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PLANNED FLIGHT TEST OF A MERCURY ION AUXILIARY PROPULSION SYSTEM I - OBJECTIVES, SYSTEMS DESCRIPTIONS, AND MISSION OPERATIONS

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PLANNED FLIGHT TEST OF A MERCURY ION AUXILIARY PROPULSION SYSTEM

I - OBJECTIVES, SYSTEMS DESCRIPTIONS, AND MISSION OPERATIONS

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ABSTRACT

A planned flight test of an 8-cm diameter, electron-bombardment mercury ion thruster system is described. The primary objective of the test is to flight qualify the 5 mN (1 mlb.) thruster system for auxiliary propulsion applications. A seven year north-south stationkeeping mission was selected as the basis for the flight test operating profile. The flight test, which will employ two thruster systems, will also generate thruster system space performance data, measure thruster-spacecraft interactions, and demonstrate thruster operation in a number of operating modes. The flight test is designated as SAMSO-601 and will be flown aboard the Shuttle-launched Air Force Space Test Program P80-1 satellite in 1981. The spacecraft will be 3-axis stabilized in its final 740 km circular orbit, which will have an inclination of $\gtrsim$ 73 degree. The spacecraft design lifetime is three years.

INTRODUCTION

Previous flight tests of electron bombardment ion thrusters have demonstrated the operational feasibility of such thrusters for auxiliary and primary propulsion applications on a wide range of spacecraft and missions. The primary mission advantage of these thrusters lies in their > 2500 seconds specific impulse, which is nearly a factor of ten larger than available from chemical auxiliary propulsion. This high specific impulse results in substantial weight savings and performance improvements for many applications.

The SERT-I flight test in 1964 of a 10-cm mercury ion thruster successfully demonstrated thrust generation and ion beam neutralization in a 20-minute ballistic flight. The SERT-II flight test of two 15-cm mercury ion thrusters (ref. 1) has demonstrated the long term operational and restart capability of such thrusters in space and also confirmed ground test performance results. To date this flight test has accomplished a total of 6800 hours of operation and 215 restarts on the two thrusters, and one of the thrusters is currently being routinely restarted and

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operated after eight years in space. The SERT-II test also verified the expected thrust levels in space, to within measurement accuracy.

The ATS-6 flight test of two 4.5 mN cesium electron bombardment ion thrusters, launched in 1974, successfully accomplished spacecraft momentum wheel unloading and demonstrated effective control of the spacecraft potential during thruster operation.

Ground tests of mercury ion thrusters have accurately predicted thruster performance characteristics in space. Various facility-related effects in ground tests, however, can lead to ambiguity in determining the field and particle thruster-spacecraft interfaces and in predicting interactions between thrusters in representative, widely-spaced configurations on a spacecraft. In general, the ground tests yield worst case results. Because of this, extended spacecraft tests are vital for refining the ground test measurements.

FLIGHT TEST PERFORMANCE OBJECTIVES AND ACCOMPLISHMENT

A spaceflight test of two 8-cm mercury ion thruster systems has been approved by NASA and funded for launch in 1981. The flight test, sponsored by the U.S. Air Force Space and Missile Systems Organization and designated as SAMSO-601, has been accepted as one experiment to be flown on the P80-1 mission of the Air Force Space Test Program.

Flight Qualification Objective and Mission Model

The primary flight test objective is to flight qualify the 8-cm mercury ion thruster system for auxiliary propulsion applications. Typical auxiliary propulsion applications for the 8-cm thruster system include:

(1) stationkeeping (north-south and east-west);

(2) station change;

(3) atmospheric drag make-up in low orbits, and

(4) attitude control (both direct and by momentum wheel dumping).

In addition, the thruster (or its neutralizer) may be utilized for overall and differential spacecraft potential control (ref. 2).

A mission application study (ref. 3) has identified long-term north-south stationkeeping of advanced communications satellites in geostationary orbit as the most demanding auxiliary propulsion application. There is a well-defined and critical need for such a capability. A representative north-south stationkeeping mission was therefore selected to define the flight qualification requirement established for the SAMSO-601 flight test. The number of hours and cycles of thruster operation needed to meet the flight qualification objective depend on assumptions concerning:

(1) spacecraft mass;

(2) spacecraft mission lifetime;

(3) number of 5 mN thrusters provided on the spacecraft for stationkeeping;

(4) nominal cant angle of thruster thrust vectors from north-south direction; and

(5) north-south stationkeeping accuracy required.

These factors and their influence on the mission selected for simulation in the flight test are discussed below.

A spacecraft mass of 1000 kg in geostationary orbit has been adopted for the mission model. This is the estimated on-orbit mass of the Intelsat V spacecraft and represents the approximate payload capacity which can be delivered to geostationary orbit by the Atlas-Centaur launch vehicle or the Space Shuttle with a spinning, solid motor upper stage (ref. 4).

A spacecraft operational lifetime of seven years has been assumed for the mission model. This is consistent with present design criteria and anticipated nominal requirements for near-term communications satellites. A weight saving of 95 kg for a 1000 kg spacecraft has been shown (ref. 4) to result over a seven-year lifetime from the use of mercury ion thrusters for north-south stationkeeping, as compared with the use of electrothermal hydrazine propulsion.

A configuration of four body-mounted 8-cm thrusters has been identified (ref. 4) as the most advantageous thruster arrangement to accomplish north-south stationkeeping of a 3-axis stabilized geostationary satellite. This configuration, shown in figure 1, has been presumed for the SAMSO-601 mission model. In it, there are two nominally north-facing and two nominally south-facing thrusters. The most reliable and efficient north-south stationkeeping strategy with this configuration is to thrust with both north-facing thrusters about the north-going orbital node, then thrust with both south-facing thrusters about the south-going orbital node. This strategy equalizes the total operating time and number of cycles required for each of the four thrusters over the mission lifetime. The indicated configuration also permits the four thrusters to be positioned in any orientation relative to the spacecraft center of mass so long as the torques exerted on the spacecraft by the thrusters (operating in pairs) null out each other. Such a condition of torque cancellation may be maintained to any arbitrary accuracy throughout the mission by means of suitable gimballing adjustments of the thrusters. Furthermore, as may be seen from figure 1, appropriate operation and gimballing of single and of pairs of thrusters permits the four-thruster configuration to efficiently accomplish east-west stationkeeping and attitude control of the spacecraft as well.

Thrusting directly north and south is obviously the most efficient way to perform north-south stationkeeping. Nearly all 3-axis stabilized geostationary satellites, however, require large, rotating solar arrays to be mounted on the north and south spacecraft surfaces. Body-mounted north- or south-pointing thrusters could create an undesirable interaction problem with such arrays. Mounting the thrusters on the ends of the solar arrays would eliminate this problem, but creates other spacecraft design and interface difficulties.

A solution to the thruster-solar array interaction problem is to cant body-mounted thrusters at an appropriate angle from the north and south directions and use a thruster beam shield to prevent any deleterious interaction between the thruster and the solar arrays. This solution and configuration, as shown in figure 1, has been adopted in the mission model and implemented in the SAMSO-601 flight system. The thrusting efficiency for north-south stationkeeping is, of course, reduced to that given by the cosine of the cant angle.

Extensive diagnostic measurements have been made (ref. 3) on the neutral and charged particle efflux from the 8-cm ion thruster - with and without a beam shield. These and other tests have conclusively established that a cant angle of 45 degrees is feasible and conservative in eliminating deleterious thruster-solar array interactions, using a beam shield. A 45 degree cant angle has therefore been adopted to derive the mission model and a beam shield design chosen to prevent thruster efflux at angles greater than 45 degrees over an appropriate azimuthal sector.

A north-south stationkeeping accuracy requirement of ± 0.01 degree in the orbital inclination has been presumed for the mission model. Since the maximum north-south precession rate of the inclination of a geostationary orbit is 0.945 degree/year (ref. 4), a north-south stationkeeping correction is needed only about once every four days to meet the ± 0.01 degree accuracy requirement. With thrusters fired equally in both the north and south directions, thrusting would thus be required only about once every eight days for each thruster.

Each stationkeeping maneuver is, however, limited to 12 hours of continuous thrusting (1/2 orbit) about the appropriate orbital node. As the required length of each thrust period is extended toward this limit, thrusting efficiency is progressively lost in accomplishing the stationkeeping correction. Disregarding possible spacecraft-imposed constraints, a practical limit of ~9 hours per stationkeeping thrust period is reasonable. For the mission model so far defined, such a limit implies north-south stationkeeping maneuvers be performed at least three out of every four days. This schedule will amply meet the ± 0.01 degree stationkeeping accuracy requirement.

For the mission model, the assumption has been made that thrusting with each thruster is performed once per day at the appropriate node. This stationkeeping schedule limits the orbital inclination to $\sim \pm 0.0013$ degree, much more than meeting the mission model requirement.

Table I presents the adopted baseline mission model plus alternative models to show the effects of other possible values for each of the five assumptions. It gives, for each case, the resulting hours of nominal thrusting required in each stationkeeping correction, as calculated from reference 4. In the calculations a yearly incremental velocity requirement of 50 $m.sec.^{-1}yr.^{-1}$ for north-south stationkeeping was assumed. This value is essentially the maximum of the actual range of 40.1 to 50.7 $m.sec.^{-1}yr.^{-1}$ necessary, the precise value required depending on the specific year. Table I also gives for each set of mission model assumptions the resulting number of cycles and total hours of nominal thrusting required for each thruster. As seen in the table, the adopted mission model implies 2557 cycles, each with 2.76 hours thrusting, and a total operating time of 7055 hours for each thruster.

Table I in addition presents the implications of each mission model in terms of the SAMSO-601 operation necessary to demonstrate the model requirements. A fixed cool-down period of two hours (including the subsequent start-up period) was assumed in each operational cycle. The flight qualification objective of the flight test thus will be fulfilled by accomplishing, with one thruster, the total number of cycles and hours of nominal thruster operation shown in Table I.

The baseline mission model requires that pairs of 8-cm thrusters, in a spacecraft configuration such as indicated in figure 1, simultaneously thrust in either the north or the south direction. Such simultaneous operation of ion thrusters has never been attempted in a space test. Hence, a corollary requirement for the flight qualification is the demonstration of nominal performance by both of two thrusters simultaneously operated in a representative spacecraft configuration. This is one major reason why two thruster systems are included in the flight test. It is also a rationale for the basic configuration of the two thrusters in an orientation consistent with the adopted mission model.

Performance and Interface Measurement Objective

An important secondary objective for the SAMSO-601 flight test is the acquisition of critical design information on the performance and spacecraft interfaces of the 8-cm mercury ion thruster system. The spacecraft interface measurements to be made during the flight test are discussed in an accompanying paper (ref. 5).

Direct and indirect measurements will provide the in-space evaluation of thrust (F), specific impulse (I_{sp}), and total thruster system input power (P_{ps}). These measurements will confirm and assess the accuracy of corresponding 8-cm thruster system ground test results for these parameters, as subsequently presented. The formulas by which the thrust and specific impulse may be calculated from measured thruster parameters are

$$F(N) = 2.0391\times10^{-3} J_S \beta\gamma V_b^{1/2} \qquad (1)$$

and

$$I_{sp}(sec) = 100.02\ \eta_{Hg} \beta\gamma V_b^{1/2} \qquad (2)$$

where J_S is the current (in A) flowing through the screen supply, β is the fraction ($\lesssim 1$) correcting the apparent thrust based on J_S for the presence of a small percentage of multiply charged mercury ions in the primary ion beam, and γ is the fraction ($\lesssim 1$) correcting the apparent thrust based on J_S for the slight divergence of the primary ion beam from a true-parallel condition. V_b in equations (1) and (2) is the net beam accelerating potential (in V) and is given by

$$V_b = V_S + V_D - |V_c| \qquad (3)$$

where V_S is the screen supply output voltage, V_D is the main discharge voltage, and V_c is the coupling voltage between the neutralizer tip and the local space potential.

The term η_{Hg} in equation (2) is the utilization efficiency, corrected for the presence of Hg^{+2} ions in the beam. It is given by

$$\eta_{Hg} = (1.707\beta - 0.707) J_S/J_{Hg} \qquad (4)$$

in which J_{Hg} is the total propellant flowrate of the thruster (in equivalent A, assuming all singly charged ions).

The quantities J_S, V_S, and V_D are directly measured parameters in the 8-cm ion thruster system, and telemetered data will be available for them in the flight test. The quantity J_{Hg} is obtainable over short-term thruster operation from measurements of the main and neutralizer vaporizer temperatures, T_{MV} and T_{NV}, using ground-determined flow calibration curves for the vaporizers. Over long-term thruster operation J_{Hg} is derivable from the change in the measured propellant pressure, p_{Hg}, at the measured propellant reservoir temperature, T_{Hg}.

The coupling voltage V_c is not directly measured by the thruster system but is given by

$$V_c = V_g + V_{sc} \tag{5}$$

where V_g is the voltage difference from the spacecraft common potential to the neutralizer tip or neutralizer common potential and V_{sc} is the spacecraft common potential relative to the space plasma potential. V_g is a directly measured thruster system parameter and available in the thruster system telemetry data as above. V_{sc} will be measured at frequent intervals during the flight test by a spacecraft potential sensor included as one of the diagnostic instruments for spacecraft interface measurements (ref. 5).

The correction factors β and γ have been carefully measured in ground tests of the 8-cm thruster under nominal operation. Though these correction factors due to thruster beam divergence and Hg^{+2} ion content will not be evaluated during the flight test, there is no reason to expect them to significantly differ from the values measured in the ground tests.

The total thruster system input power, P_{ps}, will be available during the flight test from telemetry data giving the input current, J_{ps}, and input voltage to the thruster system power processor.

The thrust, specific impulse, and total thruster system input power will be determined in the flight test over the whole course of the flight-qualification cyclic thruster operation. Thus, the effects of cyclic, long duration operation on these performance parameters will be evaluated.

Ground life tests have provided an important information base on the effects of extended thruster operation and have led to important design and operational improvements (ref. 6) to remedy clearly defined durability deficiencies of the auxiliary propulsion mercury ion thrusters. But, for reasons such as discussed in reference 7, ground life tests can only reproduce to a limited extent the thruster effects of long-term cyclic operation in space. The telemetered thruster data from the SAMSO-601 flight test, especially from the thruster system operated to meet the flight qualification objective, will permit evaluation of such effects due to any space-specific processes.

A direct demonstration and measurement of the thrust generated by the 8-cm thruster system may be made late in the SAMSO-601 flight test by measuring the change in the spacecraft orbital altitude caused by a period of continuous thrusting. Such a thrust demonstration and measurement was accomplished with the 15-cm mercury ion thrusters in the SERT-II mission (ref. 8). The SERT-II direct thrust measurements and careful ground test measurements with a 30-cm thruster (ref. 9) have confirmed the accuracy of thrust determinations made from electrical parameter readings and known thrust losses, but the authenticity of direct thrust measurements made in space is unquestionable and may be important to potential users of the 8-cm thruster system. The subsequent discussion of SAMSO-601 mission details treats the specifics of an intended direct demonstration and measurement of the thrust of the 8-cm thruster system during the flight test, employing the orbit change technique.

Thruster system start-up requirements and reliability are of substantial interest and importance to users of the 8-cm thruster system. Therefore, the demonstration of routine thruster start-ups within a prescribed, relatively short time limit is considered part of the performance measurement objective of the flight test. A time limit of 15 minutes has been adopted for thruster start-up, from an ambient, cold, off-condition to full nominal thrust generation. This duration is a somewhat arbitrary estimate of a user-acceptable thruster starting specification. To address possible user requirements for a more rapid turn-on of thrust, alternative thruster operating modes which make this possible will be demonstrated in the flight test. These alternative modes are included and described in Table II.

The simultaneous dual thruster operation required under the flight qualification objective may influence observed thruster performance. In particular, steady state effects on the neutralizer coupling voltage or on the accelerator drain current of either thruster are possible and will be carefully investigated. Any coincident instabilities and/or extinctions in the two simultaneously operating thrusters, and any anomalies in one thruster system's operation coinciding with normal transient events in the other, will also be identified. The interactive effects from simultaneous dual thruster operation will, in addition, be evaluated by means of the data from the diagnostic instrumentation included in the flight test.

A so-called "neutralizer switch" is incorporated in each of the SAMSO-601 thruster systems. This switch permits each thruster to be operated with the thruster ground (i.e., the neutralizer common or tip potential) either shorted to or isolated from the spacecraft common. The isolated configuration is nominally considered to be the standard operating configuration. The neutralizer switch has a direct effect on the spacecraft potential and thereby may impact thruster system performance. However, it is not expected that any such performance effects with one thruster in operation will be major or troublesome, and effects may not even be observable in the thruster telemetry data.

With both of the SAMSO-601 thrusters in operation and both neutralizer switches in the closed position, the value of V_c for both thrusters will be constrained to be the same. In case the normal coupling potentials of the independently operating neutralizers are significantly different, this constraint on V_c may substantially alter the operation and control characteristics of both neutralizers during dual thruster operation. Any such interactive effects or other consequences arising from thruster operation with the neutralizer switch either open or closed will be investigated.

Table II describes several alternative thruster operating modes which are of potential importance in auxiliary propulsion applications. These will be demonstrated and evaluated during the flight test, under the thruster performance measurement objective. A self explanatory short description of the operating conditions and utility of each of the alternative operating modes is given in the table.

SAMSO-601 FLIGHT TEST THRUSTER SYSTEM AND MODIFICATIONS INCORPORATED

The two thruster systems to be employed in the SAMSO-601 flight test are each identical to the presently developed 8-cm engineering model thruster system, with four modifications. These modifications include a thruster beam shield, neutralizer switch, propellant feedline valve, and digital controller and interface unit. The engineering model system and the justification for, requirements of, and experience with each of the modifications to it for the flight test are described in this section.

8-Cm Engineering Model Thruster System

The 8-cm engineering model thruster system (EMTS) integrates an 8-cm mercury ion thruster and its power processor in a spaceflight-compatible configuration. It represents the current stage in the evolutionary development (refs. 10-12) of small mercury ion thrusters for auxiliary propulsion applications. This development has been carried out principally by the NASA-Lewis Research Center and by the Hughes Research Labs (under contracts from NASA-Lewis). It has included substantial effort in designing, optimizing, fabricating, testing, and verifying both the thruster and the power processor required for a flight system.

The Hughes-built 8-cm EMTS and its performance have been extensively described (refs. 13-15). A block diagram of the system is shown in figure 2 and a photograph of the assembly is seen in figure 3. It consists of five components: (1) the thruster, (2) a gimbal, (3) a propellant reservoir and feed system, (4) a power electronics unit, and (5) a digital interface unit. A cutaway diagram of the thruster is presented in figure 4, and measured nominal values of its performance parameters are given in Table III.

Some important design limits, capacities, and measured characteristics of the EMTS gimbal and the propellant reservoir and feed system are summarized in Table IV. The gimbal permits vectoring of the thrust up to 10 degrees in any direction from the nominal axis. The propellant reservoir is sized for ~12,000 hours of nominal, cyclic thruster operation.

The EMTS power electronics unit (PEU) conditions the main input bus power (provided at 70±20 V DC) and distributes it to the nine power supplies required for thruster operation plus the two high voltage pulse igniter circuits utilized to start the main and neutralizer cathode discharges. Table V presents the maximum and nominal output power characteristics and other design specifications for these supplies. The PEU also contains sense circuits to measure and output selected analog telemetry signals.

The digital interface unit (DIU) of the EMTS operates on 28±1 V DC input power and serves several control and interfacing functions which are detailed in reference 15. Among these are to make available, as 8-bit digital telemetry output words, the EMTS operating parameters listed in Table VI. The table also indicates the parameter range presently established for each telemetry channel. (Data words from all of these telemetry channels will be collected every 30 seconds during thruster operation in the SAMSO-601 flight test.) Some important electrical design specifications and characteristics of the PEU and the DIU are presented in Table VII.

Table VIII gives a definition of nominal operation, or the nominal operating point, for the 8-cm EMTS thruster parameters. This definition will apply to the thruster operation during the flight test. The thruster design of the EMTS has been optimized (ref. 15) to achieve a wide margin of static and dynamic stability in thruster operation about the nominal operating point. The 6 mA (equivalent) neutralizer flowrate established for nominal operation was chosen to avoid long-term neutralizer erosion problems (ref. 16) and provide an adequate control characteristic of neutralizer keeper voltage versus flowrate.

EMTS testing to date has demonstrated functional system operation. Such testing required real-time manual control via commands externally transmitted to the DIU. The EMTS does not provide for automatic start-up, operation, and shut-down of the thruster. For the flight test the DIU will be modified to incorporate these capabilities, as subsequently discussed.

Thruster Beam Shield

A beam shield will be incorporated on both thrusters in the flight test. A detailed description of this shield is to be published (ref. 17). The physical design of the shield, shown in figure 5, assures that the shield intercepts all possible ion and neutral particle trajectories from the thruster grid system which diverge by ≥ 45 degrees from the thruster axis, over an azimuthal sector of 120 degrees. A photograph of the shield, as mounted on an 8-cm engineering model thruster, is shown in figure 6. Because the thruster neutralizer is an efflux source, the beam shield is mounted on the thruster so that the neutralizer is centered within the azimuthal shielding sector.

The beam shield is made from a graphite fiber-polyimide resin composite developed at the Lewis Research Center (ref. 18). Some properties of this material are summarized in Table IX. It was chosen on the basis of its superior strength-to-weight ratio, its high graphite content, its thermal and electrical characteristics, its excellent vacuum compatibility and low out-gassing properties (as post-cured), and its relatively easy fabrication into one-piece formed structures. The interior surface of the beam shield is microscopically rough, insuring good adhesion of any coating deposited during thruster operation.

As mounted on the thruster ground screen, the flight design beam shield will be electrically isolated from the ground screen by > 1 MΩ resistance. The potential assumed by the floating beam shield during nominal thruster and neutralizer operation has been observed to be a fraction of a volt positive from the neutralizer tip potential.

Possible differential charging of the electrically isolated beam shield has been examined, though such charging during the SAMSO-601 flight test is not likely because of the density of the ionospheric plasma existing at the 740 km altitude of the P80-1 spacecraft. Differential charging between the beam shield and the thruster ground screen could cause arc breakdowns between these two components. Potentially the most severe differential charging conditions are the steady-state conditions which could exist during a magnetic substorm with the thruster off, the spacecraft in sunlight, and the thruster and its beam shield totally shadowed by the spacecraft. Under such conditions the spacecraft ground and the thruster ground screen potentials would be maintained near the ambient space plasma potential by photoelectric emission from the spacecraft's sunlit surfaces. At the same time, however, the floating, shadowed beam shield would be free to charge up to large negative potentials due to high energy electron impingement.

The magnitude of the charging problem has been estimated from the applicable charging model equation in reference 19. For mounting resistance values of 1 and 5 MΩ, the model predicts steady-state differential voltages of 12 and 60 V, respectively. Voltages of this magnitude will not overstress the mounting insulation. Hence, the specification of 1 to 5 MΩ mounting resistance will prevent any beam shield charging problems, even under the worst environmental charging conditions.

Testing of 8-cm engineering model thrusters equipped with the flight design beam shield has demonstrated (refs. 20 and 21) that the shield does not significantly affect thruster performance. The only effects of the shield fairly definitely established by this testing are an ~1-2 V increase in the magnitude of the neutralizer coupling voltage and an increase of similar magnitude in the thruster ion beam floating potential. No tests have shown any skewing of the thruster's thrust vector direction due to the beam shield.

Extended engineering model thruster operation with the flight design beam shield has demonstrated that the entire interior surface of the shield is subject to net deposition of sputtered material during normal thruster operation. The material deposited on the shield is indicated to be nearly all molybdenum, sputtered from the thruster's accelerator grid.

Two possibly undesirable characteristics of the beam shield are that its electrical surface conductivity is not uniform and that it may get cold enough under eclipse conditions, even during thruster operation, to retain some condensed mercury. Both of these conditions can be obviated by coating the shield surfaces with thin metallic coatings having appropriate electrical and thermal properties. Initial tests have indicated that highly adherent coatings with the required properties can be produced on the shield surfaces (ref. 17).

Neutralizer Switch

The EMTS grounding configuration provides for isolation of the neutralizer common from the spacecraft common potential. However, zener diodes in the EMTS circuit prevent the neutralizer common potential from differing (positively or negatively) from the spacecraft common potential by more than 150 V.

For the SAMSO-601 flight test, the thruster system will be modified by the incorporation of a commandable relay, called a "neutralizer switch," in the thruster grounding circuit. The two positions of this relay provide: (1) thruster system grounding according to the normal EMTS configuration or (2) direct shorting of the neutralizer and spacecraft commons. The grounding diagram for the flight thruster systems, including the neutralizer switch, is depicted in figure 7. The switch is operated from the 28 V spacecraft power bus by discrete commands from the spacecraft command system and will be physically mounted at the base of the gimbal unit.

The primary reason for incorporation of the neutralizer switch in the flight test thruster systems is that users of the 8-cm thruster system may require either of the two grounding configurations in their applications.

Thruster operation with the neutralizer common isolated from the spacecraft common results in:

(1) isolation of the spacecraft common from the electromagnetic noise generated in the neutralizer and main discharges of the thruster;

(2) no first order effects on the spacecraft common potential, V_{sc}, due to neutralizer operation or degradation, so long as the neutralizer maintains full neutralization of the thruster ion beam; and

(3) no ground loops through the spacecraft common when two thrusters are operating simultaneouly.

On the other hand, thruster operation with the neutralizer common shorted to the spacecraft common permits:

(1) active control (during thruster beam, main discharge, or neutralizer operation) of the spacecraft common potential, V_{sc}, at -10 to -30 V relative to the ambient space plasma potential; and

(2) elimination of the neutralizer isolator in the thruster design and many parts in the thruster system PEU and DIU, parts presently required for isolation of the neutralizer common from the spacecraft common.

With maintenance of normal thruster operation in the grounded configuration, the spacecraft common potential should remain essentially constant, despite exposure of the spacecraft to environmental charging episodes. This has been demonstrated by data from the ATS-6 mission (ref. 2), obtained during operation of the cesium ion thrusters.

The anticipated and possible effects of the neutralizer switch on thruster system performance in the SAMSO-601 flight test have been pointed out previously.

Propellant Feedline Valve

The 8-cm EMTS propellant supply and distribution system contains no mercury flow control and containment means other than the main and neutralizer vaporizer plugs. Such single containment does not meet the Space Shuttle cargo requirements, which prescribe double containment of potentially hazardous pressurized fluids. Therefore, in each of the flight thruster systems a commandable latching valve has been incorporated in the mercury feedline to the thruster. Throughout the launch, Shuttle bay separation, orbit acquisition, and spacecraft stabilization phases of the P80-1 mission, the feedline valve will remain closed with no mercury in the thruster or in any feedline section downstream of the valve.

Such use of a commandable latching valve in the propellant feedline provides the user of the 8-cm thruster system several important advantages besides meeting Shuttle requirements. These advantages include the following.

(1) No restrictions need be placed on the propellant tank-to-vaporizer configuration of the feed system in the spacecraft or on the acceleration forces experienced by this system during launch, orbit acquisition, or spacecraft stabilization. Without the feedline valve such restrictions are necessary in order to prevent the development at the vaporizers either of mercury pressures high enough to intrude them or of pressures low enough to cause the mercury to retreat from the vaporizer plugs.

(2) The feedline valve allows a field joint to be made in the propellant feed system during spacecraft installation of the thruster system and permits the feedline between the valve and the gimbal unit to be adjusted in length and conformation as required during final spacecraft integration. Such flexibility can be of great assistance in facilitating spacecraft integration of the thruster systems with a wide variety of spacecraft configurations.

(3) The latching feedline valve provides the potentially important capability of positively shutting off mercury flow to the thruster in the event of a feedline break downstream of the valve or in the event of a gross vaporizer leak.

(4) The latching valve permits the possible bake-out and restoration to normal performance of an intruded vaporizer in space.

Digital Controller and Interface Unit

The DIU of the 8-cm EMTS does not have the capability of automatically starting and operating the thruster. In order to provide the necessary automatic control and operation of each thruster system during the flight test, a controller unit is required. Such a unit also allows the highly desirable implementation of a very simple command and data interface between the thruster system and the spacecraft.

The minimum functional requirements which the SAMSO-601 thruster system controller must meet include:

(1) accepting discrete and magnitude commands, including command timing information, via the command interface with the spacecraft;

(2) controlling operation of the PEU and thruster-gimbal unit by outputting commands and reference level signals to these units via the DIU interface;

(3) effecting operation and control of the ion thruster system in accordance with the commands received from the spacecraft, by means of stored program segments;

(4) accepting, via the DIU interface, digital telemetry data words which convey the status and performance of the PEU and thruster-gimbal unit,

(5) establishing, maintaining, and restoring nominal performance of the thruster system over its mission lifetime through appropriate programmed response to the thruster system telemetry information received (including any indicated thruster aging effects);

(6) outputting spacecraft-compatible telemetry data words to the spacecraft telemetry system, on interrogation, to convey the status and performance of the thruster system;

(7) outputting appropriate commands to protect the thruster system and spacecraft in the event a thruster system malfunction is sensed; and

(8) providing the capability for direct ground command control, via the spacecraft command system, of PEU and thruster-gimbal unit operation.

A controller will be integrated with the present DIU of the EMTS to create a new unit called a digital controller and interface unit (DCIU) for the flight thruster systems. The parts count in the DCIU is expected to be slightly lower than in the DIU and the complete DCIU structure will be the same size, weight, and configuration as the present DIU (fig. 3).

The DCIU will employ a low-power-demand central processing unit (CPU) and will have a memory consisting primarily of read-only memory (ROM) but containing some random access memory (RAM). The other major new hardware components or chips required in the DCIU include: (1) a command logic unit to receive and isolate the serial commands from the spacecraft command system; (2) a telemetry logic unit to output telemetry words to the spacecraft telemetry system on interrogation; (3) a digital multiplexer to convert analog signals from the PEU; (4) an input/output central logic unit to provide address reset and strobe signals for input and output functions; (5) circuitry to provide internal timing for all logic functions required in the DCIU; and (6) a device to detect CPU anomalies and, in the event of one, safely shut down the thruster and PEU and reset the CPU. All the other circuits and hardware components required in the DCIU will be the same as presently employed in the DIU.

The automatic, programmed operation of the thruster system by the DCIU will function much the same as in the microprocessor-controlled, automatic cyclic operation previously demonstrated (ref. 22) of a prototype engineering model 8-cm thruster with the thermal-vacuum breadboard prototype of the engineering model PEU and DIU. When power is applied to the DCIU, the program will cause all setpoints, reference word values, timing constants, and branching decisions utilized throughout all program segments stored in the ROM to be read and stored into RAM locations. As these setpoints, reference word values, timing constants, and branching decisions are required during program execution, the CPU will fetch them from RAM. This permits any replacement values desired or required for these quantities during the course of the flight test to be transmitted by ground command to the DCIU, via the spacecraft command system, and written over the appropriate words in their respective RAM locations. Through suitable ground command alteration of the branching decisions executed, any available alternative operating mode, alternate transition between operating modes, change in the telemetry data outputted to the spacecraft telemetry system, or direct command to the PEU or thruster-gimbal unit may be implemented. Similarly, the setpoints, reference word values, timing constants, or branching decisions in these alternate program segments themselves may be changed by appropriate ground command.

The DCIU software will provide proven algorithms to reliably and automatically accomplish the cyclic start-up, nominal operation, and shut-down of the associated thruster for the number of cycles and total hours of operation necessary to meet the flight qualification mission requirements earlier discussed. This programming will accommodate normal thruster aging effects. The start-up algorithm will reliably achieve nominal thruster operation within 15 minutes from an initial cold, ambient condition of the thruster.

Table X lists a number of thruster conditions which may constitute potential hazards to either the thruster system or to the spacecraft. These conditions are revealed by the thruster system telemetry and/or by threshold sensors and a high voltage recycle counter in the DCIU. The threshold sensors and recycle counter all output discrete interrupt flag bits to the DCIU digital multiplexer in the event their out-of-limit conditions are sensed. The DCIU will be programmed to continually monitor the thruster system telemetry data and the interrupt flag bits during the flight test for the occurrence of these out-of-limit conditions. Should any be detected, the DCIU will automatically effect corrective action to safeguard the spacecraft and the thruster system. Normally, the corrective action taken will be to shut down the thruster system and await a ground-commanded restart.

The DCIU will also provide algorithms for stably operating the thruster system in and transferring to and from each of the alternative operating modes described in Table II, after appropriate ground command. Finally, the DCIU will provide automatic sensing of, and corrective algorithms for, those thruster system anomalies among the ones listed in Table XI which are of significant probability (such as neutralizer extinction) or which can be relatively easily cured or avoided. Failure detection and correction in the DCIU may also be extended to sensing abnormal, undesirable thruster operating modes and recovering from them to normal operation.

P80-1 SPACECRAFT AND MISSION DESCRIPTION; SAMSO-601 SPECIFICATIONS, CONFIGURATION, AND MISSION OPERATIONS

P80-1 Spacecraft and Mission Description

A preliminary concept of the P80-1 spacecraft on-orbit configuration is shown in figure 8. The other two experiments to be flown on the spacecraft, in addition to the SAMSO-601 mercury ion thruster experiment, are DARPA-601, an infra-red sensor experiment, and ECOM-501, an extreme ultra-violet radiation sensing experiment.

Spacecraft and experiment power will be provided by the rotatable, canted solar array seen in figure 8, as well as by an on-board battery system. The battery system will permit experiment operation through all eclipse periods, though not always at maximum power levels. The spacecraft power bus voltage will be 28 ± 6 V DC.

The launch date for the P80-1 spacecraft is tentatively scheduled for the first quarter of 1981 aboard a Space Shuttle flight. The launch will inject the Shuttle orbiter into a circular, 185 km-altitude parking orbit at 55 degrees inclination. The spacecraft will then be ejected from the orbiter cargo bay and injected by two solid rocket motor firings into its final orbit. This orbit will be circular (maximum eccentricity 0.013) at an altitude of 740 ± 93 km and an inclination of $\gtrsim 73$ degrees. When the spacecraft has achieved its operational orbit, the orbit acquisition motors will be jettisoned, the spacecraft will be stabilized in the required 3-axis configuration by hydrazine and cold gas attitude control thrusters, and the power and all other spacecraft systems will be put into a fully operational condition. Experimental operation will commence after a period of spacecraft and experiment outgassing, checkout, and calibration. During the mission, the stabilized spacecraft attitude will be maintained by automatic operation of the momentum wheels and cold gas thrusters of the attitude control system.

Throughout the P80-1 mission, ground tracking and commanding of the spacecraft, as well as receipt of telmetry data from it, will be conducted through the remote tracking network of the Air Force Satellite Control Facility. Data acquisitions from the spacecraft will include both realtime spacecraft and experiment data and a dump of the telemetry data recorded by the spacecraft tape recorders since the previous data dump.

The P80-1 spacecraft will have two nominal attitudes during the mission. The spacecraft coordinate system used to describe these attitudes is indicated in figure 8. In one of them the +X spacecraft axis coincides with the spacecraft orbital velocity vector while the +Z axis coincides with the earth-line vector. In the other the -X spacecraft axis coincides with the velocity vector, with the +Z axis again coincident with the earth-line vector. Approximately every 70 days during the mission a 180 degree yaw maneuver from one attitude to the other will be performed by the spacecraft. These maneuvers will be coordinated with the change in the sun angle relative to the spacecraft orbit plane so as always to maintain a sun-line vector having a +Y axis component (see fig. 8). This spacecraft attitude restriction is imposed by a DARPA-601 requirement that certain sensitive radiator surfaces never view the sun. It also makes it possible for the single, pivoting solar array of the spacecraft, with solar cells on only one side of the array, to operate at efficient sun angles and generate a large fraction of its maximum power output throughout most of the mission.

The P80-1 spacecraft specifications call for the spacecraft to have a reliability factor of $\geq$ 0.90 for nominal on-orbit operation over a one-year period from spacecraft turn-on. However, the spacecraft is designed and will carry sufficient quantitites of all expendables to operate nominally and support experiment operations for a three-year period.

SAMSO-601 Specifications, Configuration, and Mission Operations

The SAMSO-601 flight test experiment comprises the hardware package depicted in figure 9. The diagnostic system components of the package are separately described in reference 5. The two thruster systems in the flight package each consist of a thruster-gimbal-beam shield unit, propellant feed system, PEU, and DCIU. As previously described, these are essentially EMTS-design components with the four modifications discussed. In addition, a few other, minor changes have been made i the EMTS designs to bring them to flight status. All components of the flight package will be qualified.

Figure 8 shows the preferred configuration of the SAMSO-601 flight test components on the P80-1 spacecraft. (Again see ref. 5 for a discussion of the diagnostic device placement and configuration about each thruster.) The conformation shown in figure 8 places one thruster always on the zenith (-Z) surface of the spacecraft and the other on an equatorial surface which is alternately the forward and rear, or ram and wake, surface of the spacecraft relative to its orbital velocity vector. Thruster system performance effects due to the different thruster orientations with respect to the environmental plasma flux stream encountered by the P80-1 spacecraft are possible. Effects large enough to be observed are not expected during single thruster operation, though, because of the low plasma densities (maximum ion or electron density $< 10^6$ cm^{-3}) at the 740 km orbital altitude. However, environmental plasma effects on the thruster performance interactions to be measured with both thrusters operating may be observably dependent on the spacecraft attitude relative to the orbital velocity vector.

The thruster-gimbal units will both be carefully aligned when mounted on the P80-1 spacecraft so that the nominal thrust vector of each passes through the spacecraft center of mass. Operation of either thruster in this orientation will not cause any perturbation in the spacecraft attitude. Also, when thrusting is performed with the thruster on the zenith (-Z) face of the spacecraft (thruster no. 1), it will have no effect on the spacecraft orbital parameters except for a negligible ($<$ 10 m) constant decrement caused in the orbital altitude.

When thrusting is performed with the thruster (thruster no. 2) which is alternately on the front (+X) and back (-X) spacecraft surface, it will lower or raise the spacecraft orbit, respectively, because the thrust vector is directly in line with the orbital velocity vector. For a small orbit change, assuming a circular orbit is maintained with no significant decrease in the spacecraft mass during the thrust period, the magnitude of the altitude change is given by the relationship:

$$\Delta r \approx 2 \frac{F}{m_{sc}} t r_o \sqrt{\frac{r_o}{\mu}} \tag{6}$$

In this equation, Δr is the small change (in m) in the orbit radius r after the thrusting period t (in sec.), r_o is the initial orbit radius (in m), F is the thrust in N, m_{sc} is the spacecraft mass (in kg), and μ is the product of the earth's mass and the universal gravitational constant. The value of μ is 3.986×10^{14} $m^3 sec^{-2}$. The thrust F is positive if the thrust vector and the velocity vector are co-directional, negative if they are anti-directional. Taking $F = 4.98 \times 10^{-3}$ N for the 8-cm thruster, $m_{sc} \approx 1180$ kg for the approximate P80-1 on-orbit mass, and $r_o = 7.118 \times 10^6$ m for the 740 km-altitude orbit, equation (6) yields $\Delta r = 29$ m per hour of thrusting with thruster number 2. When the thruster is oriented on the front face of the spacecraft, this represents a decrease in altitude; with the thruster oriented on the rear face, an increase.

As part of the performance measurement objective of the flight test, a direct thrust measurement may be performed using thruster number 2. Assuming that m_{sc} is known precisely and that each spacecraft altitude determination is accurate to within ± 365 m (1200 ft), equation (6) indicates that a direct determination of F accurate to within 10 percent requires a continuous thrusting time t of 252 hours.

The axes of the two SAMSO-601 thrusters, undeflected by gimballing, are oriented on the P80-1 spacecraft at 90 degrees to each other. (See fig. 8.) As a result, the far-field plasma interaction between the two thrusters, when simultaneously operated, will realistically simulate that between thrusters in the mission model configuration. This will be true in spite of the fact that each of the flight test thrusters will be mounted on the spacecraft so its axis is normal to the spacecraft mounting surface, contrary to the mission model configuration shown in figure 1.

The aximuthal orientation of the beam shield of each SAMSO-601 thruster system on the P80-1 spacecraft is subject to two types of constraints. One type results from the preferred orientations of the diagnostic instrumentation relative to each thruster's beam shield and relative also to the mean sun direction. The sun angle is most important in the case of the solar cell detectors which are included among the diagnostic devices. The other type of constraint arises from the need to prevent thruster efflux interactions with spacecraft systems, particularly the solar array. The beam shield orientation depicted in figure 8 meets the diagnostic instrumentation constraints and is also believed consistent with spacecraft interaction requirements.

Table XII lists the masses, steady-state power requirements, and thermal limits specified for the component units of the SAMSO-601 flight package. The maximum power requirement for the package is estimated to be $\sim$400 W, occurring during non-optimal, simultaneous operation of both thrusters at the nominal beam current condition with full operation of the diagnostic instrumentation. Simultaneous, nominal thrust operation of the two thrusters will not be required for more than six hours out of any ten-hour period. After any such operation the SAMSO-601 power requirements will be held to a substantially reduced level (< 250 W) to allow for recovery of the spacecraft battery system. The minimum temperature limits specified in Table XII for the thruster-gimbal unit and the propellant feed system have been set so as to provide an adequate safety margin above the freezing temperature of mercury, -39° C.

A preliminary mission operations schedule for the SAMSO-601 flight test is presented in Table XIII. This schedule does not take account of any constraints on SAMSO-601 operations imposed by requirements or operations of the other two experiments on the spacecraft. Not considering operation of the SAMSO-601 diagnostic instrumentation, which is intended over the entire duration of the P80-1 mission, the simultaneous operation of even one of the SAMSO-601 thrusters with DARPA-601 is probably precluded due to spacecraft power and energy storage considerations. The simultaneous operation of SAMSO-601 and ECOM-501 is possible and contemplated, however. The mission operations schedule in Table XIII also does not consider operating restrictions which may be imposed due to reduced spacecraft power availability at the beginning and end of each $\sim$ 70 day period just before and after the 180 degree spacecraft yaw maneuvers. Unfavorable sun angles on the spacecraft solar array may result in substantial reductions in the array power generating capability at these times.

The primary flight test objective - spaceflight qualification of the 8-cm thruster system - will be accomplished with thruster number 1 (zenith surface thruster), starting early in the mission. At present, it is intended that this objective be achieved by operating the thruster to meet the mission model requirements implied by the baseline assumptions, as earlier discussed and presented in Table I. That is, 2557 cycles of thruster operation will be performed with thruster number 1. Each cycle will include 2.76 hours of nominal thrusting; hence, a total of 7055 hours of such thrusting will be performed with thruster number 1.

The cooling time allowed between the periods of full thrust operation strongly impacts the time necessary to accomplish this cyclic testing. From the cooldown characteristics observed for 8-cm thrusters and from experience in restarting them from a warmer-than-ambient condition, it is estimated that a two-hour off-period is required to adequately simulate cooling to ambient conditions. (This two-hour period includes 15 minutes for restarting and reacquiring nominal thrust.) For the baseline mission model assumptions, this cool-down and restart time implies a 58 percent maximum duty cycle in the cyclic flight qualification thrusting and a minimum period of 507 days needed during the P80-1 mission to accomplish the flight qualification requirements, as indicated in Table I.

Near the beginning of the SAMSO-601 thruster testing period, thrusters numbers 1 and 2 will be started a few times simultaneously and operated together for several periods of a few hours in duration to demonstrate satisfactory dual thruster operation. This will accomplish the subsidiary objective implied by the flight qualification mission model. During these and subsequent periods of dual thruster operation, all necessary performance data will be obtained to evaluate any performance interaction effects between the two thrusters. All the dual thruster operation required may be accomplished without interfering with the ongoing cyclic operation of thruster number 1 for the flight qualification demonstration.

Throughout the flight test but especially over the course of the cyclic testing of thruster number 1, telemetered performance data on each thruster will be obtained and analyzed to fulfill the performance measurement objective of the flight test. Also, during all appropriate phases of the thruster testing, the effect of the neutralizer switch on thruster performance will be evaluated. At the beginning, middle, and end of the cyclic testing of thruster number 1, rudimentary mapping of the thruster performance will be accomplished by suitable ground-commanded variation of the controlled thruster operating parameters. These performance maps and the normal thruster operating data obtained on thruster number 1 will afford a detailed assessment of any degradation effects on thruster performance and starting which take place as a result of long-term cyclic operation in space.

Beginning at about the 150 day point of the mission operations, thruster number 2 will be used to demonstrate and determine thruster performance in the alternative operating modes described in Table II. The thruster will be operated in each such mode a sufficient period to prove the routine use of the mode in space to be feasible. Repeatable transitions between each mode and nominal full thrust operation or other required operating modes will also be demonstrated. Periodically, thruster number 2 will be briefly operated according to the standard cyclic procedure by which thruster number 1 is operated. Such operation will accomplish the required dual thruster testing and allow direct performance comparisons between the two thrusters, only one of which is subject to the effects of long-term cyclic operation.

During the latter part of the flight qualification testing of thruster number 1, the gimbal of thruster system number 2 will be extensively tested to demonstrate its functional performance and durability. Immediately following completion of the flight qualification testing of thruster number 1, short tests will jointly be conducted on the two thrusters, and with one thruster alone, to evaluate operation in the neutralizer-off alternative operating mode. These tests will assess the effects of neutralizer extinctions during nominal thruster operation. As the final formal test in the planned mission operations of SAMSO-601, continuous thrusting with thruster number 2 will be performed to accomplish a direct thrust measurement and demonstration by the resulting spacecraft orbital change. The details of such a measurement have been quantified previously.

Other useful tests and experiments which may be performed subsequently to the above-described program include: (1) thrust demonstration and measurement by thrusting in a gimballed orientation with either thruster; (2) actual attitude control of the spacecraft by the above means; (3) experiments in which the thruster operation is deliberately varied to investigate the effects thereby produced on the flight test diagnostic measurements; and (4) continued cyclic operation of thruster number 1 by the standard flight qualification procedure to accomplish extended mission model goals.

None of the SAMSO-601 flight operations, tests, and procedures described in this section will be accomplished under real-time ground control. All depend on pre-transmitted, time-tagged execution commands from the ground, with any necessary ground command modifications of the setpoints, reference values, timing constants, and program sequencing in the DCIU. This manner of operating the flight test is consistent with the automatic operation which will be required of the 8-cm thruster system in actual auxiliary propulsion applications. Hence, besides flight qualifying and measuring the performance of the basic thruster system, the flight test will, if successful, flight qualify the concept of autonomous, program-controlled operation of auxiliary propulsion systems for a variety of applications.

CONCLUDING SUMMARY

The principal objective established for the SAMSO-601 flight test is to spaceflight qualify the 8-cm mercury ion thruster system for routine use as a versatile auxiliary propulsion system. The flight qualification objective will be accomplished by simulating the mission requirements of a representative north-south stationkeeping application. The baseline mission model assumed for this application is the stationkeeping for seven years of a 1000 kg geostationary satellite, employing four body-mounted 8-cm thrusters canted 45 degrees to north-south and fired in pairs once a day.

For the flight qualification of the 8-cm thruster system, the adopted mission model implies flight test requirements of 2557 cycles of 2.76 hours nominal thrusting each, for a total of 7055 hours of thrusting. With a two hour cooldown plus start-up period per cycle, a minimum of 507 days are needed in the flight test to accomplish the required mission model demonstration. Because the mission model requires simultaneous, dual thruster operation, the flight test incorporates two thruster systems, oriented at 90 degrees to each other. These will be simultaneously operated to prove compatibility in such operation.

The second objective established for the flight test is to obtain required design data on the 8-cm thruster system performance in space and on the thruster system interfaces with the spacecraft. The critical performance parameters which will be measured in the flight test are thrust, specific impulse, and total system input power. These and the thruster system reliability will be evaluated as a function of operating time and cycles during the flight qualification testing. Any performance effects due to simultaneous dual thruster operation also will be determined. In addition, thruster operation will be demonstrated in a number of alternative operating modes of potential importance to users.

The two 8-cm thruster systems which will be flown in the SAMSO-601 flight test are flight-quality engineering model thruster systems with four significant modifications. (1) An assymmetric, 45 degree beam shield is mounted on the thruster, as a consequence of the mission model specifications of body-mounted stationkeeping thrusters canted at 45 degrees to north-south. This shield serves to protect spacecraft solar arrays or other sensitive surfaces downstream of the thruster grid plane from thruster efflux. (2) A neutralizer switch is incorporated in the thruster grounding circuit to permit operation with the neutralizer common either floating or shorted to spacecraft ground, since users may require either configuration. (3) A commandable latching valve is incorporated in the propellant feedline from the mercury reservoir to provide double containment of the mercury during Space Shuttle launch of the P80-1 spacecraft. Such a valve also removes acceleration constraints otherwise required for the thruster system in order to prevent vaporizer intrusion. And (4), the engineering model system digital interface unit is replaced with a digital controller and interface unit to provide flexible, self-contained, automatic programmed control of all thruster system operating modes and to make possible a simple spacecraft/thruster system interface.

The P80-1 spacecraft, with the SAMSO-601 flight test and two other experiments, will be launched in 1981 aboard the Space Shuttle to a 740 km circular orbit at $\gtrsim$ 73 degrees inclination. The spacecraft will be 3-axis stabilized on-orbit (relative to its earth line and velocity vector). The SAMSO-601 ion thrusters will be located on the zenith surface and on an equatorial surface which is alternately the front and rear spacecraft surface, depending on 180 degrees yaw maneuvers performed approximately every 70 days. Both thrusters will be efficiently utilized during the mission operations to expeditiously accomplish all aspects of the flight test objectives over the three-year lifetime of the spacecraft.

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Table I. North-South Stationkeeping Mission Model Profiles

Assumptions:	(*)							
Spacecraft mass, kg	1000	1000	1000	1000	1000	1000	455	1545
Launch vehicle	Atlas-Centaur; Shuttle	Atlas-Centaur; Shuttle	Atlas-Centaur; Shuttle	Atlas-Centaur; Shuttle	Atlas-Centaur; Shuttle	Atlas-Centaur; Shuttle	Delta 3914	Titan III-C
Spacecraft lifetime, yr	7	7	7	7	5	10	7	7
Total number of 5 mN thrusters equally fired for stationkeeping	4	4	2	4	4	4	2	4
Thruster cant angle to north-south, deg	45	45	45	30	45	45	45	45
Number of days between successive thruster firings	1	2	1	1	1	1	1	1
Mission model implications:								
Duration of each thruster firing+, hr	2.76	6.00	6.00	2.24	2.76	2.76	2.50	4.41
Total mission cycles	2557	1279	2557	2557	1827	3653	2557	2557
Total mission thrusting time per thruster, hr	7055	7669	15332	5717	5041	10079	6395	11281
SAMSO-601 requirements to reproduce mission model:								
Cooling + start-up time per cycle, hr	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00
Cycle duration including start-up and cooling, hr	4.76	8.00	8.00	4.24	4.76	4.76	4.50	6.41
Maximum duty cycle for thrusting, percent	58	75	75	53	58	58	56	69
Minimum total time to accomplish mission model, days	507	426	852	451	362	724	480	683

*Baseline mission model.

+Calculated from (ref. 4): $t(\text{hrs/thrusting}) = 7.64 \text{ arc sin } \left\{4.98\times10^{-6} \frac{m}{nF} \sec\theta\right\}$, in which m = spacecraft mass in kg, n = number of nodal thrustings per day (assumed constant), F = total thrust in N during each thrusting, and θ = cant angle of each thrust vector from north-south (assumed the same for all the thrusters).

Table II. Alternative Operating Modes for SAMSO-601 Demonstration

Mode	Beam on ?	Main discharge on?	Main cathode on?	Neutral-izer on?	Power conserving ?	Propellant conserving ?	Transitions required to	Description and utility
1. Cathode conditioning								Initial on-orbit cathode condition-ing and subsequent reconditioning by application of tip heater power and Hg flow
a. Main cathode	N	N	N	N	N	N	Main cathode on	
b. Neutralizer	N	N	N	N	N	N	Neutralizer on	
c. Both cathodes	N	N	N	N	N	N	Both cathodes on	
2. Cathode maintenance								
a. Main cathode	N	N	Y	N	Y	Y	Full thrust operation	Maintenance of cathode discharges during short off period or during normal off-period when cathodes threaten not to restart
b. Neutralizer	N	N	N	Y	Y	Y	Full thrust operation	
c. Both cathodes	N	N	Y	Y	Y	Y	Full thrust operation	
3. Idle								
a. High	N	Y	Y	Y	N	N	Full thrust operation	For full beam acquisition within 15 sec
b. Low	N	Y	Y	Y	Y	Y	Full thrust operation	For full beam acquisition within 2 min.
4. Throttled thrust	Y	Y	Y	Y	Y	N	Full thrust operation	4.45 mN (1.00 mlb.) thrust generation under limited power availability
5. Neutralizer off	Y	Y	Y	N	N	N	Full off	To test effects of neutralizer extinc-tion; limited time operation only
6. Gimballing	*	*	*	*	N	N	Gimbal centering	Thrust vectoring

*Thruster gimballing normally performed while thruster is off but capability required also during full thrust operation and alternative operating modes 2, 3, and 4.

Table III. 8-Cm Engineering Model Thruster-Nominal Performance

Parameter, Symbol, Units	Value (non-power)	Value (power)
Thrust, corrected,[1] F, mN	4.98	
Specific impulse, corrected,[1] I_{sp}, sec	2600	
Total input power to thruster, P_{ts}, W		119.5
Total efficiency, corrected,[1] η_t, percent	55.3	
Power efficiency, η_e, percent	72.1	
Total propellant utilization, corrected,[1] η_p, percent	76.8	
Discharge propellant utilization, corrected, η_{pD}, percent	82.8	
Total neutral flow, $\dot{m}_t$, mA equivalent	91.4	
Power/thrust, δ, W/mN	24.0	
Beam ion production energy, excluding keeper power, corrected,[1] ---, eV/ion	227	
Beam ion production energy, including keeper power, corrected,[1] W_i, eV/ion	235	
Thrust correction factor for multiply charged ions in beam, β, percent	98.5	
Thrust correction factor for beam divergence, γ, percent	99.3	
Net beam accelerating voltage, V_b, V	1197	
Beam current, J_b, mA	72	
Output beam power, P_b, W		86.2
Neutralizer coupling voltage, V_c, V	-7.9	
Neutralizer coupling power, P_c, W		0.6
Accelerator voltage, V_{Accel}, V	-300	
Accelerator drain current, J_{Accel}, µA	170	
Accelerator drain power, P_{Accel}, W		0.3
Discharge voltage, V_D, V	35	
Discharge emission current, J_E, mA	455	
Discharge power, P_D, W		15.9
Main cathode:		
Keeper voltage, V_{MK}, V	6	
Keeper current, J_{MK}, mA	100	
Keeper power, P_{MK}, W		0.6
Heater power, P_{MCH}, W		0
Vaporizer voltage, V_{MVH}, V	4.9	
Vaporizer current, J_{MVH}, A	1.68	
Vaporizer power, P_{MVH}, W		8.2
Flowrate, $\dot{m}_D$, mA equivalent	84.8	
Neutralizer:		
Keeper voltage, V_{NK}, V	13	
Keeper current, J_{NK}, mA	500	
Keeper power, P_{NK}, W		6.5
Heater power, P_{NCH}, W		0
Vaporizer voltage, V_{NVH}, V	2.0	
Vaporizer current, J_{NVH}, A	0.6	
Vaporizer power, P_{NVH}, W		1.2
Flowrate, $\dot{m}_N$, mA equivalent	6.6	

Note: 1. Corrected for beam divergence and presence of multiply charged ions in beam.

Table IV. Engineering Model Thruster System Propellant Reservoir-Feed System and Gimbal Specifications

Propellant Reservoir-Feed System	
Mass when filled with Hg propellant, kg	9.91
Hg propellant capacity, kg	8.75
Dry mass, kg	1.16
Method of propellant expulsion	Pressurized gas
Pressurized gas composition, mol percent	N_2:80, Kr:20
Gas pressure at 25° C with reservoir filled, Pa	2.41×10^5
Gas pressure at 25° C with reservoir filled, psia	35
Gas pressure at 25° C with reservoir empty, Pa	1.03×10^5
Gas pressure at 25° C with reservoir empty, psia	15
Operating temperature range, °C	-35 to +100
Pressure monitor	Pressure transducer
Temperature monitor	Pt resistance thermometer
Gimbal	
Mass (including feedline and feedline manifold), kg	1.50
Deflection capability, each of 2 orthogonal axes, deg	± 10
Deflection reproducibility, any direction, deg	± 0.25
Rezeroing deflection reproducibility from any direction, deg	± 0.25
Method of limiting gimbal drive travel	Microswitches
Maximum motor drive power, 2 gimbal motors operating, W	12
Power required to maintain vectored position, W	0
Motor steps per degree deflection[1]	2421
Total motor steps, +10 to -10 degree deflection[1]	48420
Motor steps/sec	400
Time for 1 degree deflection,[1] sec	6.1
Time for deflection[1] from +10 to -10 degree, sec	121
Operating temperature range, °C	-40 to +175

Note: 1. For rotation about gimbal hinge axis. For rotation about hinge pivot axis, divide figures given by 2.

Table V. Engineering Model Thruster System Power Electronics Unit - Power Supply Specifications and Control

PEU Power Supply	Maximum power conditions		Nominal power conditions		Power supply regulation		Control inputs	
	V	A	V	A	of[1]	±%	No. fixed setpoints	D/A reference level control and range
1. Screen	1200	0.090	1180	0.072	V	1	1	None
2. Accelerator	-300	0.008	-300	0.0002	V	1	1	None
3. Discharge	50	1.0	37	0.50	J	2	0	7-Bit word:0.20 to 1.0 A
4. Main keeper	25	0.50	8	0.06	J	3	4	None
5. Neutralizer keeper	25	0.60	15	0.50	J	3	4	None
6. Main cathode heater[2]	10.0	4.0	7.7[3]	3.1[3]	J	5	8	None
7. Neutralizer cathode heater[2]	10.0	4.0	7.9[3]	3.2[3]	J	5	8	None
8. Main vaporizer heater[2]	6.0	3.0	4.9	1.7	J	5	7	Auto-control on 7-bit word for (V_D - V_{MK}): 16 to 35 V
9. Neutralizer vaporizer heater[2]	4.0	2.0	2.0	0.6	J	5	4	Auto-control on 7-bit word for V_{NK}: 10 to 30 V
10. Main cathode igniter	5000	2.5[4]	5000	2.5[4]	V	-	1	None
11. Neutralizer cathode igniter	5000	2.5[4]	5000	2.5[4]	V	-	1	None

Notes:
1. V = voltage, J = current.
2. Power supplied is AC; values are root mean square.
3. Nominal power conditions are for nominal cathode starting. During nominal thruster operation, V, A, and power are 0.
4. Igniter circuits deliver pulses of 50 mJ energy and 5 kV voltage to keepers during cathode starting, at intervals of ~5 second. Current value is average over typical 4μ sec igniter pulse duration.

Table VI. **Engineering Model Thruster System Operating Parameter Telemetry**

Parameter Measured	Units	Range
1. Screen voltage, V_S	V	0 to 1200
2. Beam current, J_B	mA	0 to 100
3. Accelerator voltage, V_{Accel}	V	0 to -500
4. Accelerator drain current, J_{Accel}	mA	0 to 5
5. Discharge voltage, V_D	V	0 to 50
6. Discharge emission current, J_E	mA	0 to 1000
7. Main keeper voltage, V_{MK}	V	0 to 25
8. Main keeper current, J_{MK}	mA	0 to 500
9. Neutralizer keeper voltage, V_{NK}	V	0 to 32
10. Neutralizer keeper current, J_{NK}	mA	0 to 600
11. Neutralizer common potential (from spacecraft ground), V_G	V	0 to -100
12. Main vaporizer temperature (sensor resistance), T_{MV}	Ω	0 to 500
13. Neutralizer vaporizer temperature (sensor resistance), T_{NV}	Ω	0 to 500
14. Propellant pressure, p_{Hg}	psia	0 to 50
15. Propellant reservoir temperature (sensor resistance), T_{Hg}	Ω	0 to 500
16. Total thruster system 70 V input current, J_{ps}	A	0 to 5
17. Main cathode heater current, J_{MCH}	A	0 to 4
18. Neutralizer cathode heater current, J_{NCH}	A	0 to 4

Table VII. Engineering Model Thruster System Power Electronics and Digital Interface Unit Specifications

Power Electronics Unit	
Mass,[1] kg	6.97
Input power bus voltage, V	70 ± 20, DC
Input power required during nominal thrusting, W	160.3
Output power to thruster during nominal thrusting, W	119.5
Efficiency of power conversion to thruster input power during nominal thrusting, percent	74.5
Operating temperature range, ^{o}C	-30 to +60
Digital Interface Unit	
Mass,[1] kg	3.15
Input power bus voltage, V	28 ± 1, DC
Input power required,[2] W	4.7
Operating temperature range, ^{o}C	-30 to +60

Notes: 1. Not including any connecting cable harnesses.

2. Essentially constant and independent of thruster operating condition.

Table VIII. 8-Cm Thruster Nominal Operating Point

Operating Parameter	Value
Beam current, J_b, mA	72 ± 1
Screen voltage, V_S, V	1180 ± 10
Accelerator drain current, J_{Accel}, μA	100 to 300
Accelerator voltage, V_{Accel}, V	-300 ± 10
Discharge emission current, J_E, mA	400 to 600
Main keeper to anode voltage ($V_D - V_{MK}$), V	29 ± 1
Main keeper current, J_{MK}, mA	60 +20, -10
Main keeper voltage, V_{MK}, V	5 to 10
Main cathode heater power, P_{MCH}, W	0
Main vaporizer heater power, P_{MVH}, W	5 to 10
Discharge flowrate, $\dot{m}_D$, mA (equivalent)	80 to 88
Neutralizer keeper current, J_{NK}, mA	500 ± 25
Neutralizer keeper voltage, V_{NK}, V	8 to 16
Neutralizer cathode heater power, P_{NCH}, W	0
Neutralizer vaporizer heater power, P_{NVH}, W	1 to 3
Neutralizer flowrate, $\dot{m}_N$, mA (equivalent)	6 ± 1
Neutralizer coupling voltage, V_c, V	-5 to -20

Table IX. Properties of Graphite Fiber-Polyimide Resin Composite Material

Density, $g.cm.^{-3}$	1.6
Graphite fiber content, weight %	63
volume %	57
Resin content, weight %	37
volume %	43
Formulated resin molecular weight, AMU	1500
Overall composition:	
C, weight %	>85
O, N, and H, total weight %	<15
Void volume, volume %	<3
Tensile strength at 25°C, Pa	5.9×10^8
psi	85,000
Flexural strength at 25° C, Pa	6.9×10^8
psi	100,000
Interlaminar shear strength at 25° C, Pa	5.9×10^7
psi	8500
Continuous service temperature limits in vacuum:	
High temperature limit, °C	340
Low temperature limit, °C	<-100
Outgassing in vacuum at 300° C, weight % loss in 1000 hrs	<1
Bulk resistivity at 25° C, ohm cm	<0.018

Table X. Telemetry Indications of Potential Hazards to Thruster System or Spacecraft.

Thruster telemetry condition	Condition indicated	Hazard
Conditions affecting spacecraft:		
1. High T_{MV}, without main cathode lit	Excessive main vaporizer neutral Hg flow	Hg condensation on cold surfaces
2. High T_{NV}, without neutralizer lit	Excessive neutralizer vaporizer neutral Hg flow	Hg condensation on cold surfaces
3. Abnormally rapid decrease in propellant reservoir pressure, p_{Hg}	Abnormally high Hg flowrate due to feedline or vaporizer failure	Uncontrolled Hg loss
Conditions affecting thruster:		
1. Overlimit T_{MV} or T_{NV}	Vaporizer heater setpoint failure or loss of regulation; vaporizer control loop failure	Cathode or discharge chamber flooding; grid arcing; vaporizer heater burnout
2. Underlimit or decreasing T_{MV} or T_{NV} in normal thrusting	Vaporizer intrusion; vaporizer control loop failure	Loss of vaporizer flow control
3. Unplanned or excessive J_{MCH} or J_{NCH} while cathode is lit	Tip heater setpoint failure or loss of regulation. DCIU failure to sense cathode is lit	Overheating and degradation of cathode; tip heater burnout; insulator coating and breakdown.
4. Overlimit J_B, J_E, or J_{CK} during normal thrusting	J_E or J_{CK} setpoint failure or loss of regulation; J_B control loop failure	Overheating of discharge chamber and/or main cathode; insulator coating and breakdown
5. Overlimit J_A	High beam or charge exchange ion impingement of accel. grid; grid arcing	Sputter erosion of accel. grid; excessive discharge chamber deposition; insulator coating and breakdown
6. Underlimit V_S or $\|V_A\|$ or excessive high voltage recycling	Grid arcing or partial shorting; loss of V_S or V_A regulation	Grid-short welding; grid insulator coating and breakdown; failure to generate nominal thrust
7. Overlimit V_D	Main vaporizer heater or control loop failure; excessive J_E	Excessive discharge chamber sputter erosion, deposition, and heating; insulator coating and breakdown
8. Overlimit $\|V_G\|$ during normal thrusting	Neutralizer extinction or degraded performance; insufficiently high J_{NK}	Loss of nominal thrust; excessive accel. grid ion impingement and neutralizer erosion

Table XI. Thruster System Anomalies Considered for DCIU Accommodation

1. Neutralizer extinction.
2. Main cathode extinction.
3. Excessive grid arcing.
4. Grid shorting.
5. Overlimit $|V_G|$.
6. PEU telemetry failure in J_B, J_E, V_D, V_{MK}, or V_{NK}, affecting operation of feedback control loop for J_B or either vaporizer heater.
7. Failure of platinum resistance thermometer element giving T_{MV} or T_{NV}, or DCIU telemetry failure of either, disabling DCIU thruster control algorithms based on temperature measurement.
8. PEU or DCIU telemetry failure for any thruster parameter for which a shutdown limit is set in DCIU.
9. Failure of feedback control loop for J_B or either vaporizer heater.
10. Failure or shift of any fixed setpoint or reference level provided by DCIU to PEU.
11. Gimbal motor drive stall in one direction.
12. Loss of supply regulation for either cathode tip heater or either cathode keeper discharge.
13. Partial short in either cathode tip heater.
14. Significant current leakage across either cathode keeper insulator.
15. Significant current leakage across main isolator with high voltage applied.

Table XII. Masses, Power Requirements, and Temperature Limits of

SAMSO-601 Components

Masses for each thruster system, in kg:

Thruster	1.97
Beam shield	0.11
Gimbal (with neutralizer switch)	1.59
Propellant reservoir and feed system, filled (with valve)	11.11
Power electronics unit	7.03
Digital controller and interface unit	3.22
Interconnecting harness	1.59
Diagnostic system, including sensors, mounting booms and extensions, electronics, and interconnecting harness; total for flight test	27.30

Steady state power requirements for nominal operation of each thruster system, in W:

From 70±20 VDC power bus:	
PEU dissipation	40
Thruster dissipation + beam power	120
Gimbal (both drive motors operating)[1]	12
From 28±6 VDC power bus for DCIU	13
From 28±6 VDC power bus for total diagnostic system	25

Temperature limits for thruster system components, in ^{o}C:

Component	Limiting operating range	Non-operating survival range
Thruster-gimbal assembly[2]	-30 to +60	-30 to +85
Propellant reservoir and feed system[2]	-30 to +60	-30 to +70
Power electronics unit	-20 to +50	-40 to +70
Digital controller and interface unit	-20 to +50	-40 to +70

Notes: 1. Maximum continuous duration of gimbal operation is 2 minutes. Gimbal is not normally operated during nominal thruster operation.

2. Temperatures listed are baseplate mounting surface temperatures.

Table XIII. Preliminary Mission Operations Schedule for SAMSO-601.[1]

Time Period[2] (days)	Operation
1-1096	Full operation of diagnostics. Both thruster system DCIU's on.
8-9	Condition thruster No. 1 for start-up.
9-15	Operate thruster No. 1 steady-state at nominal operating point and do performance map. Shut down thruster.
17-18	Condition thruster No. 2 for start-up.
18-20	Operate thruster No. 2 steady-state at nominal operating point and do performance map. Shut down thruster.
22-275	Start-up thruster No. 1 and operate according to baseline cyclic schedule (2.76 hrs nominal thrust, 2 hrs cooldown + start-up) for 1276 cycles. Then shut down thruster.
29-42	Start-up thruster No. 2 and operate steady-state and cyclically (with appropriate off-periods) to demonstrate dual thruster operation and start-up. Shut down thruster.
150-155	Operate thruster No. 2 in short tests to demonstrate all cathode maintenance, idle, and throttled thrust alternative operating modes.
155-210	Operate thruster No. 2 steady-state and cyclically to evaluate all cathode maintenance alternative operating modes and required transitions. Shut-down thruster.
215-245	Operate thruster No. 2 steady-state and cyclically to evaluate both of idle alternative operating modes and required transitions. Shut-down thruster.
250-262	Operate thruster No. 2 steady-state and cyclically to evaluate throttled thrust alternative operating mode and required transitions.
262-275	Operate thruster No. 2 steady-state and cyclically (with appropriate off-periods) to evaluate dual thruster operation and start-up. Shut-down thruster.
280-282	Start-up thruster No. 2 and operate according to baseline cyclic schedule for 10 cycles, then do performance map and shut-down thruster.
282-287	Start-up thruster No. 1 and cyclically operate for 5 long cool-down cycles (2.76 hrs nominal thrust, 21.24 hrs cool-down + start-up), then do performance map.
287-541	Operate thruster No. 1 according to baseline cyclic schedule (2.76 hrs nominal thrust, 2 hrs cool-down + start-up) for 1276 cycles. Then do performance map and shut-down thruster.
292-340	Operate thruster No. 2 as required to complete evaluation of cathode maintenance, idle, and throttled thrust alternative operating modes and required transitions. Shut-down thruster.
340-500	Operate thruster No. 2 gimbal as required to demonstrate operation and durability.
500-540	Start-up thruster No. 2 and operate steady-state and cyclically (with appropriate off-periods) to complete evaluation of dual thruster operation and start-up. Shut-down thruster.
546-553	Perform required operation of both thrusters, jointly and individually, to evaluate neutralizer-off alternative operating mode.
553-580	Start-up thruster No. 2 and operate steady-state at nominal operating point for direct thrust demonstration by effect on orbital altitude. Then do performance map and shut-down thruster.
580-592	Recondition thruster No. 1 (by cathode conditioning alternative operating mode) and cyclically operate it for 5 long cool-down cycles (2.76 hrs nominal thrust, 21.24 hrs cool-down + start-up) and 5 long operation, long cool-down cycles (6.00 hrs nominal thrust, 18.00 hrs cool-down + start-up.
592-1096	Perform such other operation of and tests with both thrusters as advisable, based on previous results.

Notes: 1. Mission operations schedule does not take account of any constraints arising from spacecraft power limitations or operation of other P80-1 experiments.

2. Days from start of period available for experiment operation, following stabilization, outgassing, and checkout of spacecraft.

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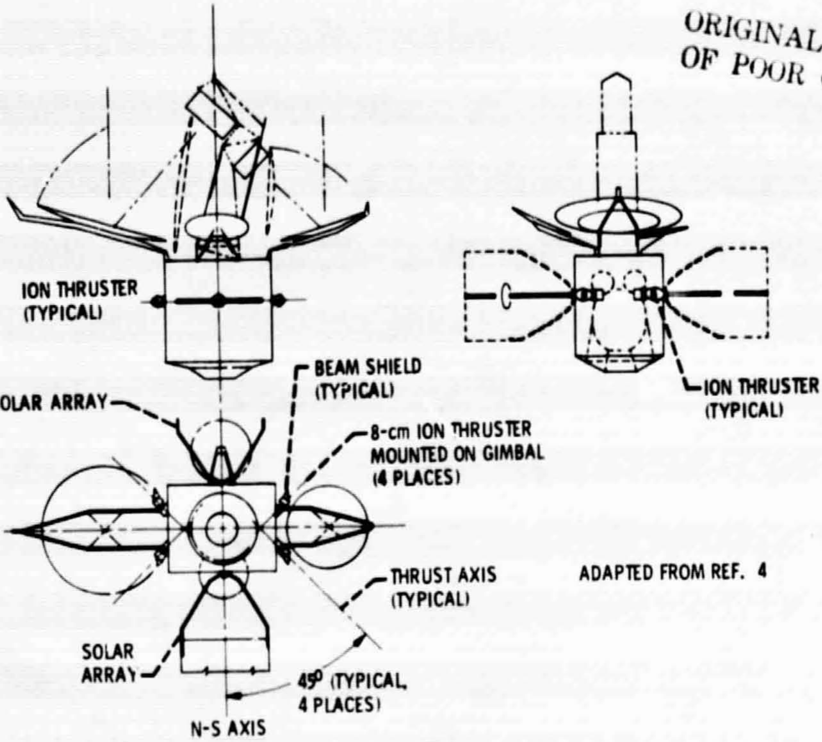


Figure 1. - Four-thruster, body-mounted configuration for auxiliary propulsion applications on 3-axis stabilized geostationary spacecraft.

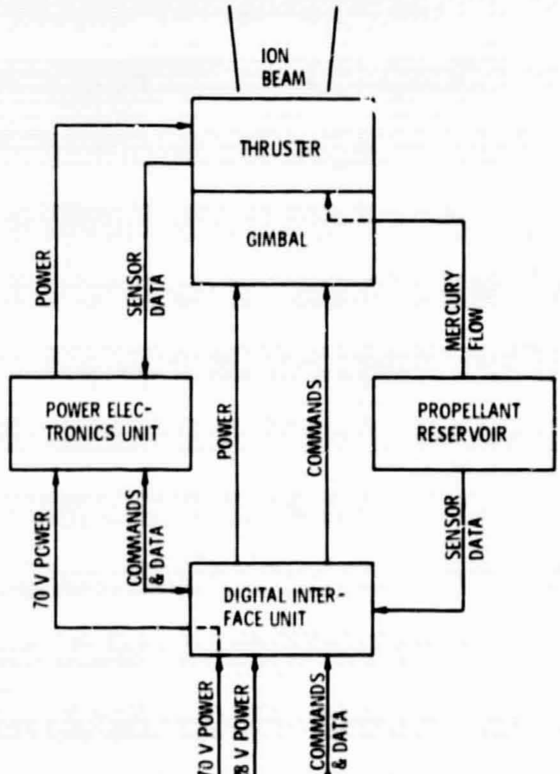
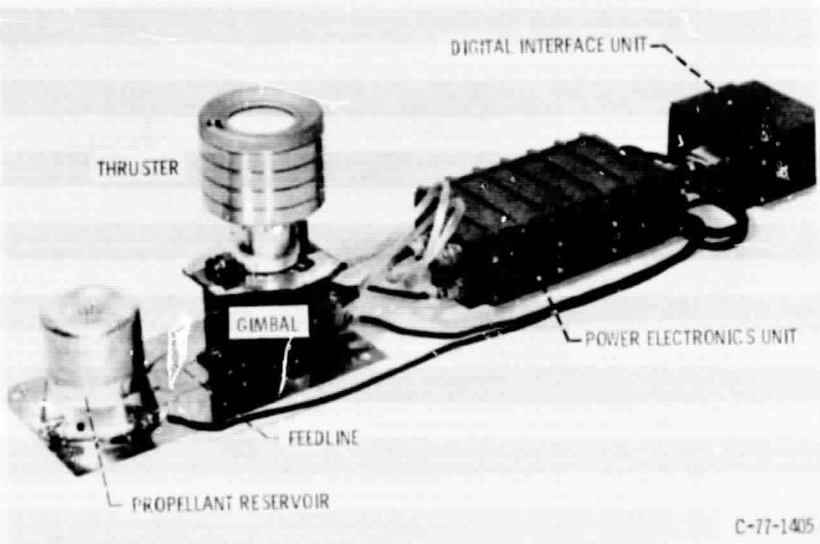


Figure 2. - Engineering model thruster system: functional block diagram.



Flnure 3. - 8-Cm engineering model thruster system.

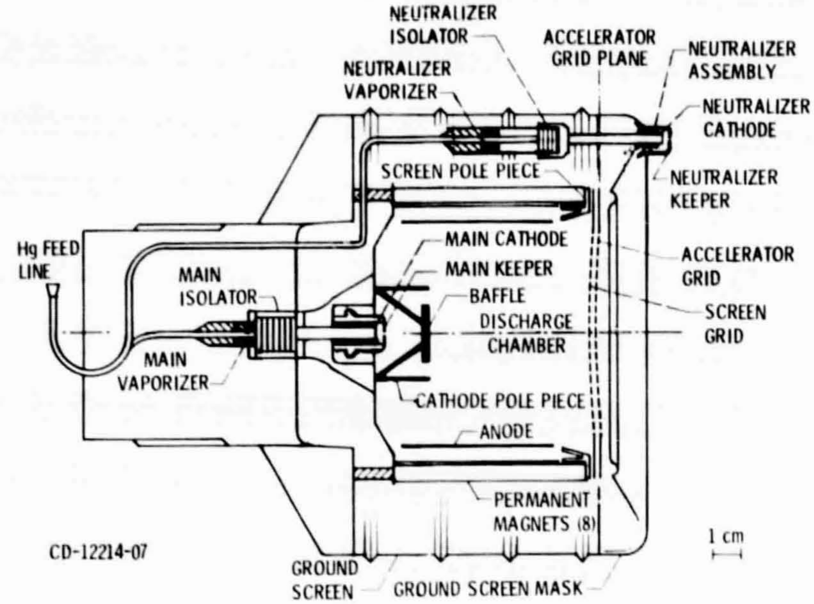


Figure 4. - 8-Centimeter engineering model thruster-cutaway view.

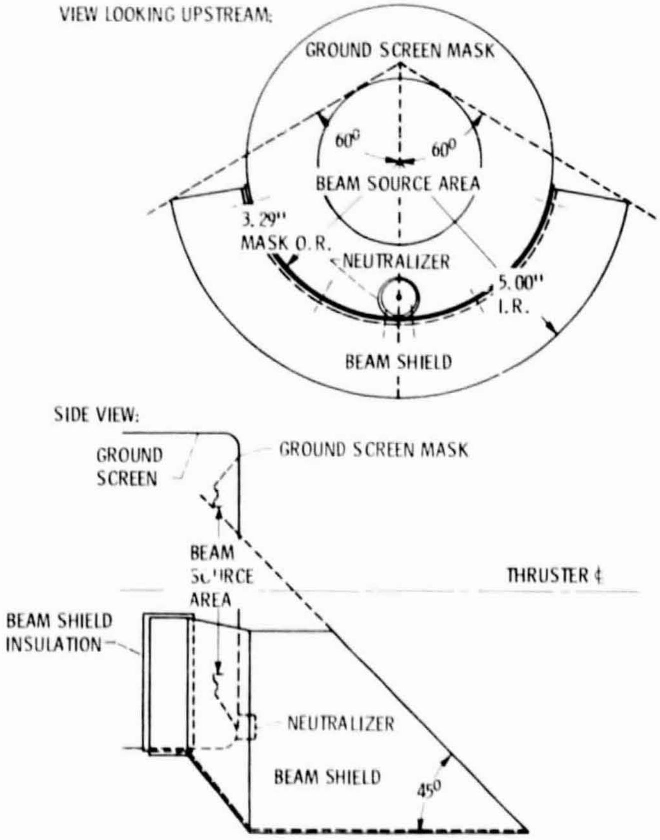


Figure 5. - 8-Centimeter EM thruster beam shield-flight test design.

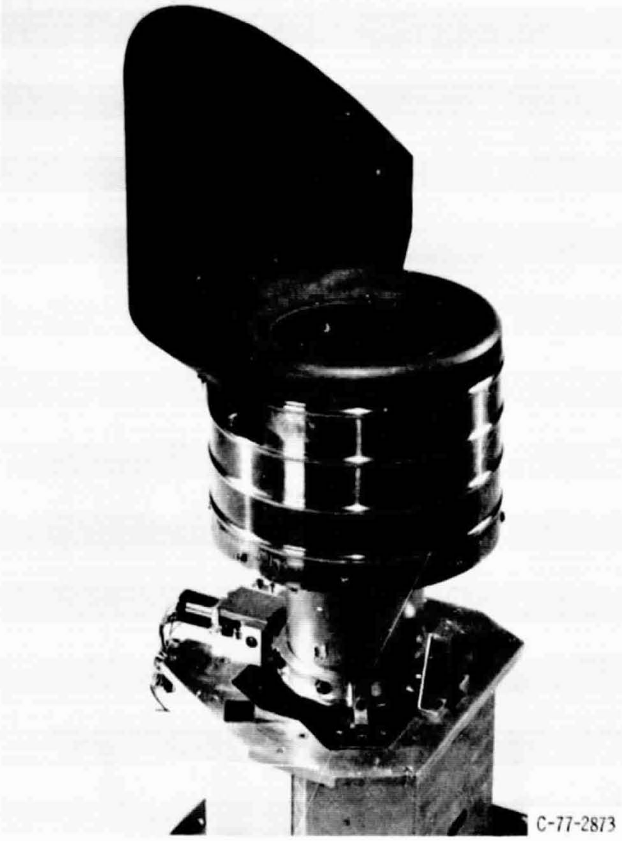


Figure 6. - Flight design beam shield, mounted on EM thruster.

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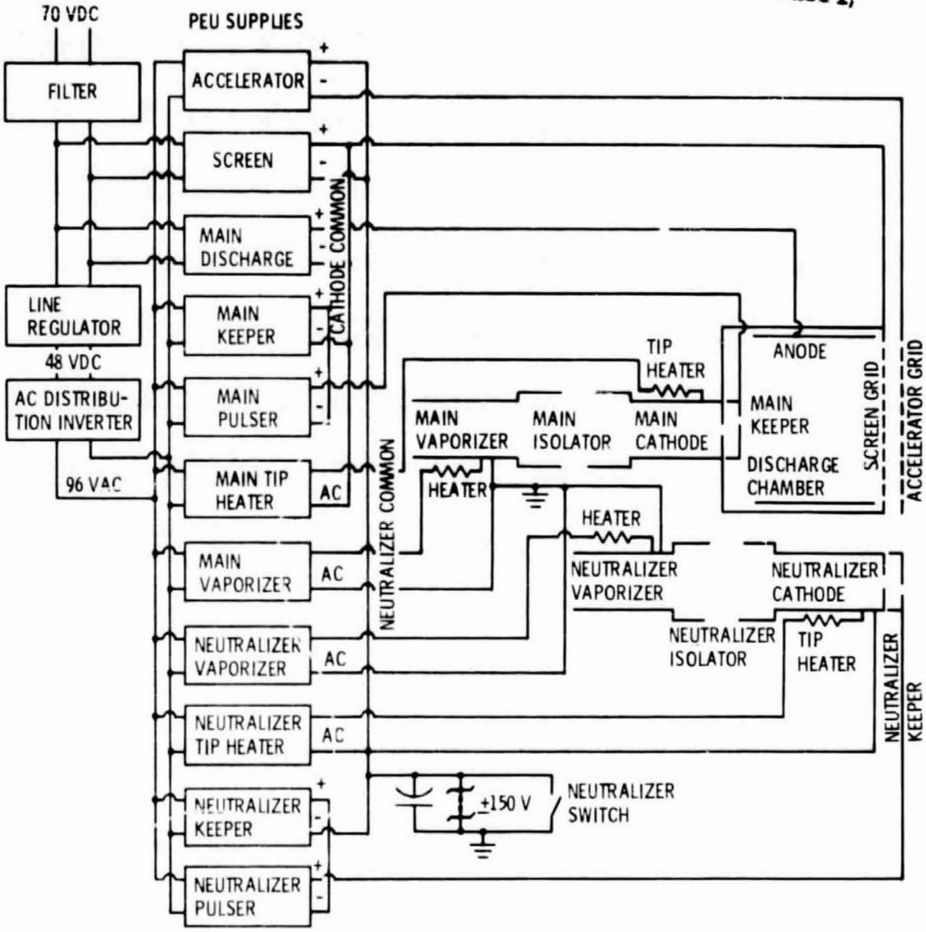

Figure 7. - Flight thruster-PEU power and grounding diagram.

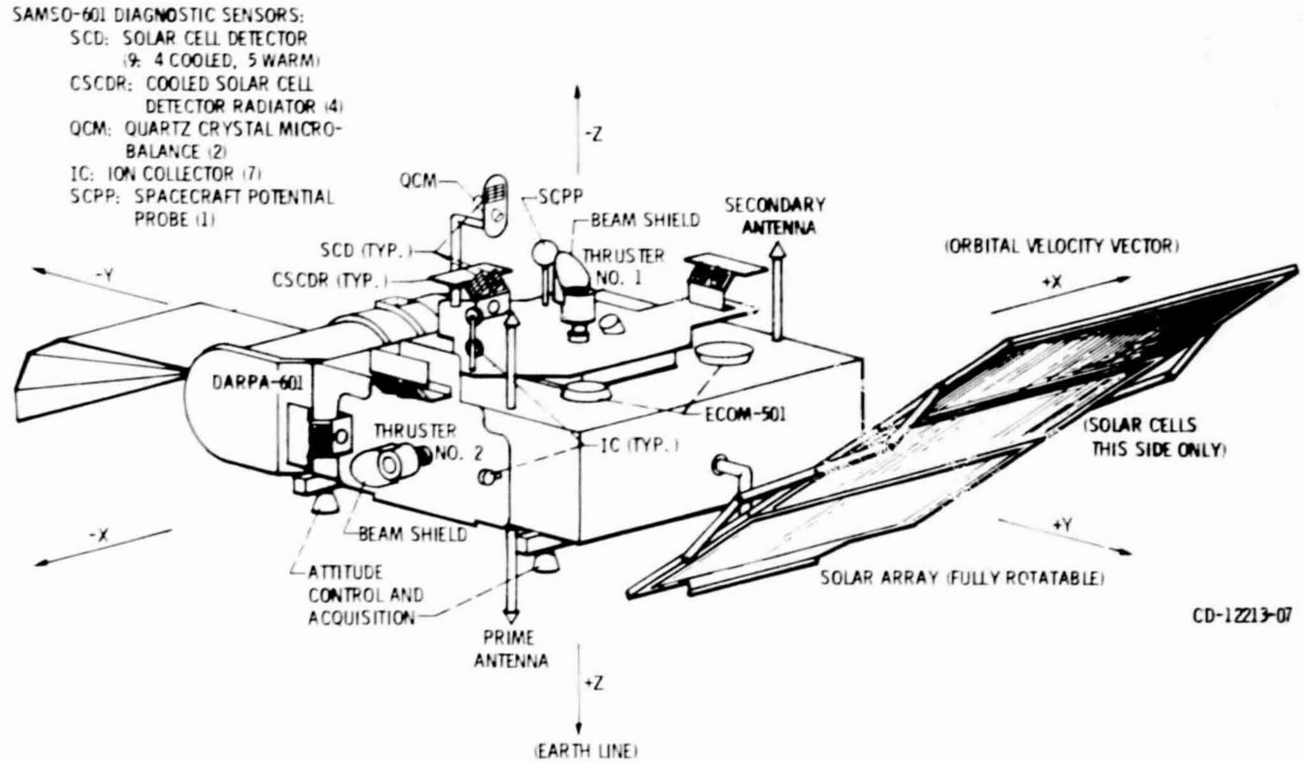

Figure 8. - Preliminary P80-1 spacecraft configuration orbit.

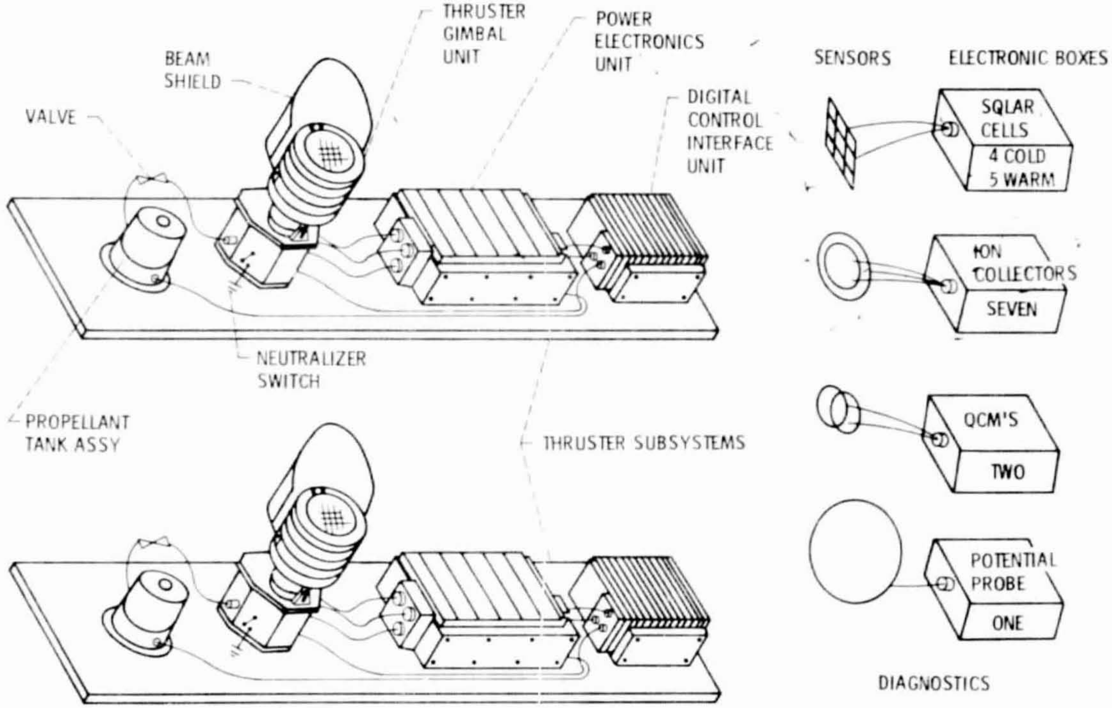

Figure 9. - SAMSO-601 flight test package.