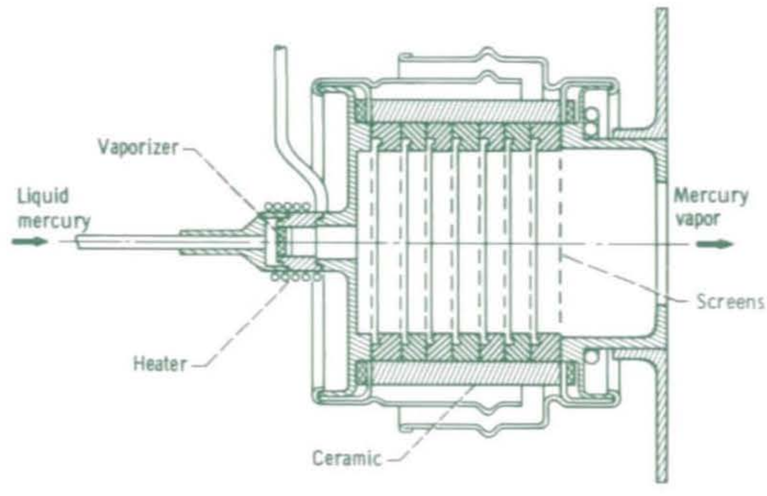

Figure 28.

cury. Typically, the vaporizers are used in a closed-loop control system to control the ion-beam current, the neutralizer discharge, and, in the case of the 30-centimeter-diameter thruster, the discharge voltage.

Propellant flow electrical isolators are located immediately downstream of the vaporizers. The isolators can be used to isolate electrically the high-voltage parts of the thruster from the main propellant system. As shown in figure 28 the vapor is passed through a series of metallic screens. The spacing of these screens is such that the minimum Paschen breakdown voltage for mercury is not obtained in any space. In this fashion, large voltage can be maintained simply by selecting the appropriate number of screens. This feature allows multiple-thruster operation for a common tankage system and provides large savings in propellant supply weight.

The mercury vapor is then passed into the discharge or the neutralizer cathode. (In the 30-cm thruster the bulk of the propellant is passed directly into the discharge chamber, while in the 8-cm thruster the bulk of the propellant is passed through the main cathode.) The cathodes are hollow (fig. 29). They consist of a tantalum tube with a welded tungsten plug containing an orifice attached to the end. The orifice size is scaled to the electron emission requirements. Inside the tantalum tube there is an insert which provides a supply of low-work-function material. This material (BaO) aids in the emission process. A separate discharge, higher in pressure than the main discharge, is established inside the hollow cathode: this discharge serves as a source of electrons.

Electrons are emitted from the cathode and pass through the discharge

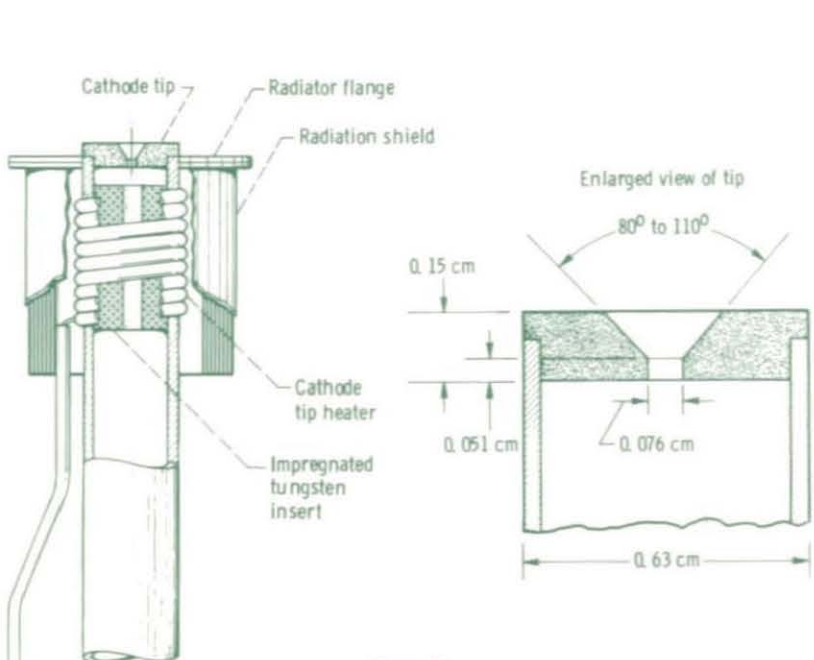

Figure 29.

chamber to the anode. For reasons of efficiency and lifetime, the applied discharge voltage is usually between two and four times the ionization potential of the propellant used. Propellant is ionized and excited by electron bombardment. An axially divergent magnetic field is maintained in the discharge chamber to improve ionization efficiency and to help control the voltage-current characteristics of the discharge. The magnitude of the magnetic field is controlled by either the number of permanent magnets or the current through the electromagnets. Typically, the optimum maximum magnetic field strength is such that an electron gyroradius calculated for the high-energy electrons in the discharge is less than the anode radius. The magnetic field is shaped by soft-ion magnetic pole pieces at the front and rear of the discharge chamber. Three fairly distinct plasmas are maintained: inside the hollow cathode, in the region between the cathode and the cathode baffle, and in the main discharge chamber. Sheaths are maintained between these plasmas and the thruster components and between the individual plasma regions. These sheaths have a strong impact on overall plasma characteristics and thruster performance. Some of the ions formed in the discharge are extracted by the multiple-aperture accelerator grid system (fig. 26). The grid systems presently in use are dished (fig. 30) and, as described previously, provide long-term stability and high performance. The energy of the ion beam is set by the positive high voltage applied to the anode, which is typically near the screen grid or cathode potential. The beam intensity is adjusted

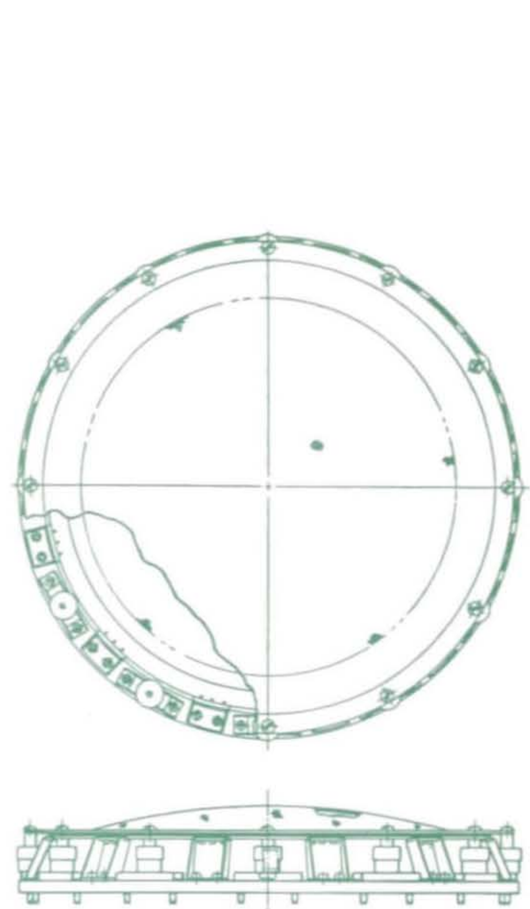

Figure 30.

by varying the discharge plasma density. The plasma density can be increased by increasing either the propellant flow or the discharge power. The negatively biased accelerator grid is placed downstream of the screen grid in order to help focus the ion beam and prevent electrons from being drawn back to the positively biased screen grid.

The neutralizer cathode emits an electron current equal to the ion beam and provides for the space charge and current neutralization of the ion beam. A perforated screen at ground potential is placed over the thruster body; this screen allows outgassing of thruster components and prevents interactions between the ambient plasma and the thruster body.

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