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ION BEAM PLUME AND EFFLUX CHARACTERIZATION FLIGHT EXPERIMENT STUDY

FINAL REPORT

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
Cleveland, Ohio

CONTRACT NAS 3-20387

DECEMBER 1, 1977

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DEFENSE AND SPACE SYSTEMS GROUP
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ION BEAM PLUME AND EFFLUX CHARACTERIZATION

FLIGHT EXPERIMENT STUDY

FINAL REPORT

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16. Abstract

The Ion Beam Plume and Efflux Characterization Flight Experiment Study has examined the definition and configuration of a flight experiment and flight experiment package for a Shuttle-borne flight test of an 8-cm mercury ion thruster. The principal emphasis in the flight experiment is to obtain charged particle and neutral particle material transport data that cannot be obtained in conventional ground based laboratory testing facilities. The principal features of the space environment to be utilized here are the absence of material boundaries and the presence of the ambient space plasma. A second objective of the Shuttle thruster flight test is the Shuttle flight test verification concept through which, by the use of both ground and space testing of ion thrusters, the flight worthiness of these ion thrusters, for other spacecraft applications, may be demonstrated.

A principal advantage of a Shuttle flight test is the recoverability of the payload. This recoverability has important implications in terms of the use of the payload hardware for serial flight testing and in terms of reduced per flight testing costs. A series of growth mode flight experiments for the thruster flight tests has been described, including the modular build-up of multi-thruster tests to examine "cluster" effects in the combined plumes and including the substitution of other thrusters in the flight test package.

A principal limitation in the Shuttle flight test of an ion thruster is in the test duration. The range of available test time lies between 10^2 and 10^3 hours with the latter figures as a possibility only after the development of the prolonged mission (40 day) Shuttle Orbiter capability. Because of these test time limitations, endurance testing of ion thrusters will continue to be a ground based laboratory test.

The flight experiment definition for the ion thruster has initially defined a broadly ranging series of flight experiments and flight test sensors. From this larger test series and sensor list, an initial flight test configuration has been selected with measurements in charged particle material transport, condensible neutral material transport, thruster internal erosion, ion beam neutralization, and ion thrust beam/space plasma electrical equilibration. These measurement areas may all be examined for a seven day Shuttle sortie mission and for available test time in the 50 - 100 hour period.

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1. INTRODUCTION

This report will describe a systems study for a flight test of an 8 centimeter electron bombardment mercury ion thruster on the Shuttle Orbiter of the Space Transportation System. The systems study will first examine the factors which guide and influence the experiment design for an ion thruster flight test in the specific context of a Shuttle borne experiment. These flight experiment planning factors are discussed in Section 2. The planning factors and previously obtained test results from ion thruster laboratory experiments lead to the flight experiment definition. Section 3 describes this flight experiment definition which includes a broadly ranging series of flight experiments. From this broadly ranging series of experiments, a sub-group has been selected as an initial flight experiment for the Shuttle Orbiter. The rationale for this sub-group selection is discussed and two configurations of the flight test equipment in the Orbiter are described. Section 4 contains these flight experiment configurations. A following section of this report, Section 5, examines the overall program plan for the flight experiment, discusses the experiment support requirements, and provides an estimate of the program costs. A summary of this report is given in Section 6.

2. FLIGHT EXPERIMENT PLANNING FACTORS

2.1 OVERALL FLIGHT TEST RATIONALE AND GOALS

Ion thruster testing in space involves program costs which must be justified by the specific advantages and circumstances of flight experimentation as compared to ground based laboratory experimentation. The program costs for flight testing involve not only hardware acquisition costs but also the time expenditure costs in adapting a thruster development program to the periods required to initiate, or to reiterate, flight experiments. This flight test planning study will develop a series of flight experiments which will utilize previously obtained ground based experimental measurements, which will continue these measurements under conditions which cannot be effectively duplicated in laboratory facilities, and which will justify the necessary program expenditures.

The goals for the ion thruster flight experiment will divide into two groups. The first group, TECHNOLOGY GOALS, will be the acquisition of material transport data (for both charged and neutral particles) and spacecraft electrical equilibration data which cannot be obtained in the presence of the material boundaries of conventional (ground based) testing facilities. The second group of goals comprise the SHUTTLE FLIGHT TEST VERIFICATION CONCEPT, which, utilizing both laboratory and flight experiment data, provides a verification of flight worthiness for ion thrusters for other spacecraft applications. These flight test goals will be more fully developed and discussed in the sections which follow.

2.2 OPPORTUNITIES AND CONSTRAINTS IN THE USE OF THE SHUTTLE ORBITER FOR AN ION THRUSTER FLIGHT TEST

Before examining specific opportunities and constraints for an ion thruster flight test in the Shuttle Orbiter, the specific advantages of space testing, irrespective of the host vehicle, should be described. There are, basically, three properties in the space test configuration that the thruster tests will utilize. These are:

1) the zero gravity condition,

2) the absence of material boundaries, and,

3) the presence of the ambient space plasma.

The zero gravity condition in (1) above assures that the thruster feed systems are operating under those conditions required for space flight utilization of ion thrusters. The absence of material boundaries in the space condition is of significant value for two reasons. First, because of the "high pumping speed" of space, the neutral mercury atoms from the thruster have a single "outward bound" traversal of the space and are not returned to the thruster vicinity (as they are in the laboratory) to participate in "facility generated" Group IV ion reactions. Second, the absence of material boundaries results in a single "outward bound" traversal of the space by the thrust ions and thus removes the facility generated sputtered metal atoms which result in the laboratory as a result of thrust ion impact on collectors. The remaining property of space (item 3 above) is the presence of the ambient space plasma which permits the thruster flight experiment to examine the thrust beam plasma-to-space plasma electrical equilibration reactions which will be present for space operated ion thrusters.

In addition to the space properties described above, a Shuttle Orbiter flight test will possess several, Orbiter specific, opportunities. These are:

1) payload recoverability,

2) payload power, weight, and volume capabilities,

3) manned participation, and

4) Orbiter "facilities" utilization.

Payload recoverability is an important feature of the Orbiter flight test. Recoverability permits the detailed post-flight examination of the ion thruster and the diagnostic payload for internal erosion and material transport experiments. Recoverability also permits the amortization of the ion thruster costs and diagnostic payload costs over a more extended flight series, where it has been assumed here that the serial mission capability of the Orbiter can be utilized for an iterated series of flight experiments. The possibility of a series of flight tests also influences the experiment design because it permits initial (and, perhaps, simplified) flight experiments at lower costs with potential add-on capabilities in the growth modes of the experiment. The payload, power, weight, and volume capabilities of the Orbiter (Item 2 above) are listed as an "opportunity" for flight test design in that the design can proceed with generally relaxed requirements in these areas when compared to typical conditions of automated spacecraft. For these automated flights, the significant units in payload design may be in pounds of experiment weight, requiring watts for operation, and occupying liters of spacecraft volume. The Shuttle Orbiter can employ a totally different level of specification in these parameters and the experiment design to be developed in succeeding sections will utilize this expanded capability. The use of power, weight, and volume in the experiment design will not be gratuitous, however, in view of a significant number of anticipated Shuttle Orbiter users which can result in reinstated premiums on these payload parameters. The manned participation (item 3, above) is considered an opportunity for the flight experiment design in view of constraints in Shuttle Orbiter operational time and in orientation (to be discussed below). Manned participation also permits direct viewing of the payload in the event of either thruster or diagnostic array malfunction and thus provides an additional level of payload observation. Finally, the Orbiter "facilities" utilization (item 4, above) is listed as an experiment design opportunity. The facilities considered here are (in addition to power) the thermal control capability (fluid cooling loops) and experiment command and data management (using the on-board systems in the Orbiter avionics). As with Orbiter payload power, weight, and volume, an expanded list of payload users can transform the opportunity of the Orbiter facilities utilization into a constraint.

There are two principal constraints in the use of the Shuttle Orbiter for an ion thruster flight test. These constraints are:

1) total operational time, and

2) Orbiter orientation.

Operational time for presently configured Orbiters clearly limits the period of thruster operation to, at most, seven days. This period of time is sufficient to evaluate thruster internal erosion and external material transport. Long term testing of the ion thruster (10^3 hours, for example) cannot be considered in the context of the present Orbiter. Alteration of the Orbiter to a 30-day mission capability (a proposed uprating) would make the "thousand hour" ion thruster flight test possible, in principle, but would also raise problems in total energy consumption ($\sim$ 200 kilowatt hours for the thousand hour 8-cm thruster test). The remaining Orbiter constraint (item 2, above) is in Orbiter orientation. Several of the charged particle measurements in the thruster test may require specific Orbiter orientations to create a "plasma wake" in the space plasma. Although such Orbiter orientations can be carried out, the constraints on total mission duration and in total mission reorientation indicate that experiment requirements on Orbiter orientation should be carefully considered in advance and must also be compatible with the demands of the many other payload elements on the flight.

The opportunities and constraints discussed as generally applicable mission planning factors in the preceding paragraphs will be utilized further in the development and definition of the flight experiment. The use of those factors must remain, necessarily, somewhat qualitative in view of both an implicit multi-mission flight test concept (involving future ion thruster flight tests on Orbiters with unknown loading factors in the remaining payload) and an initial flight which also exists as a portion of an otherwise unspecified total payload.

2.3 REQUIRED FLEXIBILITY IN FLIGHT EXPERIMENT PLANNING

Section 2.2 has noted that the initial (and, perhaps, future) ion thruster flight experiment is an element of a presently undetermined payload. The use of the Orbiter, with its serial experiment planning capabilities, demands that the flight experiment possess a high level of flexibility in its integration into the total payload. Two forms of this flexibility will be discussed here. These are:

1) experiment mounting flexibility, and,

2) experiment operational period flexibility.

Experiment mounting flexibility must be present so that the experiment can accommodate easily to the available payload mounting locations in the Orbiter bay and in conjunction with other, still to be determined, companion payloads. A failure to develop such integration flexibility can seriously erode the number of available Orbiter flight opportunities. For this reason, the flight experiment package has been designed for convenient mounting in a series of mounting modes. Section 4 will present two of these payload mounting modes and will discuss the benefits and potential problem areas in each mode. In addition to experiment mounting flexibility, there is an experiment operational period flexibility (item 2, above). This operational period flexibility is required because of presently unknown demands on the total available Orbiter flight period by other payload elements and because of variations in available operational time from one Orbiter flight to another. To approach this operational period flexibility, the planned experiments will be designated in one or the other of three levels. These levels are:

1) Level I. These experiments can be conducted in comparatively brief periods of experiment duration. The function of these experiments is to assure that the thruster is operating under nominal conditions.

2) Level II. These experiments can also be conducted in comparatively brief periods of time. The function of these experiments is to examine short-term behavior in space that cannot be effectively duplicated in laboratory facilities.

3) Level III. These experiments also utilize the specific operational conditions of the space environment but will require "prolonged" operation in space.

Examples of experiments in Level I are thruster start-up, short-term running, and restart experiments. Charged particle transport, thruster internal erosion measurements, and thrust beam plasma/space plasma electrical equilibration are examples of Level II experiments. Both Level I and Level II experiments can be considered effective within operational time constraints of the order of 10 hours. An example of a Level III experiment (where "prolonged" can be intended to mean of the order of 100 hours of operation) is material transport to and deposition on the deposition plates.

The use of operational time flexibility allows a flight experiment to configure to the time constraints of a given Orbiter flight. A seven day Orbiter mission permits experiments at Levels I, II, and III. If the operational period of the Orbiter is reduced to one or two days, Level III experiments could not be carried out effectively but effective pursuit of experiments in the Level I and Level II categories would be possible.

2.4 GROWTH MODES IN THE ION THRUSTER FLIGHT EXPERIMENTATION

Sections 2.2 and 2.3 have discussed the concept of an iterated series of flight experiments, in view of payload recoverability and in view of the multiple mission Orbiter capability. These iterated flight experiments can, as noted, act to amortize thruster and diagnostic payload costs. An iterated flight series also permits growth modes in its original design and permits the initial experiment to assume more modest (and cost effective) goals (and more modest start-up costs) than if only a single time flight experiment is planned.

Section 3.0 will present the FLIGHT EXPERIMENT DEFINITION. For completeness there, an extensive series of tests will be designated and defined. The initial flight experiment will, however, contain only a sub-group of this larger experiment list. Reasons will be presented there for the selection or the de-selection of a specific test for this initial thruster flight test. Additions to the basic flight package for subsequent flights can then be determined using the previous flight experience as well as any other (then) available Orbiter or laboratory data. The presently important aspect of experiment design is that the initial flight experiment be capable of expansion into the several possible growth modes and it is believed that the flight experiment design in Section 4.0 is capable of such later add-on capabilities.

In addition to growth modes in the flight testing of a single 8-cm thruster, there are two other growth modes which should be considered. These additional growth modes are:

1) Flight experiments involving multiple thrusters ("cluster" effect studies) which could utilize, in principle, modular add-ons to the initial single thruster test package, and,

2) Flight experiments involving substitution of other ion thrusters (perhaps of varying engine diameter) within the original 8-cm thruster test package.

A modular capability (growth mode 1, above) is clearly present for the thruster test package to be described in Section 4. A substitution capability (growth mode 2, above) will require specific re-examination in terms of the volumes and power requirements of the ion thruster and its associated power processing units.

2.5 ION THRUSTER COMPATIBILITY WITH OTHER PAYLOAD ELEMENTS

It is not possible, in principle, to determine absolutely the compatibility of the ion thruster with other payload elements in the Orbiter bay, in the absence of detailed knowledge of the properties of those other payloads. In practice, however, it may be considered that the ion thruster will not be capable of interference with other payload elements. The reasons for this compatibility are described more fully in Section 3 where it is shown that detectability of material transport products of the ion thruster with the deposition plates (which are a portion of the thruster diagnostic array) will require very sensitive post-flight examination, using sophisticated surface analyses. In order to have detectability of material accretion on these plates, moreover, it is required that the plates be in close proximity to the thruster. It is quite unlikely, thus, that the ion thruster will be capable of material transport impact on other payload elements.

While the discussion above and in Section 3 indicates that the thruster will be compatible with remaining payload elements, conservatively based mission planning and integration should re-examine these questions as each specific Orbiter payload becomes defined.

2.6 SHUTTLE FLIGHT TEST VERIFICATION CONCEPT

Section 2.1 has identified a set of goals described as a SHUTTLE FLIGHT TEST VERIFICATION CONCEPT. In this testing approach, the joint use of ground based testing and testing on the Shuttle Orbiter would be used to demonstrate flight readiness for an ion thruster for other spacecraft applications. Because the period of operation on the Orbiter is limited, endurance testing (both in steady state operation and in cycled operation) would be carried out in ground based facilities. The operation of the thruster in the Shuttle Orbiter would demonstrate the following:

1) Total system (thruster plus power processor plus digital interface unit) integrity through the spacecraft launch.

2) Total system start-up and operational capability under zero gravity conditions and the thermal and environmental conditions of space in the Orbiter/space equilibration.

3) Total system restart capability through a pre-determined set of thruster close-downs and restarts.

4) Operational compatibility (thermal, and conducted and radiated electromagnetic interference) with the host spacecraft and with the remaining payload elements.

The recoverability of the thruster and its post flight examination are a valuable element in this verification of thruster flight readiness.

The SHUTTLE FLIGHT TEST VERIFICATION CONCEPT has particular applicability to the ion thrusters which may emerge in future developments of these engines. Such ion thrusters could entail either major modifications from previously developed thrusters (for example, variations in engine diameter or in propellant material) or minor reworkings of such previous thrusters (for example, component changes). In either condition, it appears desirable to provide a simplified, two component, testing approach using both laboratory measurements and Shuttle measurements to demonstrate flight readiness for newly developed ion engines, and the utilization of both flight and ground based operation may be able to reduce the required total resources for flight readiness verification compared to those required resources using only a single means (either ground or space) of verification testing.

3. FLIGHT EXPERIMENT DEFINITION

3.1 ION BEAM PLUME AND EFFLUX DOCUMENTATION TESTS

3.1.1 Ground Based Laboratory Measurements

The measurements of the beam and efflux characteristics of the 8-cm thruster in ground based testing facilities provide a base for the definition of the flight experiment. These ground based laboratory results and the FLIGHT EXPERIMENT PLANNING FACTORS in Section 2 lead to a series of flight test experiments whose principal goals are the determination of the various material transport, erosion, and deposition fluxes as a result of thruster operation, and the determination of the electrical equilibration between the thrust beam plasma and the space plasma. Section 2 has also described the SHUTTLE FLIGHT TEST VERIFICATION CONCEPT which is an implicit goal in the Shuttle Orbiter flight test.

The relevant ground based test results are contained in two references. Reference 1 (G. K. Komatsu and J. M. Sellen, Jr., "Beam Efflux Measurements," Report No. NASA CR-135038 (1 June 1976)) describes an extensive series of measurements of various ion fluxes from a mercury ion bombardment thruster. The thruster in these measurements was a 30-cm diameter mercury ion engine. The results, however, are applicable to the 8-cm flight program, particularly in terms of "facility generated" efflux components which impose (ultimately) a limit on the use of ground based testing and call for resolution via the "unbounded geometry" flight test condition. A second reference, Reference 2 ("Ion Engine Auxiliary Propulsion Applications and Integration Study," S. Zafran, ed., TRW Final Report (to be published Fall of 1977)) contains the laboratory measurements of 8-cm thruster beam and efflux characteristics for both a "baseline" thruster and a thruster equipped with a sputter shield. This reference contains, in addition, numerical analyses of the potential impact of the thruster effluxes upon operational spacecraft and establishes the permissible level of such material transport for specific spacecraft and spacecraft missions. Both of the references above are considered to be applicable documents under the present flight experiment definition study and it is advised that the results in both of these previous programs be reviewed as introductory material for the present report. For convenience, this report will review the method of definition of normalized effluxes (ε) and will describe the (generally) permissible levels of ε for specific spacecraft and spacecraft missions.

3.1.2 Normalized Thruster Effluxes

In analyses of material transport and deposition for ion thrusters on spacecraft, a useful formalism is the normalized efflux. This normalized efflux can be stated for either charged or neutral particle fluxes. For charged particles in the thrust beam plume, three ion species are of interest. These are the thrust ions (described as Group I ions), which possess high energies and move along the thrust beam axis within a comparatively limited (30 degree half angle) cone of directions, the Group II ions (which possess high energies but emerge over a much broader cone of directions, albeit at greatly reduced flux levels) and the Group IV ions (which are created in charge transfer reactions in the ion beam plume and emerge at high divergence angles but with low energies. For the thrust ions the normalized efflux is

$$\varepsilon_{+,t} = \frac{J_{+,t}}{J_B} \tag{1}$$

where $J_{+,t}$ is the current density of thrust ions in amperes per square centimeter at a given point in space and J_B is total thrust beam current in amperes (using the specific convention for current nomenclature for the 8-cm thruster). The units of $\varepsilon_{+,t}$ are in cm^{-2}. For the Group II and Group IV ions the normalized effluxes are

$$\varepsilon_{+II} = \frac{J_{+II}}{J_B} \tag{2}$$

and

$$\varepsilon_{+IV} = \frac{J_{+IV}}{J_B} \tag{3}$$

where J_{+II} and J_{+IV} are Group II and Group IV ions flux density in amperes per square centimeter.

Normalized effluxes may also be stated for the neutral efflux components. For these effluxes it is understood that the current density of a neutral flow (atoms per square centimeter per second) is expressed in terms of an equivalent current (amperes per square centimeter) and, thus, ε retains the dimensions of cm^{-2}. For an 8-cm thruster equipped with a sputter shield, three neutral efflux terms are of interest. These are

$$\varepsilon_o = \frac{J_o}{J_B} \quad , \tag{4}$$

where J_o is the equivalent current density of non-ionized mercury atoms from the 8-cm thruster bombardment discharge and neutralizer discharge,

$$\varepsilon_{mag} = \frac{J_{mag}}{J_B} \quad , \tag{5}$$

where J_{mag} is the equivalent current density of sputtered metal atoms from the thruster accelerator grid, and

$$\varepsilon_{mas} = \frac{J_{mas}}{J_B} \tag{6}$$

where J_{mas} is the equivalent current density of sputtered metal atoms from the thruster sputter shield. For the final efflux term (ε_{mas}) the principal sputtering agent is the thrust ions. The efflux term, ε_{mag}, results from the sputtering actions of thrust ions and Group III charge exchange ions.

3.1.3 Constraints on Permissible Efflux Levels

The normalized effluxes defined in 3.1.2 above and the mission factors of total thruster throughput and permissible material accumulation (or erosion) lead to the allowable upper bounds on a given ε. For thrust ion interception on a spacecraft surface, and assuming a sputtering ratio of unity on thrust ion impact, and that maximum allowable surface erosion during the flight is 10^{17} atoms per square centimeter, it follows that

$$\int_o^T J_{+,t} dt \le 10^{17} \ cm^{-2} \tag{7}$$

where T is the total mission life time. If the total thruster throughput on the mission is 2×10^{25} ions (a specific case for North-South stationkeeping over a 7 year period on a 1000 kilogram spacecraft), it follows that

$$\varepsilon_{+,t} \leq 5\ (10)^{-9}\ \mathrm{cm}^{-2} \tag{8}$$

where use has been made of

$$J_{+,t} = \varepsilon_{+,t}\ J_B \tag{1}$$

and

$$\int_o^T J_B dt = 2 \times 10^{25} \text{ ions (see above).} \tag{9}$$

Surface placements in a spacecraft bearing an ion thruster must, thus, satisfy a condition of $\varepsilon_{+,t} \leq 5(10)^{-9}\ \mathrm{cm}^{-2}$, and both laboratory and flight measurements of $\varepsilon_{+,t}$ must be capable of detection of particle effluxes at this level.

Upper bound constraints upon ε_{+II} and ε_{+IV} are less well defined than for $\varepsilon_{+,t}$ because both of these ion effluxes are at smaller sputtering ratios than the thrust ions and, hence, have larger allowable mission integrated total fluxes. A conservative position on thruster integration, however, will maintain the same upper bound on ε_{+II} as upon $\varepsilon_{+,t}$. A reasonable upper bound on ε_{+IV} is $5(10)^{-8}\ \mathrm{cm}^{-2}$ where acknowledgement has been made of the reduced sputtering action of the Group IV ions.

Allowable upper bounds on ε_o are more difficult to define than for the thrust ions and Group II and Group IV ions. This difficulty in definition results because the low energy mercury atoms are capable of re-evaporation from spacecraft surfaces (unless these surfaces are at very low temperatures) and there is no present evidence of any long term surface alteration as a result of mercury atom residence there prior to re-evaporation. Metal atoms, however, from the thruster accelerator grid and the sputter shield will not, in general, re-evaporate from the spacecraft surface upon which they initially impinge and, hence, will require an upper bound on their arrival level. Assuming that total arrival during the mission must remain below 10^{17} atoms per square centimeter leads to

$$\varepsilon_{mag} < 5(10)^{-9} \text{ cm}^{-2} \tag{10}$$

and

$$\varepsilon_{mas} < 5(10)^{-9} \text{ cm}^{-2} \quad . \tag{11}$$

The upper bound levels on ε derived above establish a required sensitivity level on both the ground based and the flight experiments. In order that the flight results will be applicable to spacecraft integration needs, the experiments must be capable of defining with sufficient accuracy the location in space of the contours along which the various normalized effluxes are at their permissible upper bound levels.

3.1.4 Thruster Test Definition

A thruster test definition for a flight experiment includes the test objective, the sensor requirements, the instrumentation requirements, the in-flight procedure, the test duration, the requirements of the Orbiter, and possible post-flight activities. These items have been described for a series of ten flight experiments (designated T1 through T10). For convenience in its use, this TEST DEFINITION PACKAGE is given as Appendix A of this report. For purposes of the present discussion, selected portions of the TEST DEFINITION PACKAGE will be given in this section.

Figure 1 illustrates the ion thruster, the sputter shield and the principal planes in which the ion beam plume measurements are to be made. The first plane is designated as the Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane and the plane normal to this first plane has been designated as the Transverse Plane. The Retarding Potential Analyzer/Faraday Cups and the Floating Probe used in the ion beam plume measurements will move in either one or the other of these two planes.

Table 1 lists the Test Title and the Test Designation for the 10 tests and their respective sub-tests. These tests may also be grouped into more general categories. Table 2 presents this grouping in terms of plume measurements, efflux and deposition effects measurements, charged particle drainage measurements, sputter shield effectiveness measurements, thruster internal

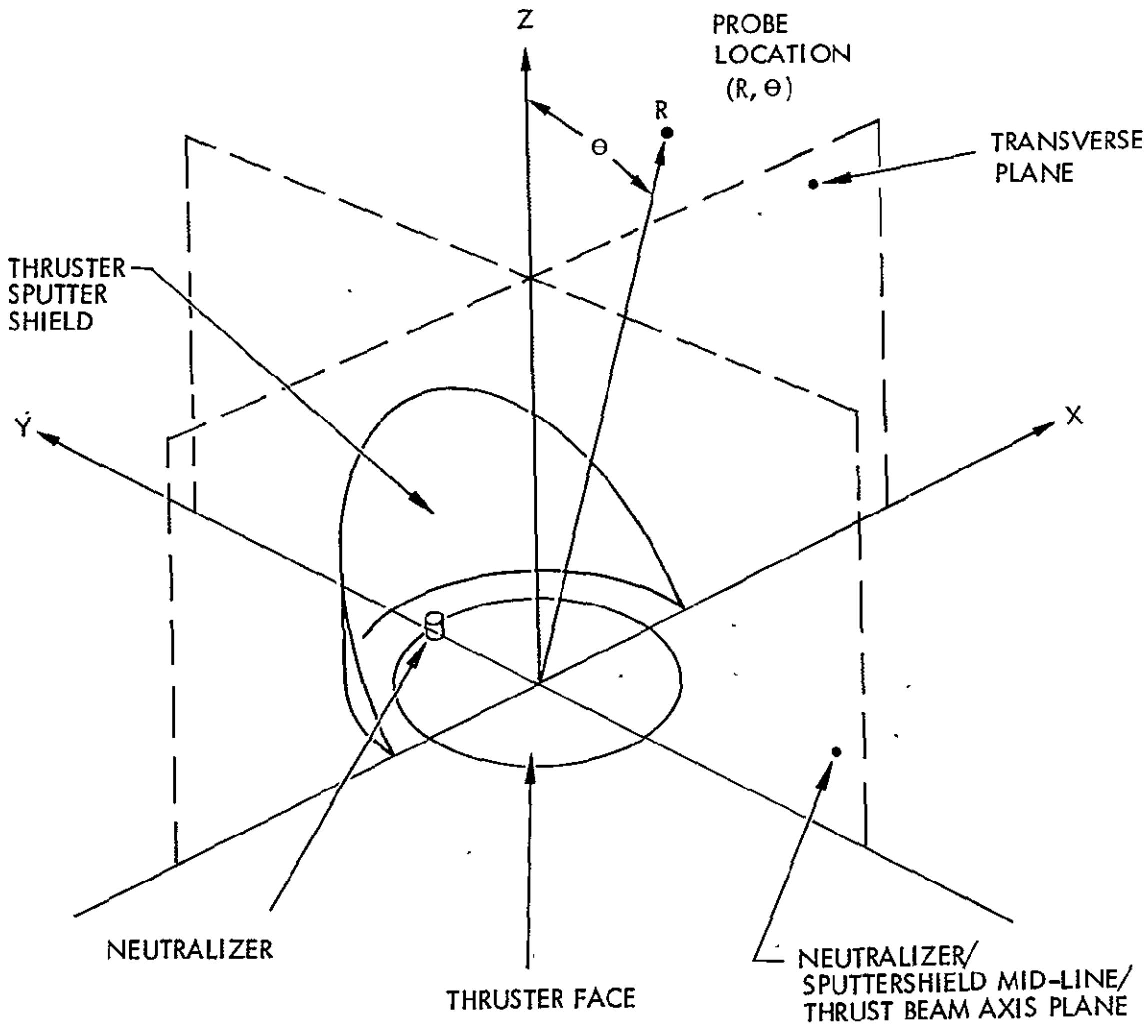


Figure 1. Ion Thruster, Ion Thruster Sputter Shield, and the Principal Planes of Measurements for the Ion Beam Plume.

Table 1. Ion Beam Plume and Efflux Characterization Flight Test Titles and Designations

Test Designation	Test Title
T1	Group I (Thrust) Ion Plume Measurements
T1A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T1B	Transverse Plane Measurements
T2	Group II (High Energy High Angle) Ion Plume Measurements
T2A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T2B	Transverse Plane Measurements
T3	Ion Thrust Beam Neutralization Measurements
T3A	Thrust Beam Plasma Potential Measurements
T3B	Thrust Beam Neutralizing Electron Temperature Measurements
T4	Group IV (Charge Exchange) Ion Plume Measurements
T4A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T4B	Transverse Plane Measurements
T5	Condensible Neutral Efflux Measurements
T5A	Deposition Plate Measurements
T5A1	Fixed Position Deposition Plates
T5A1a	In-Flight Deposition Effects Measurements
T5A1b	Post-Flight Deposition Effects Measurements
T5A2	Movable Position Deposition Plates
T5A2a	In-Flight Deposition Effects Measurements
T5A2b	Post-Flight Deposition Effects Measurements
T5B	Quartz Crystal Microbalance Measurements
T6	Non-Condensible Neutral Effects Measurements
T6A	Ionization Gauge Measurements
T6A1	Fixed Position Ionization Gauge
T6A2	Movable Position Ionization Gauge
T6B	Residual Gas Analyzer
T7	Thruster Internal Erosion Measurements
T8	Charged Particle Drainage to Electrically Biased Surface Measurements
T9	Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurements
T10	Multiply Charged Ion Production Measurements

Table 2. Ion Beam Plume and Efflux Flight Test Measurement Areas and Associated Flight Test Designations

MEASUREMENT AREA	OVERALL TEST DESIGNATION									
	T1	T2	T3	T4	T5	T6	T7	T8	T9	T10
PLUME MEASUREMENTS	●	●	●	●						●
EFFLUX AND DEPOSITION EFFECTS MEASUREMENTS					●	●				
CHARGED PARTICLE DRAIN-AGE MEASUREMENTS				●				●		
SPUTTER SHIELD EFFECTIVE-NESS MEASUREMENTS	●	●		●	●	●				
THRUSTER INTERNAL EROSION MEASUREMENTS							●			
ELECTRICAL EQUILIBRATION MEASUREMENTS			●						●	

FOLDOUT FRAME 1

Table 3. Test Titles and Objectives

T1: Group I (Thrust) Ion Plume Measurements

Objective: The objective of the Group I Ion Plume Measurements is the determination of Hg^+ thrust ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T2: Group II (High Energy High Angle) Ion Plume Measurements

Objective: The objective of the Group II Ion Plume Measurements is the determination of high energy high angle Hg^+ ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T3: Ion Thrust Beam Neutralization Measurements

Objective: The objective of the Ion Thrust Beam Neutralization Measurements is the determination of the thrust beam plasma potential and the thrust beam plasma neutralizing electron temperature as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" plane.

T4: Group IV (Charge Exchange) Ion Plume Measurements

Objective: The objective of the Group IV Plume Measurements is the determination of low energy, high angle, charge exchange Hg^+ ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T5: Condensible Neutral Efflux Measurements

Objective: The objective of the Condensible Neutral Efflux Measurements is the determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster and the surface properties effects of such effluxes at selected locations in the thruster system coordinate space.

FOLDOUT FRAME 2

T6: Non-Condensible Neutral Efflux Measurements

Objective: The objective of the Non-Condensible Neutral Efflux Measurements is a determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster at selected locations in the thruster system coordinate space.

T7: Thruster Internal Erosion Measurements

Objective: The objective of the Thruster Internal Erosion Measurement is the determination of the rate of material loss at specified internal locations of the ion thruster during in-flight operation.

T8: Charged Particle Drainage to Electrically Biased Surfaces Measurements

Objective: The objective of the Charged Particle Drainage to Electrically Biased Surfaces Measurement is the determination of the charged particle flow from the ion thruster exhaust plume to specified surfaces at varying levels of electrical bias and under varying degrees of insulating encapsulation.

T9: Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurements

Objective: The objective of the Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurement is the determination of the Orbiter electrical potential relative to the potential of the space plasma for varying orientations between the thrust beam vector, $\vec{v}_+$, and the Earth's magnetic field vector, $\vec{B}_e$, and for varying configurations of the ionosperic plasma wake (created by Orbiter motion through the space plasma) and the ion thruster beam plasma.

T10: Multiply-Charged Ion Production Measurements

Objective: The objective of the Multiply Charged Ion Production Measurements is to determine the ratio of doubly charged thrust ions to singly charged thrust ions (Hg^{++}/Hg^+) as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" plane.

erosion measurements, and electrical equilibration measurements. As a further clarification of the purpose and content of the various tests, Table 3 lists the test titles and the objectives of the various tests.

The several tests in Tables 1-3 permit a successful completion of the goals of the thruster flight experiment as these goals have been described in Section 2 and in this present section. The tests described in this series would also permit an evaluation of the interaction between the ion beam plume and surfaces with an electrical potential. The charged particle drainage determination goal had not been previously identified, and, for reasons to be discussed in sections to follow, will be de-emphasized in the initial, first flight, test configuration. In compliance with the Statement of Work, however, provision for a charged particle drainage test has been included in this section and in the TEST DEFINITION PACKAGE.

The tests described in Tables 1-3 may also be viewed in terms of the Level I, Level II, and Level III categories described earlier in Section 2. In this regard, Level I tests (to assure that nominal thruster operation exists and requiring only short term operation periods) are Tests T1, T2, T3 and T4. Level II tests (also of short duration but now directed to the more subtle "boundless geometry" efflux generation and flow characteristics) include Tests T1, T2, T3, T4, T9 and (possibly) T7. The Depositions Effects Measurements (T5) and (possibly) the Thruster Internal Erosion Measurements (T7) are Level III (long operational period, requiring "boundless geometry") category tests. The use of the Level I, II, and III categories will be useful in later discussions examining the impact on the overall flight experiment planning if the Orbiter flight operational period should be reduced below the seven day point, or, (for even the seven day mission), if other and competing payload demands result in a diminished operational period for the ion thruster flight experiment.

3.1.5 Proposed Initial Flight Experiment

The tests described in Tables 1-3, including all of the various sub-tests, constitute a very extensive series of thruster flight examinations. As earlier sections have pointed out, however, there may be many expected constraints on the thruster experiment ranging from the overall Orbiter on-orbit time to the competing demands of other payloads. There are at present, moreover, many factors concerning the Orbiter environment which are not well known. For these reasons, and because the flight test definition has been in terms of a serial experiment with a developing capability, the initial flight test package will consist of a simplified test series utilizing several of the tests in Table 1, but de-selecting other elements of this test series. This section will identify the (proposed) selected first flight configuration and will discuss briefly the reasons for test de-selection where this occurred. The overall goal for this simplified first flight package is to retain experiment effectiveness but in the context of reduced, first-flight, hardware costs and for reduced, first-flight, requirements on the Orbiter. As in-flight experience grows, and, because the flight experiment hardware can be supplemented with additional diagnostics in later flights, it is anticipated that others of the (presently) de-selected tests will be included in the flight experiment.

Table 4 lists the selected and de-selected first flight experiments together with a brief description of the reason(s) for de-selection. The selected experiments include ion plume measurements for Group I, Group II, and Group IV ions (in both principal planes), ion thrust beam neutralization measurements, fixed position deposition plates (analyzed post-flight), thruster internal erosion meausrements, and thrust beam/space plasma/Orbiter electrical equilibration measurements. The de-selected experiments include both fixed and movable deposition plates utilizing in-flight analysis and movable deposition plates utilizing post flight analysis. The reasons for this de-selection include the anticipated experiment cost and complexity for in-flight analysis (as Section 3.2 will discuss, genuine thruster deposition levels are at very low levels and are, thus, difficult to detect even via post-flight laboratory methods), and the possibilities of simultaneously present Orbiter contaminants (of presently unknown species and flow rates) which can mask the genuine thruster deposition materials. A final element here for de-selection of movable deposition plates includes both costs and complexity as well as the possibility of the generation of cross-contaminants as the arms bearing the movable position plates will also be subject to deposition and/or erosion by the thruster plume constituents.

The reasons of experiment cost and complexity and Orbiter contaminants are also present in the de-selection of quartz crystal microbalances, ionization gauges, and residual gas analyzers. It should also be emphasized

Table 4. Selected and De-Selected Tests for an Initial Orbiter Ion Thruster Flight Experiment

Selected Experiments	T1, T2, T3, T4, (Both A, B) T5A1b T7 T9
<u>De-Selected Experiments</u>	<u>Reasons for De-Selection</u>
T5A1a, T5A2a, T5A2b	Experiment cost and complexity Possible Orbiter contaminants Possible cross-contaminant generation
T5B, T6A1, T6A2, T6B	Experiment cost and complexity Possible Orbiter contaminants
T8	Competing effects of space plasma
T10	Experiment costs and complexity Laboratory measurements may be sufficient

here, however, that some of these instruments will be included on other Orbiter payloads and the ion test flight planning will be able to benefit from these Orbiter contaminant measurements without direct cost to the ion thruster flight experiment. If the Orbiter generated contaminants are at significant levels, the detection of thruster generated depositions will be difficult and may require sophisticated post-flight analyses with all in-flight analyses as beyond reasonable possibilities and pursuit.

Test T8, Charged Particle Drainage to Electrically Biased Surface Measurements, has been de-selected because of the presence of the comparatively dense ionospheric plasma at this Orbiter altitude. As Section 3.4 will develop, it will be possible to eliminate some of these ambient plasma effects by appropriate Orbiter orientation (in order to carry out T4A and T4B at high angles). It will be even more difficult, however, to eliminate ambient plasma effects in Test T8, even for an optimally oriented Orbiter.

A final de-selection, Test T10, Multiply Charged Ion Production Measurements, has been carried out because of experiment costs and complexity (an additional and complicated probe will be required for this measurement) and because laboratory measurements of these ion species may be sufficient (there being, at present, no demonstrated "facility effect" in multiply charged ion generation).

3.1.6 Shuttle Flight Test Verification Concept Experiments

The goal of the Shuttle Flight Test Verification Concept is a demonstration of flight worthiness, through integration, launch, and re-entry, of the thruster system. Although this goal is not stated explicitly in the tests in Tables 1-3, that goal is present (implicitly) in the testing. In addition to the implicit goal of flight verification in the designated tests, a series of start-restart tests have been listed in the Flight Experiment Schedule (Section 3.1.7). During the start-restart exercises, a Retarding Potential Analyzer/Faraday Cup is rotated to a given polar angle position and is then held fixed in this position as the thruster is cycled from an OFF state to an ON state and, after close-down, back to the OFF state. The Faraday cup outputs and the ion thruster currents and voltages are observed during both turn-on, steady state, and turn-off periods as the test proceeds through its pre-determined number of cycles. This start-restart exercise has been scheduled for the final day in Orbit for the seven day mission. In the shorter versions of the flight test, with either compelled or desired test conclusion within a four day span and, finally, a two day span, the start-restart exercise has not been scheduled.

3.1.7 Flight Experiment Schedule

Figure 2 presents an outline of the flight experiment schedule for a seven day, four day, and two day Orbiter mission. The experiment schedule here is, as noted, only qualitatively outlined and detailed time-lining of the thruster flight test cannot proceed until a specific Orbiter flight and Orbiter payload have been identified. Factors which will influence this time-lining are discussed in the requirements presentation in Section 3.4.

3.2 FLIGHT EXPERIMENT SENSOR DESCRIPTION

3.2.1 Flight Test Sensor Designations and Associated Tests

Section 3.1 has described the Ion Beam Plume and Efflux Documentation Tests and Appendix A, the TEST DEFINITION PACKAGE, provides additional detail on the flight test sensors and their operational procedure. This section, 3.2, of the flight test study will excerpt material from Appendix A and will discuss specific first flight sensor characteristics and sensitivities.

Tables 1-3 in Section 3.1 have presented the Flight Test Titles and Designations, have grouped these experiments into the broader measurement areas, and have described Test Objectives. Tables 5 and 6, presented here, provide a list of the Ion Beam Plume and Efflux Characterization Flight Test Sensors and Sensor Designations and the Designation of the Test for Each Sensor. Table 6 also contains Required Test Fixtures for the Flight Test, the Fixture Designation, and the Associated Flight Test. As may be noted from the tables, many of the tests can be performed with a relatively small number of sensors and a single test fixture, thus allowing a first flight test configuration with reduced costs and complexities but which retains a broad degree of diagnostic capability.

In addition to these Tables, Table 7, drawn from Appendix A, lists all of the sensors and their required sensitivities. The sensitivities

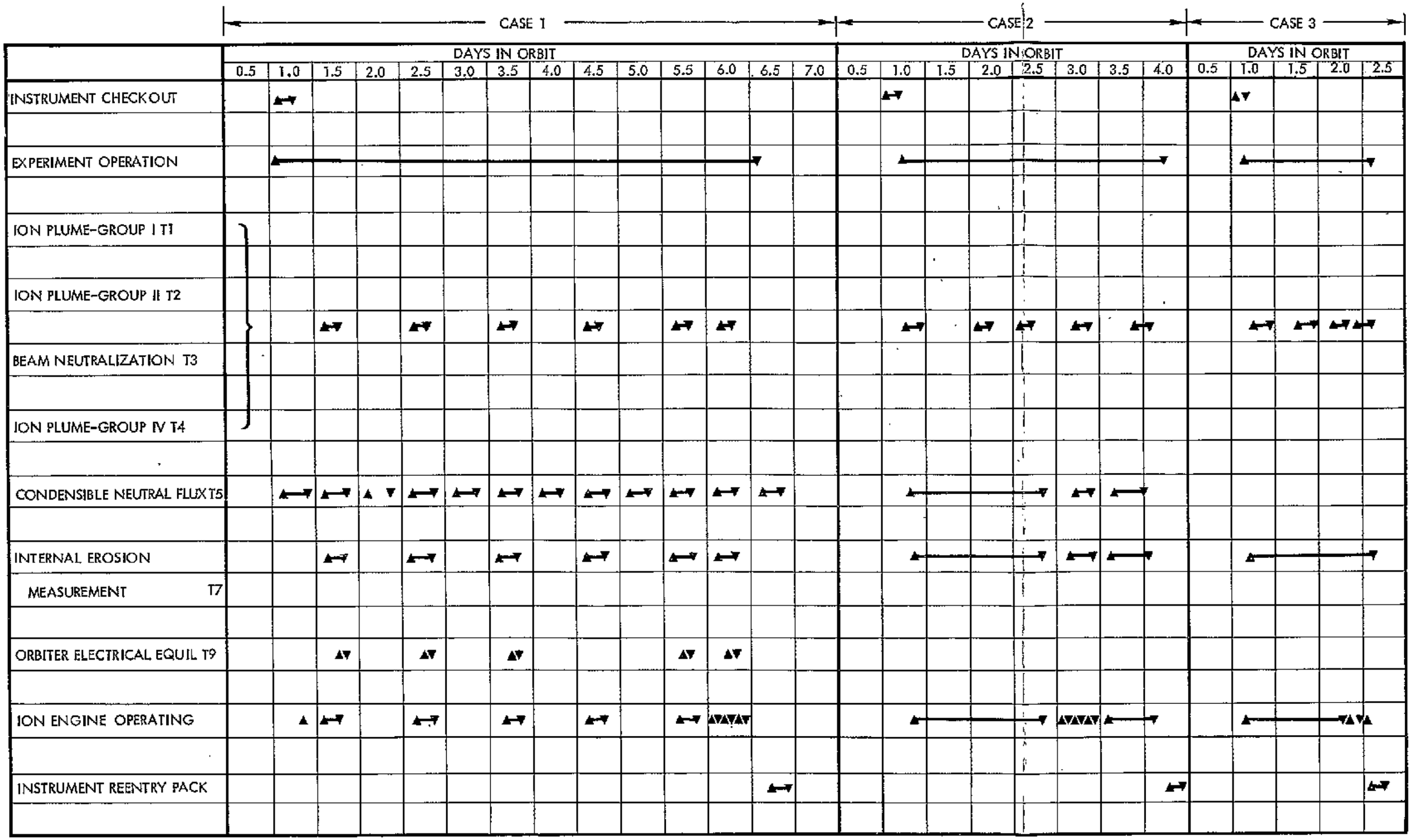


Figure 2. Flight Experiment Schedules for a Seven Day, Four Day, and Two Day Shuttle Flight Test of the Ion Thruster.

Table 5. Ion Beam Plume and Efflux Characterization Flight Test Sensors and Sensor Designation

Sensor	Designation
Retarding Potential Analyzer/Faraday Cup (Neutralizer/Sputter-Shield Mid-Line/Thrust Beam Axis Plane)	RPA/FC1
Retarding Potential Analyzer/Faraday Cup (Transverse Plane)	RPA/FC2
Floating (Cold) Potential Probe (Transverse Plane)	FPP
Deposition Plate (Fixed Position)	DPF
Deposition Plate (Movable)	DPM
Quartz Crystal Microbalance	QCM
Ionization Gauge (Fixed Position)	IGF
Ionization Gauge (Movable)	IGM
Residual Gas Analyzer	RGA
In-Flight Optical Properties Analyzer	IOA
Internal Erosion Sample	IES
Electrically Biasable Surface	EBS
Orbiter Floating Potential Probe	OFP
Multiply-Charged Ion Probe	MIP

Table 6. Ion Beam Plume and Efflux Charcterization Flight Test Sensors and Associated Test Designations and Required Test Fixtures for Ion Beam Plume and Efflux Characterization Flight Test, Test Fixture Designation, and Associated In-Flight Test Designations

Sensor	Test Designation
RPA/FC1	T1A, T2A, T4A, T8, T9
RPA/FC2	T1B, T2B, T3B, T4B, T8, T9
FPP	T3A, T3B, T8, T9
DPF	T5A1a, T5A1b
DPM	T5A2a, T5A2b
QCM	T5B
IGF	T6A1
IGM	T6A2
RGA	T6B
IOA	T5A1a, T5A2a
IES	T7
EBS	T8
OFP	T9
MIP	T10

Required Test Fixtures

Fixture	Designation	Test
Thruster Test Fixture	TTF	T1, T2, T3, T4, T5, T6, T7, T9, T10
Remote Test Fixture	RTF	T8

Table 7. Ion Thruster Flight Test Sensor and Sensitivity Requirements

Sensor	Sensitivity Requirement
Retarding Potential Analyzer/ Faraday Cup (RPA/FC)	Lower End Current Density Sensitivity, 10^{-8} A/cm^2, for Ion Group I, II, IV
Floating Potential Probe (FPP)	1 Volt in Plasma Floating Potential
Deposition Plates (DPF and DPM)	Lower End Deposition Level Sensitivity, $5(10)^{16}$ particles/cm^2
Quartz Crystal Microbalance	Lower End Deposition Level Sensitivity, 10^{15} particles/cm^2
Ionization Gauge (IGF and IGM)	Lower End Flux Density Sensitivity, $3(10)^{11}$ particles/cm^2/sec
Residual Gas Analyzer (RGA)	Lower End Flux Density Sensitivity, $3(10)^{11}$ particles/AMU/cm^2/sec
In-Flight Optical Properties Analyzer (IOA)	Lower End Deposition Level Sensitivity, 10^{16} particles/cm^2
Internal Erosion Sample (IES)	Lower End Erosion Level Sensitivity, 100 Angstroms
Electrically Biasable Surface (EBS)	Requirements are Mission Specific
Orbiter Floating Potential Probe (OFP)	1 Volt in Plasma Floating Potential
Multiply-Charged Ion Probe (MIP)	Lower End Current Density Sensitivity, 10^{-6} A/cm^2 for Hg^+, 10^{-8} A/cm^2 for Hg^{++}

of specific sensors will be examined in further detail in Section 3.2.2 for the proposed initial flight configuration.

3.2.2 Description of Sensors for Proposed Initial Flight Test Package

3.2.2.1 Retarding Potential Analyzer/Faraday Cups

Sensor Configuration. The Retarding Potential Analyzer/Faraday Cups (RPA/FC1 and RPA/FC2) are required for Tests T1, T2, T3, and T4 (both A and B in all tests) and for T8 and T9. One of these sensors (RPA/FC1) moves in the Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane and the other sensor (RPA/FC2) moves in the Transverse Plane. Section 3.2.2.4 will describe these cup movements and positioning in further detail.

Figure 3 provides a detailed illustration of the Retarding Potential Analyzer/Faraday Cup. Both RPA/FC1 and RPA/FC2 are built to this configuration. The three grids shown there consist of a first grid and third grid which will be held at small and constant negative voltages and a middle, variable potential, grid which is placed at a series of potentials ranging from zero volts to positive potentials of several hundred volts. The purposes of these grids are as follows:

- First Grid: The (small) negative potential on this grid prevents electrons from the thrust beam plasma or the space plasma from being attracted to and collected at the (positively biased) middle grid. The prevention of electron drainage to the middle grid aids the experiment operation by removing a possible current drainage load on the power supply providing the middle grid bias potential and also prevents any disruption of the thrust beam neutralization by the thruster plasma discharge neutralizer.

- Middle Grid: The (varying) positive potential on this grid either prevents or allows the passage of an ion through the analyzer (depending on ion energy) and thus provides an analysis of the ion flow into the constituent groups, Group I, Group II, and Group IV.

- Final Grid: The small (negative) potential on this grid suppresses both secondary emission electrons (from energetic ion impact) and photoemission electrons (from the solar ultraviolet) at the surface of the two ion collectors. Suppression of secondary and photoelectrons is vital in order that the Faraday cups retain their lower end current sensitivity requirements.

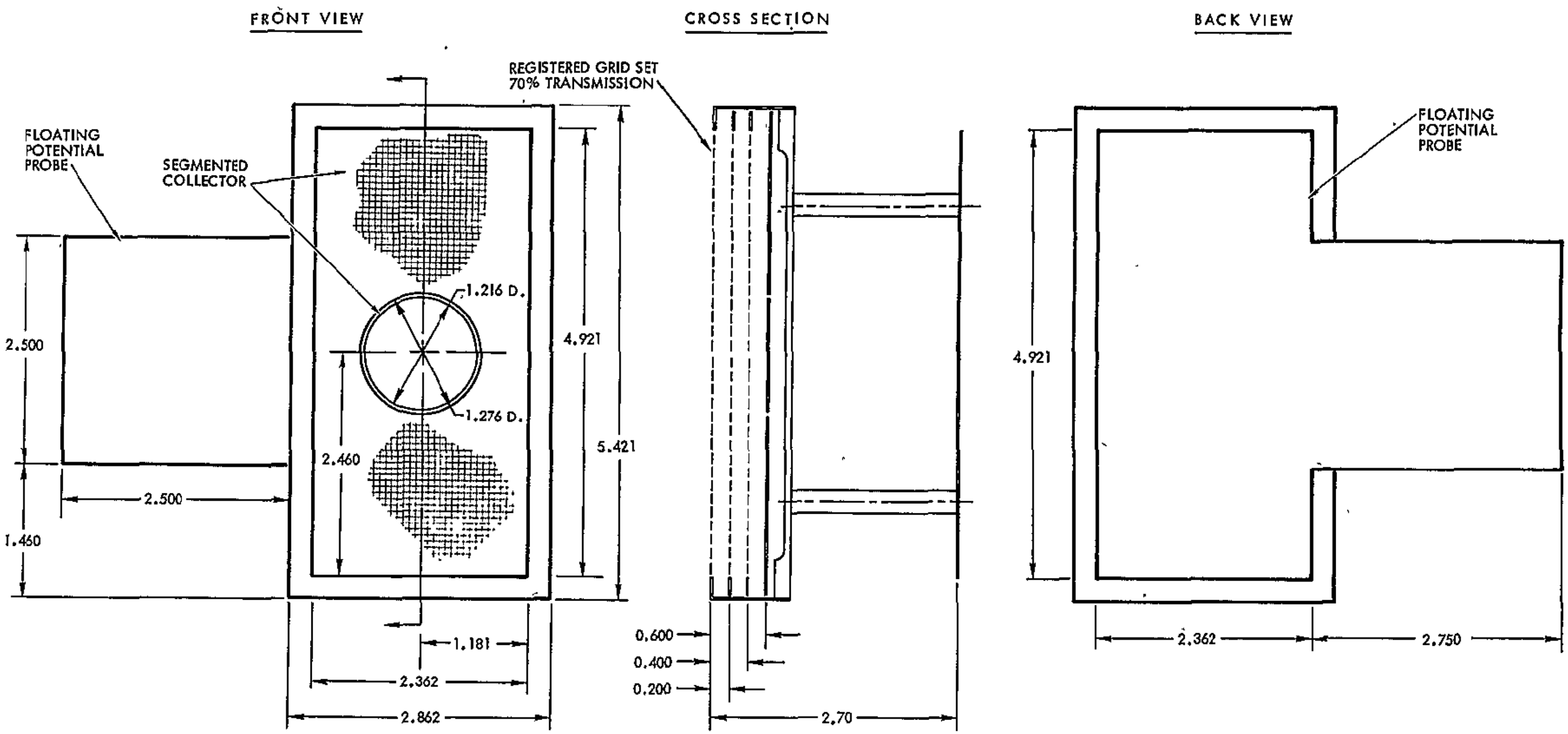


Figure 3. Retarding Potential Analyzer/Faraday Cup Construction
(Floating Potential Probe Attached to Rear of Cup).

Each Retarding Potential Analyzer/Faraday Cup contains two ion collectors arranged in an inner ring and an outer plate. The area of the outer plate is approximately one order of magnitude larger than the inner ring. The purpose of these collectors is as follows:

- Inner Collector: The inner collector, because of its smaller size, provides more definition to the measurements of ion current density as a function of the polar angle. The principal use of the inner collector is in the more dense regions of the thrust beam plume. If sufficient readout electronics capability is present, however, the inner collector can be used throughout the entire swing (±90°) in polar angle.

- Outer Collector: The larger size of the outer collector prevents this sensor element from having the angular definition of the inner ring. At large polar angles, however, the outer area aids in improving the signal level of (primarily) Group II and Group IV ion flows. For these latter two ion flows, angular resolution requirements are reduced because of comparatively reduced rates of flow variation with polar angle.

Sensor Sensitivity. The notion of normalized thruster effluxes is introduced in Section 3.1.2, and Section 3.1.3 has described constraints on permissible efflux levels. For Group I and Group II ions, permissible ε levels have been set at $5(10)^{-9}$ cm^{-2} for a specific example of a North-South stationkeeping communications spacecraft. Using $J_B \sim .1$ ampere for this example mission leads to upper bounds on Group I and Group II ion fluxes of $\sim 5(10)^{-10}$ amperes per square centimeter at the point of spacecraft surface placement in the example mission. The placement of spacecraft surfaces will be, of course, highly specific to each spacecraft utilizing such an ion thruster (in addition to mission dependences). For the Shuttle Orbiter flight test and for the comparatively close spacing between the thruster and the sensor location used there (see Section 4 for a complete design of this package) it is probably acceptable that the sensor have a lower end current measuring capability of 10^{-8} A/cm^2 and it is this sensitivity figure which will be utilized here. (Note that an ion flow of 10^{-8} A/cm^2 at $R \sim 30$ cm (the general level of thruster-to-probe separation distance for the Shuttle flight test package) will have diminished to $\sim 10^{-9}$ A/cm^2 at $R \sim 1$ meter (which is generally characteristic of the separation distances between the thruster and spacecraft surfaces on long-term operational spacecraft) because of the "spherical" expansion (as $\sim 1/R^2$) of the ion flow for those distances, R, large compared to the source size (in this instance, the 8 centimeter diameter of the ion thruster)).

The lower end current flow capability of 10^{-8} A/cm^2 and the ion collector areas ($\sim$ 10 cm^2 for the inner collector, and $\sim$ 90 cm^2 for the outer collector) leads to lower end current signals of $\sim 10^{-7}$ A and $\sim 10^{-6}$ A on the two collectors. In order to assure that this lower end capability is maintained without significant loss of measurement accuracy, the electronics readout packages for these collectors have been designed to maintain accuracy for signal levels of 10^{-8} A from the inner collector and 10^{-7} A from the outer collector. If this electronics package capability can be maintained, the lower end current sensitivity of the Faraday cups will be at 10^{-9} A/cm^2, representing an improvement of an order of magnitude over the previously stated 10^{-8} A/cm^2 lower end flux capability and the sensitivity of the ion flux. The determinations on the Shuttle Orbiter Flight Test would exceed by approximately one order of magnitude the sensitivity required for future integration and application efforts for ion thrusters on operational spacecraft. Such an excess of sensitivity could be of value if specific future spacecraft applications should develop with particularly stringent surface property requirements.

3.2.2.2 Floating Probe

Sensor Configuration. The Floating Potential Probe (FPP) is required for Tests T3A and T3B and Tests T8 and T9 in the overall test series and for T3A, T3B, and T9 in the proposed first flight experiment.

Figure 4 illustrates this Floating Potential Probe. It consists of a single metal plate, electrically isolated from the probe mounting arm, which connects through a cable to a high input impedance voltmeter. When the probe enters the thrust beam plasma, the plate acquires the floating potential of the plasma. The measurement of this plasma floating potential (by the high impedance voltmeter) as the probe moves through the thrust beam plasma determines the effectiveness of the thrust beam neutralizer by the plasma discharge neutralizer (Test T3A). The combination of these floating potential measurements with the plasma density measurements (T1, T2, T4 by RPA/FC1 and RPA/FC2) and the use of the "electrostatic" barometric equation provides a measurement of the thrust beam neutralizing electron temperature (Test T3B).

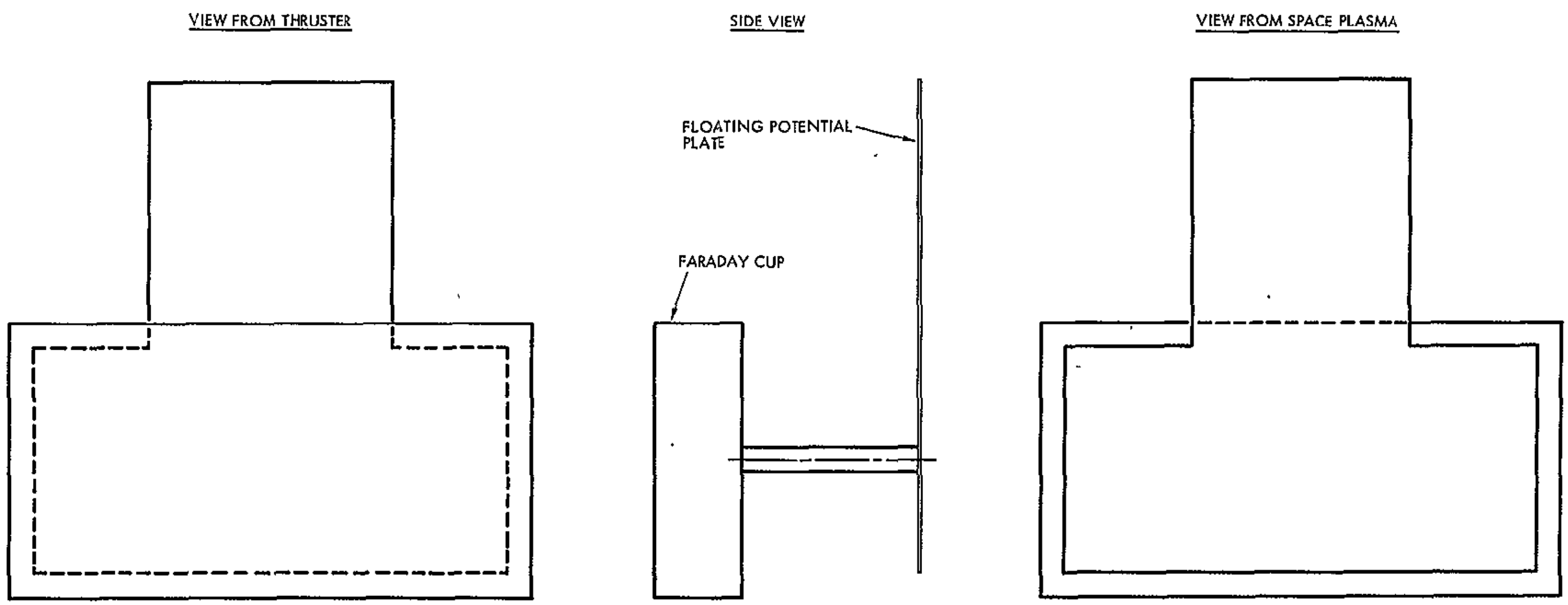


Figure 4. Floating Potential Probe Construction.

The flight test sensor designations and sensor assignments in Tables 5 and 6 indicate that both an Orbiter Floating Potential Probe (OFP) and the Floating Potential Probe (FPP) are required for the electrical equilibration measurements of Test T9. The OFP probe, however, requires a separate test fixture (RTF) for its mounting and, in the simplified payload version of the proposed first flight, this remote test fixture has not been included. It may be possible, nevertheless, to obtain the necessary data for the electrical equilibration measurements by using only the Floating Potential Probe and by positioning this probe in different locations. The first set of measurements of potential would be obtained in the thrust beam plasma and the second set of measurements would be obtained with the probe rotated out of the thrust beam plasma and into the more dilute surrounding regions of the ambient space plasma. Such probe movements require operational time and there is no present evidence to indicate that sufficient operational time is not available. If later, and more accurate, experiment time-lining of the experiment should indicate that sufficient time is not available for probe movement, then an additional floating potential probe can be mounted to the second probe mounting arm. For the present configuration, however, only a single Floating Potential Probe is included in the payload package and it is mounted so that it moves in the transverse plane. For either condition (one or two floating probes) it will be required that the probe move into the ambient space plasma in order to acquire the space plasma floating potential for Test T9 to be carried out. The requirement for probe immersion in this ambient space plasma will lead to specific requirements on the Shuttle Orbiter attitude as will be discussed in Section 3.4. The arrangement of the probe relative to the Faraday cups and the thruster is described further in Section 3.2.2.4.

Sensor Sensitivity. The effectiveness of the Floating Probe is determined by its surface area and the input impedance of the voltage measurement circuit. The probe surface area requires a minimum of 10 cm^2 and minimum input resistance is 10 megohms. The voltage circuit measurement accuracy should be within 1 volt (for 10% accuracy at floating potentials of 10 volts, 1% accuracy of floating potential at 100 volts). Maximum floating potential measurement capability is at +200 volts relative to

Orbiter ground.

The use of the floating probe for both thrust beam plasma and ambient space plasma floating potentials sets up some conflict in probe area requirements. Measurements in the ambient space plasma may require a larger probe surface area than measurements in the denser thrust beam plasma. The use of the larger probe area, however, causes a loss of angular resolution in the Ion Thrust Beam Neutralization Measurements (T3A and T3B). This conflict has been resolved in the proposed first flight payload design in favor of Test T3, and Test T9 may experience some loss in measurement accuracy. If later re-examinations of this design choice tends to favor a growth of capability in T9, then the area of the floating probe can be increased, or a second floating probe (of larger area) can be installed on the second mounting arm, or the Remote Test Fixture can be installed with a larger area Orbiter Floating Potential Probe mounted on this fixture.

3.2.2.3 Fixed Position Deposition Plates

Sensor Configuration. The Fixed Position Deposition Plates are required for T5A1. In the proposed first flight payload, no in-flight analysis of these probes is carried out, and the experiment to be conducted is T5A1b, Condensible Neutral Efflux Measurements, Fixed Position Deposition Plates, Post-Flight Deposition Effects Measurements.

Figure 5 illustrates the fixed position deposition plate, the housing, and the aperture shutter. The housing and the aperture shutter are to prevent extraneous deposition signals from being present on the deposition plates. In the flight experiment the shutter would be opened during appropriate periods of the thruster operation to determine if deposition products (principally sputtered metal atoms) are being generated by the ion thruster. Because there will also be background contaminants from the Orbiter and its remaining payload elements, each measurement deposition plate has an accompanying "monitor" plate. The monitor plate aperture is shuttered during thruster operation (at which time the measurement deposition plate shutter is open) and the monitor plate aperture is later opened during periods of thruster inactivity) for a comparable exposure time. The signal differences between the measurement plate and the monitor plate

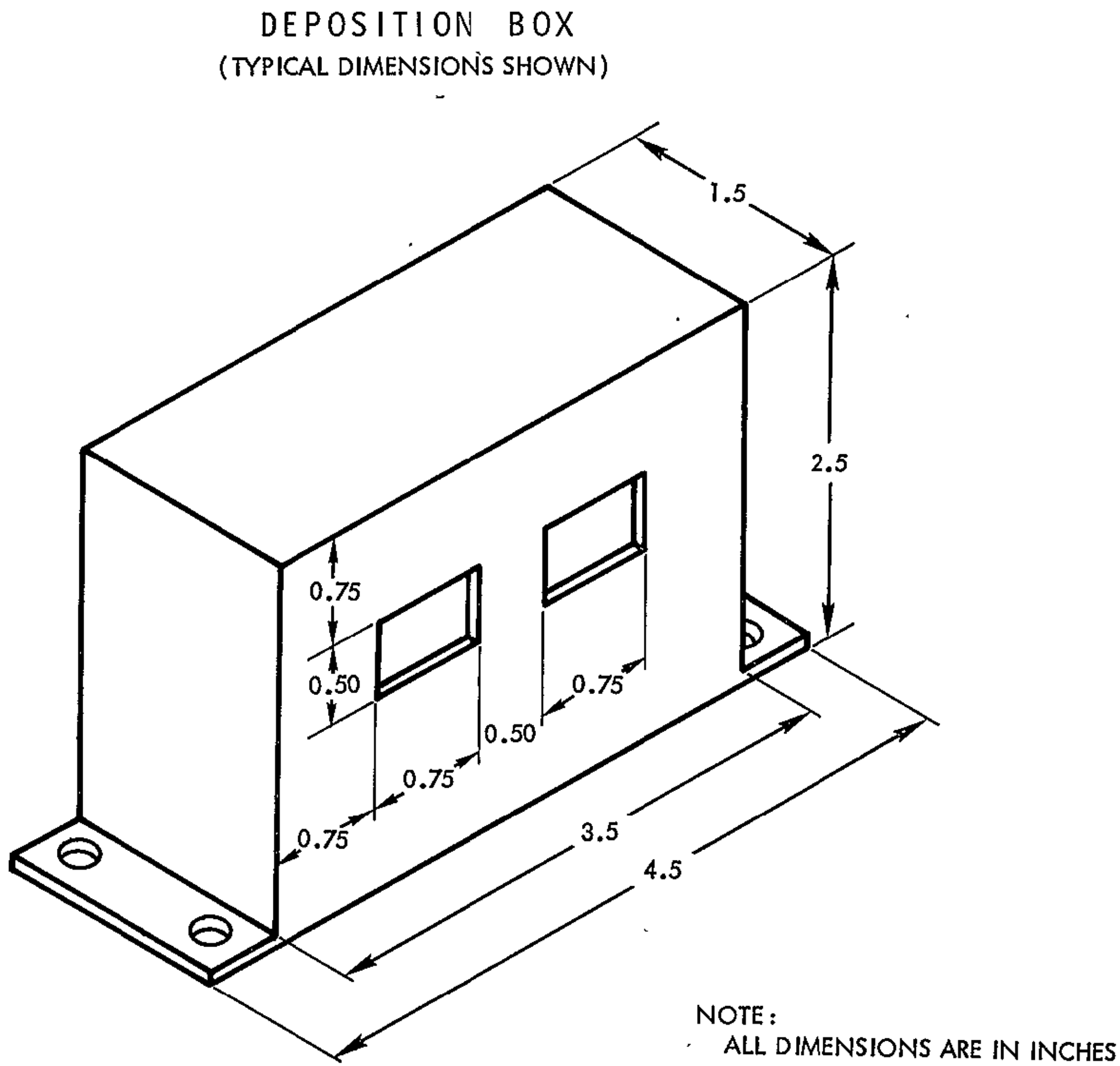


Figure 5. Deposition Plate Holder Construction.

permit an evaluation of contaminant buildups on the plates due to the Orbiter and remaining payload elements and, thus, permits an isolation and evaluation of thruster deposition products from those deposition fluxes from other sources.*

The proposed first flight payload consists of two deposition plate locations with each location containing a measurement plate and a monitor plate. The arrangement of these plates relative to the thruster is described further in Section 3.2.2.4.

Sensor Sensitivity. The detection of an accumulated layer of deposited, condensible neutral atoms on the deposition plates presents the flight test with one of its most difficult diagnoses and establishes an experiment operational time requirement which is the most demanding in view of the limited total period in orbit for the Orbiter. The upper bound limits on allowable normalized neutral effluxes, ε_{mag} and ε_{mas}, are at the same value, $5(10)^{-9}$ cm^{-2}, as for thrust ions and Group II ions (see Section 3.1.3). Because the detection of charged particles is comparatively straightforward through the current flows they create in the sensing detector, and because this current measurement approach cannot be utilized for the neutral, charge-free, depositing atoms, surface detection of the accumulated atoms must employ other methods. Before examining these methods, however, it is of interest to examine the neutral layer buildup at a given ε and for a given experiment deposition duration.

If a deposition plate is set at a position in space in which ε_{mas}, for example, is $5(10)^{-9}$ cm^{-2}, the atom arrival rate is $J_B\varepsilon_{mas}$ and for J_B of $4(10)^{17}$ thrust ions per second ($\sim$ 72 milliamperes), $J_B\varepsilon_{mas}$ is $\sim 2(10)^9$ atoms/cm^2/sec. An exposure carried out over $5(10)^5$ seconds (almost the entirety of a seven day Shuttle flight) would accumulate $\sim 10^{15}$ deposited atoms per square centimeter which is a deposition depth of approximately one monolayer. Surface detection of a one monolayer deposition by any surface measurement techniques can be difficult under even the best of exposure conditions. The Shuttle Orbiter, however, does not present the best deposition plate exposure conditions because many other contaminants may be present in quantities larger than the sputtered metal atoms of concern here, and the joint deposition of these extraneous contaminants and the sputtered metal atoms can substantially mask the detection of that

*(This analysis assumes that the rate of orbiter contaminant release is constant, irrespective of the ion thruster ON-OFF condition).

mere monolayer of deposited metal atoms from the example calculation above.

Considering the experimental conditions above, it becomes necessary to increase the signal level of the metal atoms, both in an absolute sense and also relative to the Orbiter contaminant materials. To do this, the deposition plate is placed at a point in space in which much higher ε values exist. Because the sputtered metal atoms expand in moving away from their source (either the accelerator grid or, in the present example, the sputter shield) the ε value will depend approximately as d^{-2} where d is the separation distance from the metal atom source point to the deposition plate location, provided that d is large compared to the metal atom source dimensions. To increase the ε values by two orders of magnitude, thus, the distance d is diminished by one order of magnitude from its previous, assumed, position. In practical terms, this leads to the placement of the deposition plate holder at relatively small separation distances (of the order of 30 centimeters from the source of the metal atoms) which is to the advantage of the flight test design in that it permits the mounting of the plate holder to be on the thruster test fixture itself, thus eliminating the need for any separate, remote, test fixtures.

With the placement of the deposition plate at an ε level of approximately $5(10)^{-7}$ cm^{-2}, an exposure of $5(10)^{5}$ seconds now leads to a deposition of $\sim 10^{17}$ metal atoms/cm^2 ($\sim$ 100 monolayers) which is much more capable of accurate measurement and which may be comparable to or larger than the Orbiter contaminant buildup. The example experiment, thus, would be capable of determining the position of the $5(10)^{-7}$ cm^{-2} contour, and, by using an approximately $1/d^2$ expansion of the metal atom flow, the location of the $5(10)^{-9}$ cm^{-2} contour can be estimated. It is this latter contour position, of course, that is ultimately demanded for the integration analyses of the example spacecraft mission (seven years North-South stationkeeping at geosynchronous with a 1000 kilogram vehicle).

From the discussion above, it appears that detectable buildups of $\sim$ 100 monolayers of sputtered metal atoms can result for a closely separated source-to-deposition plate configuration ($\sim$ 30 cm) for an exposure somewhat in excess of 100 hours. The methods of this post-flight plate analysis are discussed further in the following section. A reduction in experiment operation time to the 50 hour point may still permit some accuracy in the measurements of the buildup. The experiment is, however, on the edge of becoming marginal if further reductions in exposure time are carried out. For these reasons, the metal atom deposition experiment, T5A1b, has been classified as a Level III category experiment and probably will require a seven day Orbiter flight. For a four day Orbiter flight, data in this buildup can still be acquired, although at reduced levels of accuracy. At the two day Orbiter flight level, the experiment cannot be effectively carried out, assuming here that other thruster flight experiments in Level I and Level II categories will require portions of this Orbital period, and that competing demands from other Orbiter payloads will also exist.

Post-Flight Analysis. The deposition plates are one of the two flight tests requiring post-flight analysis. (See also Test T7, Thruster Internal Erosion Measurements, Section 3.3). Several methods of analysis will be discussed here and general recommendations will be made. The analysis of surface deposits at such low levels is comparatively difficult, however, and additional laboratory studies will be required in order to more accurately plan this program activity.

Three methods of surface analysis can be suggested. These methods are the electron beam microprobe, the ion beam microprobe, and ESCA (Electron Scanning for Chemical Analysis). In the program results described in Reference 2, both electron beam microprobing and ESCA were applied to deposition plates which had buildups of sputtered metal atoms. The electron beam microprobe did not have sufficient sensitivity to detect the comparatively minute levels of accumulated metal atoms. As this buildup in the laboratory is believed to be comparable to the possible buildup levels for the flight test, electron beam microprobing of the flight test samples is not recommended.

The ESCA approach to deposition plate study in the Reference 2 program was able to detect the deposited metal atoms. There were, however, severe complications in this experiment because of the presence of other contaminant materials which act to overlay the metal atoms and mask their signal. By using a sputtering clean-off of the upper contaminant layers, the underlying metal atom layers were detectable. The ESCA approach, however, does not produce an absolute determination of the metal atom layer thickness and, while ESCA may be employed in the post-flight analysis, the analysis should also utilize other approaches.

The most appealing approach to the metal atom surface layer measurement is ion beam microprobing. This method is mass specific and, used in an appropriate manner can be used to determine the absolute value of metal atom accumulation. Sputtering action of the impinging ions in the microprobe does occur, however, and the flight test sample will be altered (layers removed) as a result of the analysis process. For this reason, a pre-calibration of this method should be undertaken with known metal atom types and layer thicknesses and, perhaps, with additional contaminant materials present at levels estimated to be possible in the later, Orbiter, flight condition.

3.2.2.4 Sensor Array Configuration

Figure 6 illustrates the sensor array configuration, the principal planes of the measurements, and the orientation of the ion thruster and its sputter shield.

The RPA/FC1 and RPA/FC2 are shown in the planes of their movement in Figure 6. For RPA/FC2, which moves in the Transverse Plane, the probe mounting arm also contains the Floating Potential Probe (FPP). When the mounting arm is at its extreme position (see Figure 6), the FPP no longer couples to the thrust beam plasma and, for suitable Orbiter orientation (see Section 3.4), the FPP couples to the ambient space plasma. This coupling of the FPP to either one or the other of the two plasmas allows the electrical equilibration measurements, T9, to be carried out without the Orbiter Floating Potential Probe (see also Section 3.2.2.2).

Two sets of deposition plates (each set consisting of a deposition plate and its monitor plate) are shown in Figure 6. The first set of deposition plates examines sputtered metal atoms from the forward (irradiated) side of the thruster sputter shield. The second set of plates examines any metal atom transport into the umbra region behind the thruster sputter shield. Metal atom deposition should not be expected in this region and the deposition plate here is expected to confirm this absence of deposition. Deposition plates to determine the metal atoms sputtered from the thruster grid have not been provided in the array

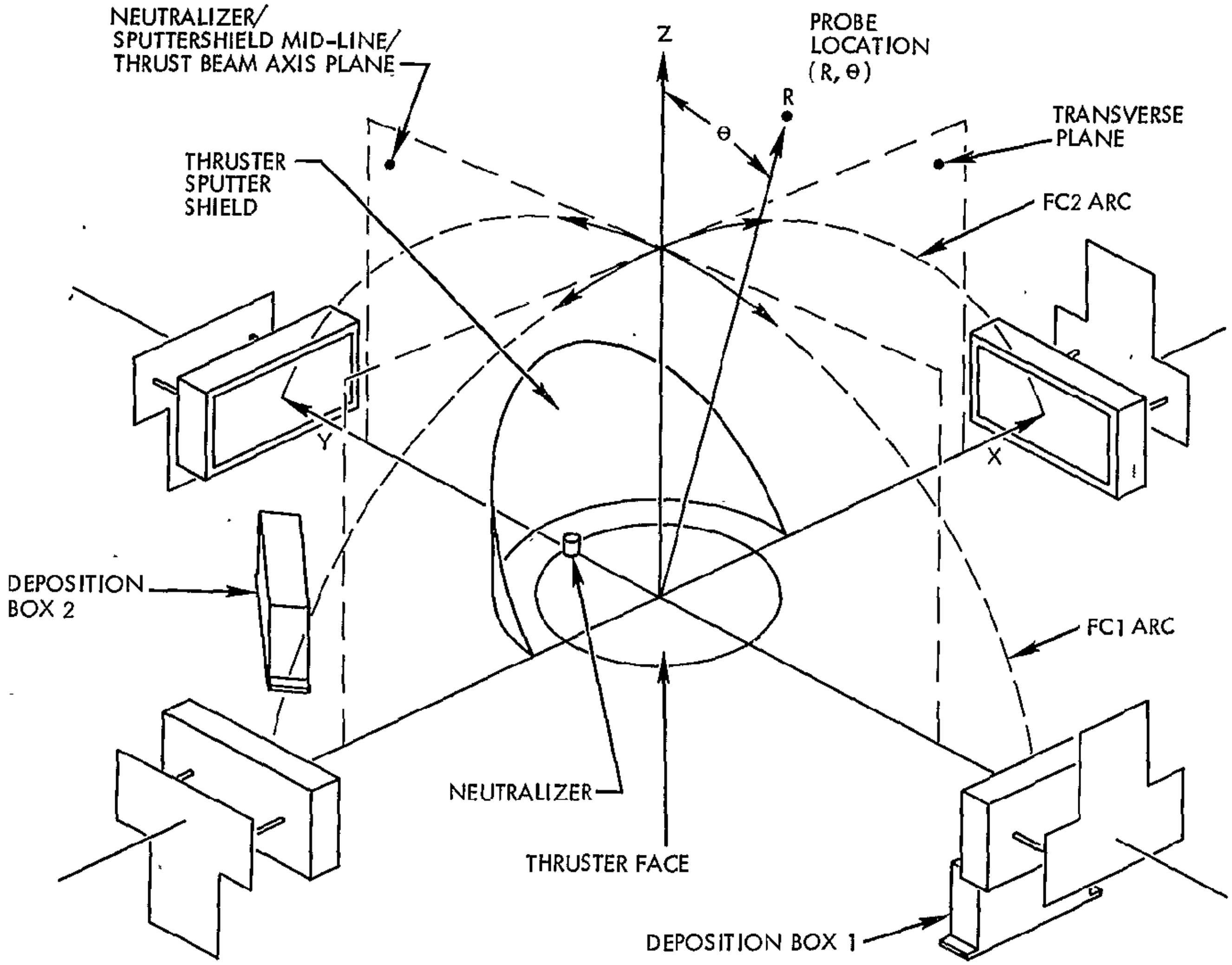


Figure 6. Sensor Array Configuration and Principal Planes of Measurement.

illustrated in Figure 6. This determination of ε_{mag} (metal atoms from the accelerator grid) has not been carried out because to examine this flow would require the elevation of the plate holder so that proper viewing of the accelerator grid is provided. The (now) elevated deposition plate holder is then subject to ion impact and sputtering which creates, in turn, an additional source of sputtered metal atoms. Thus, for simplicity in the proposed first flight payload, the deposition plates and the holder to examine sputtering products from the accelerator electrode have not been included in the diagnostic array.

3.2.3 Growth Mode Sensors

Many of the probes listed in Table 5 have not been included in the first flight sensor configuration. The thruster test flight is, however, capable of expansion in its later versions and additional diagnostic capability can be added in these follow-on flight experiments.

One probe of particular interest in the growth mode of this thruster flight test is the Quartz Crystal Microbalance (QCM). Various versions of this instrument are being prepared for other Shuttle flights and for the purpose of determining Orbiter contamination levels. Because the Orbiter payloads are recoverable, it may be possible for the thruster flight test to include (ultimately) in its diagnostic array, one or another of the QCM's being prepared for Shuttle Orbiter measurements. This would provide the thruster flight test with additional diagnostic capability at a low level of costs (perhaps refurbishment) to the thruster flight test program and would allow the thruster test to utilize a diagnostic probe which would possess, at that juncture, a considerable level of Orbiter flight experience.

3.3 THRUSTER INTERNAL EROSION MEASUREMENTS

3.3.1 Flight Experiment Definition and Conditions

The Thruster Internal Erosion Measurements experiment, Test T7, is the determination of the material removal and/or deposition at selected points within the ion thruster as the result of the thruster operation on the Orbiter and the comparison of these in-flight measurements with measurements of internal erosion for a similar thruster operating in laboratory testing facilities over a comparable period. In previous listings of this experiment, Test T7 has been designated as either of Level III category or

Level II category. In both designations the experiment presumes that operation of the experiment in the "boundless geometry" condition is of importance and the distinction between the categories is, then, the amount of thruster experiment time required to produce sufficient erosion and/or deposition for the post-flight measurements.

The experiment goals described above and the designation of the experiment level category both suggest that thruster internal erosion/deposition may be dependent upon testing facility conditions. At present, however, there is no firm experimental data base to support these conjectures. It will be of interest, nevertheless, to examine possible facility dependent reactions in internal erosion/deposition, and a following section (3.3.2) will discuss these conceptual processes. These conceptual facility dependent processes suggest, in turn, a requirement during the Orbiter flight for appropriate monitoring of the Orbiter "facility" conditions, particularly in regards to the levels and species of contaminant molecules in the vicinity of the ion thruster in the Orbiter payload bay (Section 3.3.5).

Because the measurements of thruster internal erosion/deposition will require that at least two thruster tests be carried out (one laboratory and one space) and because the thrusters under examination will require a special preparation of the internal surfaces for prompt (and accurate) determination of the material transport, some discussion will be given (Section 3.3.3) of possible methods for determining either the removal or the deposition of comparatively small quantities of material.

3.3.2 Conceptual Facility Effects in Ion Thruster Internal Erosion Processes

In the examination of ion thruster operation, emphasis is usually directed to the motion of material from the interior of the ion thruster to exterior regions. While it is correct that the bulk of material transport in the thruster is from the engine interior to the exterior regions, material from the exterior regions can move from those positions into the thruster interior if the atoms or molecules in question are appropriately directed in their velocities. This "back diffusion" of material into the thruster can be present at comparatively minute rates in laboratory testing facilities and can even occur for thrusters operating on spacecraft in space, albeit at reduced levels of transport for this space condition.

An atom or molecule entering the interior of the ion thruster can engage in several possible reactions. These are:

- The particle may become ionized and, in the ionized state, be re-transported to the thruster exterior in the same manner as a mercury ion from the bombardment discharge.

- The particle may become ionized and, in the ionized state, can impact upon thruster internal surfaces, or,

- the particle can remain in the neutral state and will move until it encounters an internal surface of the thruster.

In both the second and third conditions above the back diffusing atom or molecule has, in one fashion or another, managed to contact the thruster internal surfaces. Upon such contact, several other possible reactions may be considered. These are:

- The particle may cause surface erosion via a sputtering of an atom from the surface material, or,

- the particle may accommodate to the surface (without erosion) and may remain there until its later removal by other, later arriving, particles.

In both of the surface reactions above, and in the second and third conditions previously discussed, possibilities exist for a facility dependent effect on thruster internal erosion if the facility can determine the amount and the species of the back diffusing particles. Because the back diffusing particles in the laboratory testing chamber are, more likely than not, of different species and at different rates of arrival than the back diffusing particles in space, there is a basis for a possible difference between laboratory and space testing, and, within the framework of testing in space, variances from one spacecraft to another or, for a given spacecraft, from one period of operation to another.

While the conjectures above are of interest in the flight test planning, there is, at present, no firm data base for either the levels or the particular species which may be present at any given point in the Orbiter payload bay for the back diffusing atoms or molecules described in the truster reactions above. A detailed study of material transport in the vicinity of the Orbiter has been carried out (Reference 3, Rantanen and Ress, Payload/Orbiter Contamination Control Assessment Support, MCR 75-13, NAS9-14212, June, 1975), and, while this study has progressed to significant levels of detail, the underlying material release terms are assigned parametrically. In order to carry these considerations further, direct in-flight measurements of contaminant presence and species must be made on the Orbiter and such experiments are in the development stage for the early training flights.

3.3.3 Methods of Determining Thruster Internal Erosion/Deposition

Thruster internal erosion can be determined by two different methods, each of which possesses advantages and disadvantages. The two methods are:

a) Observations of the "color" patterns of the surface in question after thruster operation and for a coating on the surface of a multi-layer thin film "sandwich" of dissimilar (in color) metals.

b) Measurements of the surface profile using a sensitive stylus for the surface in question and after thruster operation.

Method (a), above, has been used in the surface erosion measurements reported in Reference 2, using 21 layer sandwiches of alternating thin films ($\sim$ 250 Å in thickness for each film) of chromium and copper. An advantage of the multi-layer surface coating is that it can conform to a wide variety of body or surface shapes. A disadvantage is that the material in question in the sandwich is either chromium or copper (for Cr/Cu sandwiches) and will not possess, in general, the same sputtering coefficients as the underlying base material for the ion thruster which is, after all, the material of specific importance. It may even be argued that surface accommodation of arriving atoms or molecules (see 3.3.2) will depend specifically on the arriving molecular species <u>and</u> upon the substrate material and, thus, that the multi-layer overcoat will not possess either the surface sticking coefficients or the sputtering coefficients of the material whose erosion is of specific interest. A final disadvantage in the multi-layer overcoat material is in resolution, which cannot be better than the layer thickness ($\sim$ 250 Å).

The use of a sensitive stylus for precise measurements of surface profiles has advantages in that the material examined can be the true surface material of the ion thruster and, hence, has the correct sputtering and accommodation coefficients. Resolution, for very flat specimens, can also be at precise levels of the order of 100 Å or less. The disadvantage of the method is difficulty in adaptability to the many shapes of the surfaces or objects of erosion interest in the thruster interior.

Methods of examining material depositions on surfaces include electron beam microprobes, ion beam microprobes, and ESCA. These surface deposition measurement methods have been discussed in Section 3.2.2.3.

3.3.4 Suggested Positions for Thruster Internal Erosion Measurements

The selection of the measurement points for thruster internal erosion/deposition is carried out by NASA/LeRC as a government furnished item in this study. Table 8 contains the selected sites for these measurements, grouped into erosion sites, deposition sites, and sites at which either deposition or erosion may occur.

3.3.5 Orbiter Experiment Requirements

The principal requirement for the conduction and completion of the thruster internal erosion test is in the total operational period. Perhaps the longest operational period is required for Test T5A1b, the external deposition plate measurements where periods of thruster operation between 50 and 100 hours are considered as the proper operational time. The thruster internal erosion measurements may be carried out in a somewhat shorter period in that the surface removal rates in the bombardment discharge chamber are at larger values, because of the concentration of the bombarding fluxes, than are the surface deposition rates of this transported material to the (now comparatively distant) external deposition plate sites. Thus, while T5A1b may require 50 to 100 hours, T7 can, in principle, be carried out for periods of $\sim$ 50 hours. In practice, however, it will be desirable to have as long an operational period as is possible (within the seven day Orbiter flight limit) for both T5A1b and T7. One additional consideration in this respect is that the likely situation will be that the internal erosion rates will be functions of the period of life for the thruster with one erosion rate for a freshly started engine, another erosion rate for an engine in mid-life (5000 hours to 10,000 hours) and a third erosion rate near engine terminus (15,000 - 20,000 hours). Because the Orbiter flight can explore only one of these various stages in the total life of an ion thruster, some consideration should be given to the selection for the flight

Table 8. Discharge Chamber Erosion and Deposition Sites

EROSION SITES

cathode pole piece tip and outer diameter

baffle downstream surface

screen pole piece tip
(screen grid upstream surface - center)
(accelerator grid downstream surface)

DEPOSITION SITES

anode upstream, midstream, downstream
(screen grid upstream surface - periphery)

EROSION OR DEPOSITION SITES

cathode keeper downstream surface

cathode chamber - representative location

baffle upstream surface

endplate inner and outer diameter (exposed) test of a thruster that has already had an operational period in the laboratory (perhaps, at least, through the initial, break-in, period).

A second requirement for the thruster internal erosion test is a flight log of the various material releases from the Orbiter and the time and place of these material releases. Section 3.3.2 has discussed conceptual facility effects (including Orbiter facility effects) on the material removal and/or deposition on thruster internal surfaces. Although there is, at present, no clearly evident presence of an Orbiter facility effect process in internal erosion/deposition, the Orbiter total system operation should be monitored and logged during the flight against the possibility that some of the released materials may have affected the thruster internal erosion process.

3.4 ION THRUSTER SYSTEM OPERATION REQUIREMENTS

3.4.1 General Considerations in the Thruster Flight Test Definitions Requirements

This section (3.4) will examine several requirements for the operation of the thruster in the Orbiter thruster flight test. The requirements to be examined here will be selected from a larger series of requirement areas that may be identified at present. For convenience, however, some of these other and remaining requirements will be discussed in other, and more appropriate, sections of this report.

Table 9 identifies a series of requirement areas arranged into requirement groups. Items 1, 2, and 3 in this table identify the experiment power, the experiment energy (power integral over the flight), and the experiment power/time (power demand) as requirement areas in Requirement Group A, electrical operation of the thruster and its associated diagnostic array. These power and energy requirement areas will be discussed further in Section 3.4.2. Items 4, 5, 6, and 7 identify the experiment weight, the experiment volume, and the location and orientation of that experiment volume as a second Requirement Group, B. For convenience, these requirements will be discussed in Section 4, FLIGHT EXPERIMENT CONFIGURATION in the specific context of a flight experiment design there. The thermal requirements of the experiment, Item 8 in Table 9, Requirement Group C, will also be discussed in Section 4. in the specific context of a flight

Table 9. Requirement Areas and Groups

Item Number	Requirement Area	Requirement Group
1	Experiment Power	A
2	Experiment Energy	A
3	Experiment Power/Time	A
4	Experiment Weight	B
5	Experiment Volume	B
6	Experiment Volume Location	B
7	Experiment Volume Orientation	B
8	Experiment Thermal Requirements	C
9	Experiment Propellant	D
10	Experiment Operation Period	E
11	Experiment Daylight/ Darkness Condition	F
12	Orbiter Attitude	F
13	Orbiter Orbit Altitude	F
14	Orbiter Orbit Plane Inclination	F
15	Command and Data Management	G
16	Payload Specialist Support	G
17	Orbiter Re-entry and Post-Flight Payload Handling Conditions	H

experiment configuration.

The thruster propellant requirement, Item 9, Group C, will be discussed in Section 3.4.3. This requirement is not specific to the experiment configuration. For extensive periods of thruster operation, the requirement can be affected by experiment duration. For the comparatively brief Orbiter flight, however, only a minimal quantity of propellant is required.

Item 10, Experiment Operation Period, Requirement Group E, will be one of the major requirements of the flight experiment. This requirement will be discussed in Section 3.4.4 and has also been discussed in various other aspects of the experiment in several portions of this report.

Items 11, 12, 13, and 14, which constitute Requirement Group F, designate requirement areas in the daylight or darkness condition of the Orbiter, the Orbiter attitude relative to its orbital velocity vector, the altitude of the Orbiter orbit, and the inclination of the orbit plane. These requirements will be discussed in Section 3.4.5, and one of the requirement areas, Item 12, may emerge as a major planning factor in the ultimate time-lining of an Orbiter flight.

Items 15 and 16 designate the requirements of the Command and Data Management Systems (CDMS) and the Payload Specialist Support. Because of the many possible options in the CDMS, a separate section of this report, 3.5, will be utilized to discuss these payload elements. Because the CDMS design affects the required Payload Specialist Support, that item has also been placed in Section 3.5.

A final requirements area, Item 17, Group H, is the condition of the Orbiter during re-entry and in the post-flight payload handling period. Section 3.4.6 will discuss these requirements.

3.4.2 Thruster Experiment Power and Energy Requirements

3.4.2.1 Experiment Power Requirements

The experiment power requirements occur in three areas. These are:

1) the operation of the thruster,

2) the operation of the diagnostic array (including the stepper motors, the RPA/FC grid power supplies, the collector current measurement circuits, and the associated CDMS system), and,

3) the operation of the thruster experiment enclosure heaters (if necessary).

In item (1) above, it is assumed that an appropriate voltage conversion circuit will be provided for the thruster to convert the 28 ± 4 volt Orbiter DC power to the (presently) required 70 volt input to the thruster power processing unit. The required input power for the thruster during its operational periods will be 200 watts.

The power required for the operation of the diagnostic array and the associated CDMS will be specific to the methods of control and data management. In the baseline system, however, this power requirement is not a significant factor in overall experiment power requirements (see Section 3.5). The substitution of the CAMAC option (also 3.5), however, may enact power penalties (albeit at possible experiment hardware cost savings). In this present section, attention will be restricted to the baseline diagnostic system, and CDMS for which the power requirements are 5 watts.

The final element in the experiment power requirement is for the occasional use of the experiment enclosure heaters. During periods when the Orbiter is not sunlit, the payload bay temperatures can drop to levels of ∿ -150°C. To prevent a freezing of the mercury propellant in the propellant reservoir, some heating may be required. Estimates of this heating power requirement (Section 4) are less than 70 watts.

The sum of the various power requirements is not appropriate because requirements in one area may not be present during requirements in a second area. For example, the operation of the thruster provides sufficient heat into the experiment enclosure such that experiment heating (via the enclosure heaters) is not required, irrespective of the presence or absence of sunlight or of Orbiter orientation in whatever radiation is present. A maximum power requirement for the experiment is 205 watts and represents the power drain for the simultaneous operation of the thruster and the diagnostic array system.

3.4.2.2 Experiment Energy Requirements

The experiment energy requirements are determined by the power loads of the various payload elements (1, 2, and 3 in Section 3.4.2.1) over the duration of their various operations. For a 100 hour thruster operation, the thruster energy requirement is 20 kilowatt-hours. The energy requirement for the enclosure heaters over a seven day Orbiter flight and for a low altitude, low inclination orbit (∿ 50% sunlit, ∿ 50% darkness) is estimated at 1.5 kilowatt hours. For this present example flight experiment the total energy requirements are 22.0 kilowatt hours.

While the energy requirements for the 100 hour Orbiter flight experiment described above does not represent a large fraction of the available energy from the present Orbiter fuel cell system (∿ 850 kilowatt hours for a standard H_2/O_2 tankage loading), the extension of the thruster test time to 1000 hours (assuming here that the on-orbit time of the Orbiter can be increased, ultimately, to the 1000 hour point) would lead to experiment energy requirements of 200 kilowatt-hours. This latter figure does represent a significant loading on the fuel system and does point out potential difficulties for long term thruster testing in space on the Orbiter under its present energy generation system.

3.4.2.3 Experiment Power/Time Requirements

The power as a function of time requirements for an experiment can be a significant requirements area if the experiment power represents a large fraction of the available power generation capability of the Orbiter fuel cell or if the thruster experiment chooses to operate during periods of high power demand by other payload elements. The present Orbiter fuel cell power capability is approximately 7 kilowatts over prolonged periods of time with shorter allowed operational periods at powers up to ∿ 11 kilowatts In either instance, the 8-cm thruster experiment does not represent a significantly large load and will not, by itself, lead to power consumption problems on the Orbiter. (Note that a "cluster" experiment of several simultaneously operating 30-cm thrusters can lead to a full load condition on the fuel cell, for such possible "growth mode" experiments). The remaining possible problem area here is the power/time demand of a thruster experiment for many other operating payloads such that the fuel cell system is at a maximum load condition. This latter problem cannot be addressed here and must be re-examined in terms of a specific Orbiter flight and a specific set of operating payload elements.

3.4.3 Thruster Experiment Propellant Requirements

During normal operation (1 millipound thrust) the 8-cm ion thruster uses approximately 1.64 millipounds of mercury propellant per hour. Over the 100 hour operation period discussed as an estimated flight experiment duration (and carrying out all levels of the flight test), the total propellant consumption is 0.16 pounds.

The propellant reservoir capability is for ∿ 20 pounds of mercury and, if fully loaded, would represent a propellant mass approximately 125 times the amount of mercury required for the 100 hour test. There is no apparent reason that the major portion of this propellant cannot be off-loaded, and, in view of the comparatively light weight of the baseline flight experiment system (see Section 4), a substantial fractional reduction of payload weight can be achieved by such an off-loading. A thruster propellant reservoir with 2 pounds of mercury at launch would possess an order of magnitude more propellant than is required for the 100 hour flight test duration goal and would still represent an ∿ 18 pound reduction of payload weight when compared to a thruster system launched with a completely filled propellant reservoir.

3.4.4 Thruster Experiment Operational Time Requirements

Several preceding sections (2.2, 2.3, 3.1.7, and 3.2.2.3) have discussed the use of Level I, Level II, and Level III experiment categories and the required operational time to complete the experiments in these levels. Briefly, the requirements are:

1) Completion of Level I, Level II, and Level III will require from 50 to 100 hours of flight operation, with a preference for operational time at the upper end of this 50 to 100 hour range.

2) Completion of Level I and Level II experiments will require in-flight thruster operational periods of ∿ 25 hours. Requests for experiment operational time should not be reduced below this 25 hour level.

3.4.5 Orbiter Environmental and Orbital Requirements

3.4.5.1 Solar Ultraviolet Radiation Effects

Solar ultraviolet radiation incident on the metal surfaces of the grids and collectors of the Retarding Potential Analyzer/Faraday Cups (see Figure 3) can create photoemission electron current densities of the order of several nanoamperes per square centimeter. The measurements of charged particle fluxes, however, (and particularly for Group II and

Group IV ions) are expected to be carried out to levels as low as $5(10)^{-10}$ A/cm^2 (corresponding to normalized efflux values of $\sim 5(10)^{-9}$ cm^{-2}). These spurious photoemission electron currents can create, thus, significant errors in the measurements of ion fluxes for the lower end of the range of ion current densities.

To reduce the photoelectron emission signals, a third grid, G3, has been added in the RPA/FC. The negative potential on this grid relative to the collectors suppresses the emission of both photoelectrons and secondary electrons from collector surfaces and eliminates the bulk of these spurious current signals. Very weak photocurrents may still exist in the RPA/FC, however, from the interception of the solar UV by grid G3 and the movement of (a fraction) of such photoelectrons to the collector surfaces. It is estimated that these remaining photocurrent signals are below the point of introducing significant errors in any portion of the ion current density range. To verify that such photosignals are, indeed, negligible, the thruster flight experiment can schedule a portion of its operation for the dark (non-sunlit) portions of the orbit or the body of the Orbiter may be placed so that, even in the sunlit portions of the orbit, the thruster experiment package and the RPA/FCs are shielded from the solar radiation. A daylight/darkness experiment can, thus, be carried out without introducing major requirements on the Orbiter. In the ultimate time-lining of a specific Orbiter flight, the inclusion of such an experiment is of interest. The pursuit of the daylight/darkness experiment is not recommended, however, if the required Orbiter orientation (in the context of a particular time-lining) should impose a severe burden on the remaining flight experiment performance.

3.4.5.2 Orbiter Attitude Requirements

Orbiter Attitude Requirements Relative to Orbiter Velocity, $\vec{v}_{orb}$.

The velocity of the Orbiter in its (assumed near Earth circular) orbit is approximately 7.7 kilometers/second. Because this orbiting velocity exceeds the thermal velocities of the ions in the ambient ionospheric plasma by approximately one order of magnitude, these ions appear to flow past the Orbiter at $\sim$ the 7.7 km/sec figure stated above and will possess kinetic energies (depending on ion mass) in the range from a few electron volts to approximately fifteen electron volts. These ion "ram" currents have current densities which reach maximum values, in the daytime ionosphere and for the peak of the F2 layer, of $\sim$ 100 nanoamperes/cm^2. As Section 3.4.5.1 has noted, the ion current density measurements capability of the thruster flight experiment is expected to extend to values as low as 10^{-9} amperes/cm^2. It is essential, thus, that the configuration of the Orbiter and the thruster flight experiment be such that these ambient ion ram currents cannot mask or interfere with the measurements of the thruster ion flux currents.

Some relief from the ion ram currents may be possible through the use of the ion energy analysis capability of the RPA/FC. The ram ions possess energies in the range from $\sim$ 5 ev to $\sim$ 15 ev while the majority of the Group IV ions from the 8-cm thruster possess energies in the 25 ev to 50 ev range (see Reference 2). The voltages applied to Grid G2 of the RPA/FC can be stepped, thus, from 0 volts to +20 volts and would essentially separate the ram ion currents from the Group IV ions which could still proceed (for the most part) into the RPA/FC against the +20 volts retarding potential applied to the middle grid.

A second method of preventing the ion ram currents from entering in or being in the vicinity of the RPA/FCs is by orientation of the Orbiter body so as to create a "plasma wake" condition in the payload bay. Figure 7 illustrates an Orbiter orientation which creates an optimally sized and located plasma wake from the ionospheric plasma ions in the payload bay and in the regions around an ion thruster flight experiment payload mounted within the bay. The creation of this plasma wake condition becomes more and more difficult for payloads deployed at greater distances from the Orbiter and the desire to retain a strong plasma wake generation capability is one of the factors which led to the in-bay payload placement to be utilized in Section 4, the FLIGHT EXPERIMENT CONFIGURATION.

The Orbiter motion required to produce the ambient plasma flow condition in Figure 7 is a 90° pitch maneuver (plus roll) from the conventional (nose forward) Orbiter flight reference position. An alternative approach to the Figure 7 configuration is a 90° yaw plus a roll maneuver. The actual flight procedure to pick up this required attitude will depend, of course, on the time-lining of a specific Orbiter flight, and the Orbiter

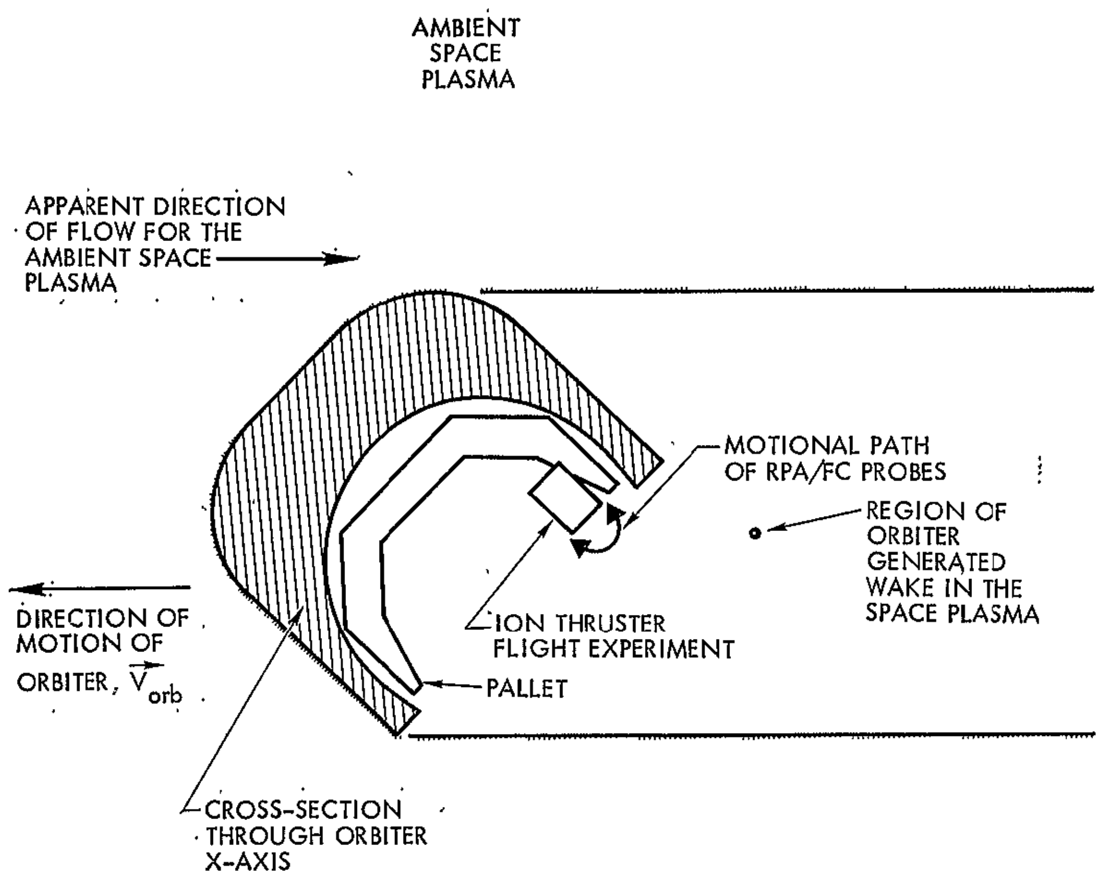


Figure 7. Attitude of the Orbiter to Provide a Plasma Wake Region in the Vicinity of the Ion Thruster Flight Experiment.

attitudes required for the experiments preceding and following the ion thruster experiment.

Of the two methods for protection of the experiment from the ion ram current signals discussed above, the most desirable course, it is believed, is the plasma wake generation approach. As will be discussed in the following section, it will also be essential for the ionospheric plasma condition near the thruster to be variable to examine the electrical coupling between the thrust beam plasma and the space plasma, Test T9.

Orbiter Attitude Requirements Relative to the Earth's Magnetic Field, $\vec{B}_e$. The electrical coupling of one plasma to another plasma will depend on the plasma density and electron temperature conditions over the regions which are common to both of the plasmas. For Test T9, the two plasmas are the thrust beam plasma and the ambient space plasma. A third element in the electrical equilibration will be, to some (as yet unknown) extent, the Orbiter. One further parameter of interest in this interaction may be the strength and the orientation of the Earth's magnetic field, $\vec{B}_e$, relative to the two plasmas and their "common" (overlap) region. For convenience, both the common region features of this interaction and the magnetic field features of this interaction will be discussed in this section, which is designated as a requirements section relative to $\vec{B}_e$.

Figure 8 illustrates several arrangements between the plasma "flow" of the ambient space plasma and the thrust beam plasma. In arrangement a, the coupling between the two plasmas is maximized, while arrangement c represents a minimum coupling condition. Arrangement b in Figure 8 possesses a coupling which is intermediate between a and c, but is probably closer to a than to c.

Section 3.2.2.2 has discussed the use of the Floating Potential Probe, FPP, as a diagnostic in Test T9 and has pointed out that the ambient space plasma must be in contact with the probe surface in order to determine the differences in plasma floating potential between the thrust beam plasma and the space plasma. In Figure 8, only configuration a will allow the FPP to establish enough contact with the space plasma to provide the necessary measurements for Test T9. Configuration 8a is then a required Orbiter orientation for the electrical equilibration experiment when that experiment

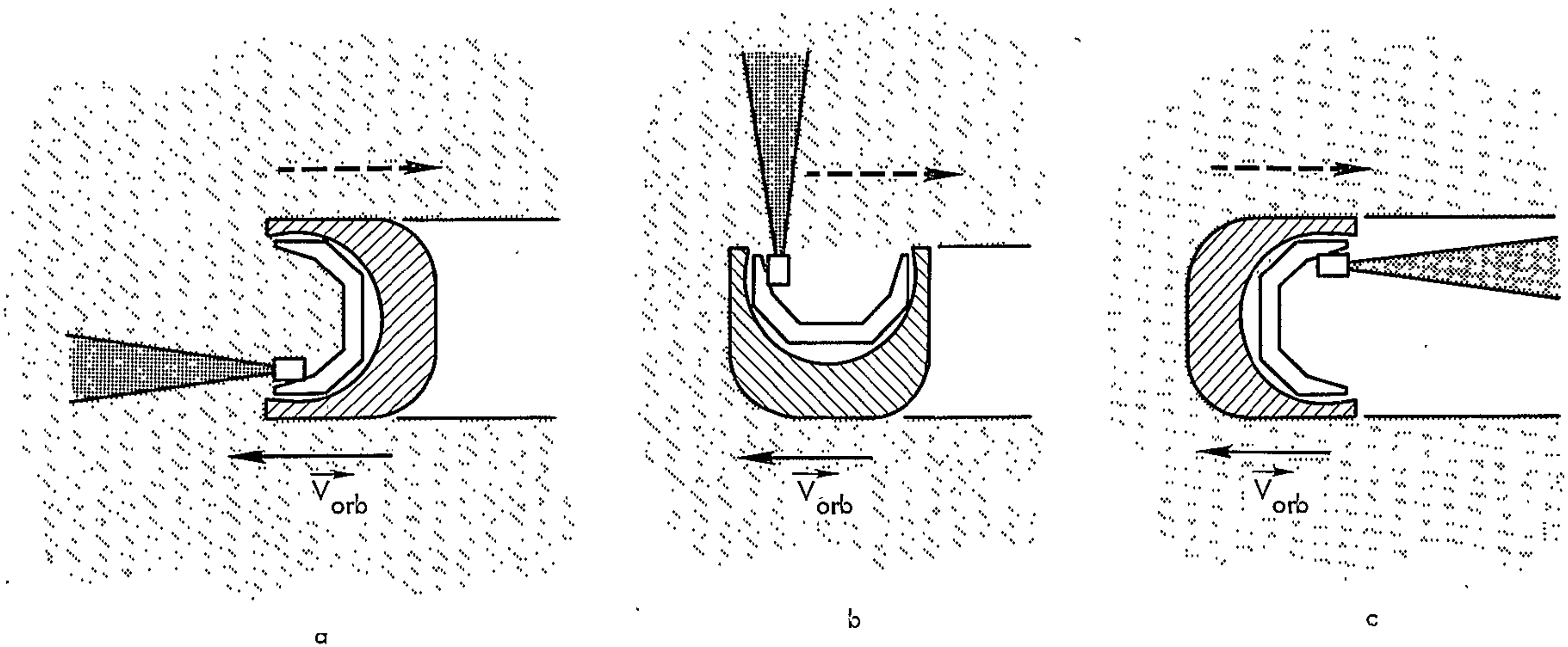


Figure 8. Possible Arrangements Between Apparent Flow Direction of Ambient Space Plasma and the Plasma Beam of the Ion Thruster.

has the FPP as its only diagnostic probe.

The extent to which the Earth's magnetic field affects the charged particle interchange between the two plasmas is conjectural at present. These conjectures also extend to the importance of the angles between $\vec{v}_+$, $\vec{v}_{orb}$, and $\vec{B}_e$, as a result of consideration of motionally generated potentials, $\vec{v}_+ \times \vec{B}_e$ and $\vec{v}_{orb} \times \vec{B}_e$, where $\vec{B}_e$ and $\vec{v}_{orb}$ have been previously identified, and $\vec{v}_+$ is the vector direction of the thrust ions.

There is no immediate and accurate method to predict the effects of the "common region" coupling between the two plasmas and the influence on this coupling of $\vec{B}_e$ orientation and magnitude, and the flight experiment stands as the most effective method, at present, of determining these effects. To carry out the experiment will require both variations in the common region conditions and in the $\vec{B}_e$ orientation. Because the FPP must remain in contact with the space plasma for the initial simplified flight payload, the configuration must remain in the form in Figure 8a. A variation of common region conditions can be carried out, however, by retaining the Figure 8a arrangement and performing the experiment both during the daytime and during the night. This day/night variation in conditions primarily affects the ambient space plasma density (which diminishes in the night-time ionosphere by approximately one order of magnitude from the day-time ionospheric values).

The variation of the orientation of $\vec{B}_e$ relative to vectors $\vec{v}_+$ and $\vec{v}_{orb}$ is more difficult to discuss in the absence of a specific Orbiter flight. For orbit planes at low inclination angles, (and assuming a circular orbit) the orbiter velocity is horizontal and primarily eastward while $\vec{B}_e$ is predominantly horizontal and predominantly along the local northern direction. For these orbit conditions, major variations in the angles between $\vec{v}_{orb}$ and $\vec{B}_e$ will not occur. The direction of $\vec{v}_+$ can be varied, of course, by reorientation of the Orbiter attitude. For an orbit plane at high inclination angles, major variations occur in the angles between $\vec{v}_{orb}$ and $\vec{B}_e$ and, thus, (together with $\vec{v}_+$ direction orientations) allows a much broader matrix of angular conditions to be set up between these several vector directions. In either instance, (small or large orbit plane inclination angles) the experiments, while interesting, do not appear of sufficient importance relative to other experiment requirements to become a primary driver in mission planning. The flight experiment planning procedure should be, after an assignment to a specific flight has been made, that the orbital history of $\vec{B}_e$ relative to $\vec{v}_{orb}$ be examined and selected points in this orbit be assigned to electrical equilibration measurements to determine if, in point of fact, the orientation of $\vec{B}_e$ is of any major consequence in the thrust beam - space plasma equilibration.

3.4.5.3 Orbiter Altitude Requirements

The electrical equilibration measurements in Test T9 require that the ambient space plasma be of sufficient density to provide an electrical coupling to the thrust beam plasma. Over the altitude range from approximately 200 kilometers to 600 kilometers, these ambient plasma density conditions are at sufficiently large levels to provide an effective plasma-to-plasma coupling. The expected orbital altitudes of the Shuttle Orbiter lie, generally, in the middle of the altitude range given above. There is, thus, no major requirement on Orbiter altitude for the ion thruster flight experiment.

3.4.5.4 Orbiter Orbit Plane Inclination Requirements

Section 3.4.5.2 (Orbiter Attitude Requirements section) has discussed orbit plane inclination in terms of possible effects on the electrical equilibration measurements. The conclusion of the discussion there is that orbit plane inclination should not be considered as a requirement for the ion thruster flight test.

3.4.6 Orbiter Re-entry and Post-Flight Payload Handling Conditions

Two thruster tests, T5A1b and T7, require a post-flight analysis of the surface conditions of materials. In the first of these two experiments, T5A1b, the material depositions on the deposition plates, can be affected by gases present in the Orbiter bay during re-entry and landing and by the gases present after the opening of the Orbiter bay and the removal of the deposition plate holders and their plates. The likely condition is that additional contaminant layers will be added to the deposition plates. It should be emphasized, however, that the ion microprobe analysis of the plates after recovery is mass specific and will not be affected by the presence of contaminants provided that the contaminant materials are not present in such large quantities as to totally obscure the underlying metal atom depositions. To prevent any such massive contaminant buildup, the deposition plates have been placed in deposition plate enclosures with shuttered apertures (see Figure 5).

The examination of the thruster internal surfaces for erosion/depositi (Test T7) may also be impacted if contaminant buildups occur. In the flight experiment design to be shown in Section 4, no provision has been made for encapsulation of the thruster during re-entry and post-landing payload removal periods. The inclusion of an encapsulation provision would incur hardware development costs, and, because of the shapes and extent of the thruster and the thruster sputter shield, would not be a simple addition to the experiment package. A recommendation for present action is that encapsulation of the thruster should not be included in the first flight payload design. The extent of the Orbiter contaminant deposition on payload bay elements will be examined in the early Orbiter flight however, and these deposition studies should be examined for possible impact on the thruster internal erosion measurements, with a possible trade study then being made between increased experiment benefits to T7 if encapsulation of the thruster is employed and increased experiment hardware costs for these additional hardware elements.

3.4.7 Operations Requirements Summary

Operations requirements discussed in this section are summarized in Table 10. Additional requirements will be summarized in Section 3.5 and in Section 4.

3.5 COMMAND AND DATA MANAGEMENT SYSTEMS REQUIREMENTS

3.5.1 Thruster Experiment Diagnostic Array Requirements

3.5.1.1 Instrumentation Electronics

Instrumentation electronics, which are separate from the electronics needed to power and control the ion thruster, are required to control the two Faraday cup positions, to amplify and digitize the sensor and housekeeping data and to format it for subsequent telemetering. In order to minimize these electronics, the conceptual design philosophy maximized the use of the Command and Data Management System (CDMS) avionics available for payload use. The CDMS, which is part of the Shuttle Spacelab, provides a

Table 10. Ion Thruster Flight Test Operations Requirements

Element	Requirement
Experiment Power	205 watts (maximum)
Experiment Energy	22 Kilowatt-hours
Propellant (1000 hour load)	2 pounds
Operation Time	50-100 hours (Levels I, II, III) 25 hours (Levels I, II only)
Daylight/Darkness Variation	Desirable
Space Plasma Wake Condition	Required
Space Plasma Ram Condition	Required
Orbiter Altitude	200 km < h < 600 km
Orbit Plane Inclination	No Requirement

dedicated experiment computer, a data display system and multiple Remote Acquisition Units (RAU).

The RAU is the prime interface for command and low rate data for experiment payloads. The unit provides both discrete and serial PCM commands for control and accepts both discrete and analog data inputs. The data inputs, called "Flexible Inputs" are software programmable and provide 8-bit analog-to-digital conversion for analog data inputs.

A block diagram of the instrumentation electronics is shown in Figure 9. It interfaces with the Spacelab through a pallet RAU for commands and data, and through an Experiment Power Distribution Box for power. The interface with the RAU consists of 19 discrete commands, 23 analog inputs and 16 discrete inputs. Not shown in the interface are the control signals and power required by the thruster electronics. The thruster electronics require three discrete commands and one serial PCM command link. It also requires an undefined number of data inputs. The instrumentation requirements versus the RAU capabilities are shown in Table 11. It can be seen that approximately one-fourth of the RAU capabilities are required.

In this initial conceptual design, the instrumentation electronics are mounted in three separate packages. Each Faraday cup has its low level electronics (LLE) circuitry mounted on the back of the cup. The LLE outputs are cabled down to the main instrumentation electronics box. The main electronics package contains the buffer and range channel electronics (signal processing electroincs), the stepper motor and solenoid drivers and the instrumentation power supply. Each of these circuits is described in more detail in the following section. If thermal considerations indicate difficulties in placing the LLE in the Faraday cup enclosures, then the LLE will be transferred into the thruster flight experiment container.

Of particular concern during the conceptual design was that the Spacelab CDMS might not be available for flight having this experiment on board. Two other configurations were examined to determine what impact these configurations would have on the hardware and software. The two other configurations examined were the hybrid (smart) pallet and direct coupling into the Orbiter avionics.

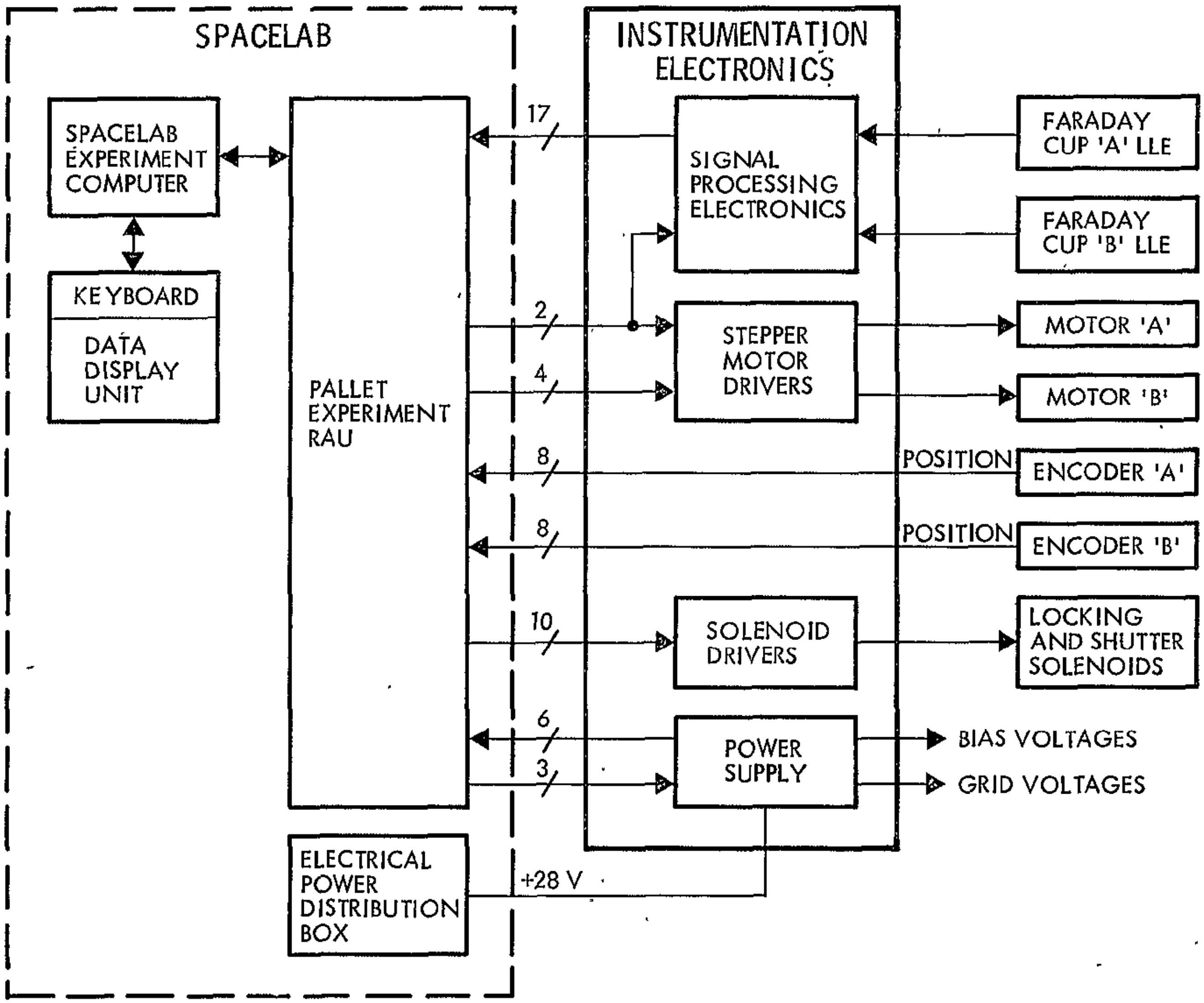


Figure 9. Instrumentation Electronics Block Diagram.

Table 11. RAU Capabilities and Instrumentation Electronics Requirements

RAU Capabilities	Requirements
1 User Time Clock	None
1 User Clock Update	None
4 Serial PCM Command Channels	1 Serial PCM Command Channel
4 Serial PCM Data Channels	None
128 Flexible Inputs	23 Analog 16 Discrete
64 Discrete Commands	22 Discrete Commands

The hybrid pallet concept provides a simulated Spacelab CDMS interface for the payload experiments using the NASA Standard Command and Data Handling avionics from the Multi-Mission Spacecraft. The experiment interface would be through a Remote Interface Unit (RIU) and would be controlled by the NASA standard spacecraft computer NSSC-I. The NSSC-I would in turn interface with the Orbiter aft flight deck data display system. The RIU interface to the instrumentation and thruster electronics is identical with that provided by the RAU so there would be no avionics hardware impact. The software however would have to be completely re-written. This is because the NSSC-I computer does not have a high order language compiler such as FORTRAN or HAL and its interface protocol with the RIU is completely different than that used by the Space lab experiment computer and RAU.

The Orbiter configuration would provide an interface through the Multiplixer-De-Multiplexer (MDM). The computer control would be provided by the Orbiter General Purpose Computer (GPC). This interface is again identical to the instrumentation interface provided by the RAU. The thruster electronics interface will change, however, because the serial command link is transmitted with transformer coupled Manchester II bi-phase code from the MDM instead of NRZ PCM code. Most of the software (FORTRAN) would be transferable to the GPC.

An alternate avionics architecture was also examined to lessen the demand on the Spacelab (or NSSC-I/GPC) computer. It is configured around CAMAC (Computer Automated Measurement and Control) and has on-board intelligence. This alternate system is described in Section 3.5.1.7.

3.5.1.2 Signal Accuracy and Dynamic Range

The desired accuracy for the ion current signal readout is five percent. This accuracy presents no problems for the RAU analog-to-digital converter (ADC) which is accurate to about one percent. The dynamic range is a problem, however, since the RAU analog-to-digital converter provides only 8-bits of resolution ($\approx 1/100$). The dynamic range for the inner plate ion current is 10×10^{-9} amps to 10×10^{-3} amp or 6 decades. Similarly, the outer plate being nine times the size of the inner plate has a dynamic range of 90×10^{-9} amps to 90×10^{-3} amps.

The dynamic range of the ion current can be handled by breaking up the range into appropriate segments that the ADC can properly resolve. The number of range segments required to maintain the five percent accuracy requirement is eight. The ranges are shown below.

RANGE	INNER PLATE			OUTER PLATE		
1	2na	to	200na	18na	to	1.8μa
2	10na	to	1μa	90na	to	9μa
3	50na	to	5μa	450na	to	45μa
4	250na	to	25μa	2.25μa	to	225μa
5	1.25μa	to	125μa	11.25μa	to	1.12ma
6	6.25μa	to	625μa	56.25μa	to	5.62ma
7	31.25μa	to	3.12ma	281μa	to	28.1ma
8	156μa	to	15.6ma	1.4ma	to	140.6ma

Table 12. Faraday Cup Collector Plate Current Ranges.

The maximum plasma potential expected is less than 100V. With a resolution requirement for this measurement of 1V, the dynamic range of the RAU ADC provides adequate resolution without range changing.

3.5.1.3 Signal Processing Electronics

The signal processing electronics as shown in the instrumentation electronics block diagram (Figure 9) are located in three separate areas. The low level electronics (LLE) for the ion current and plasma voltage measurement are located (at present) on the back sides of the two Faraday cups. The range amplifiers for these measurements are located in the main instrumentation electronics box. The signal processing electronics is shown in Figure 10.

The LLE consists of a buffer amplifier for the plasma potential and preamplifiers for the ion current measurement. These amplifiers are located at the Faraday cup to provide a low source impedance line driver to reduce noise problems and signal degradation. This is especially important when measuring ion currents of 10^{-9} amps.

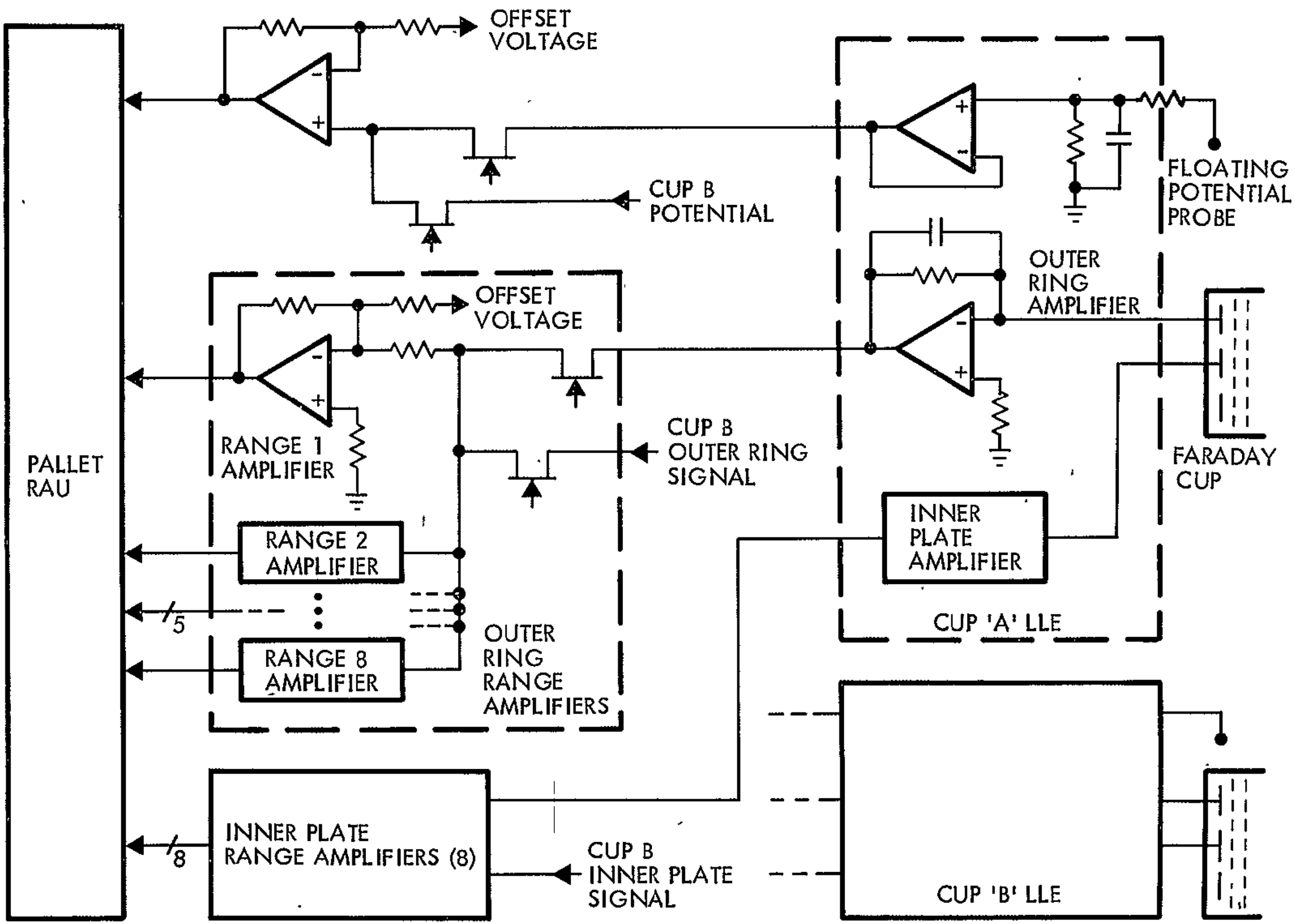


Figure 10. Signal Processing Electronics.

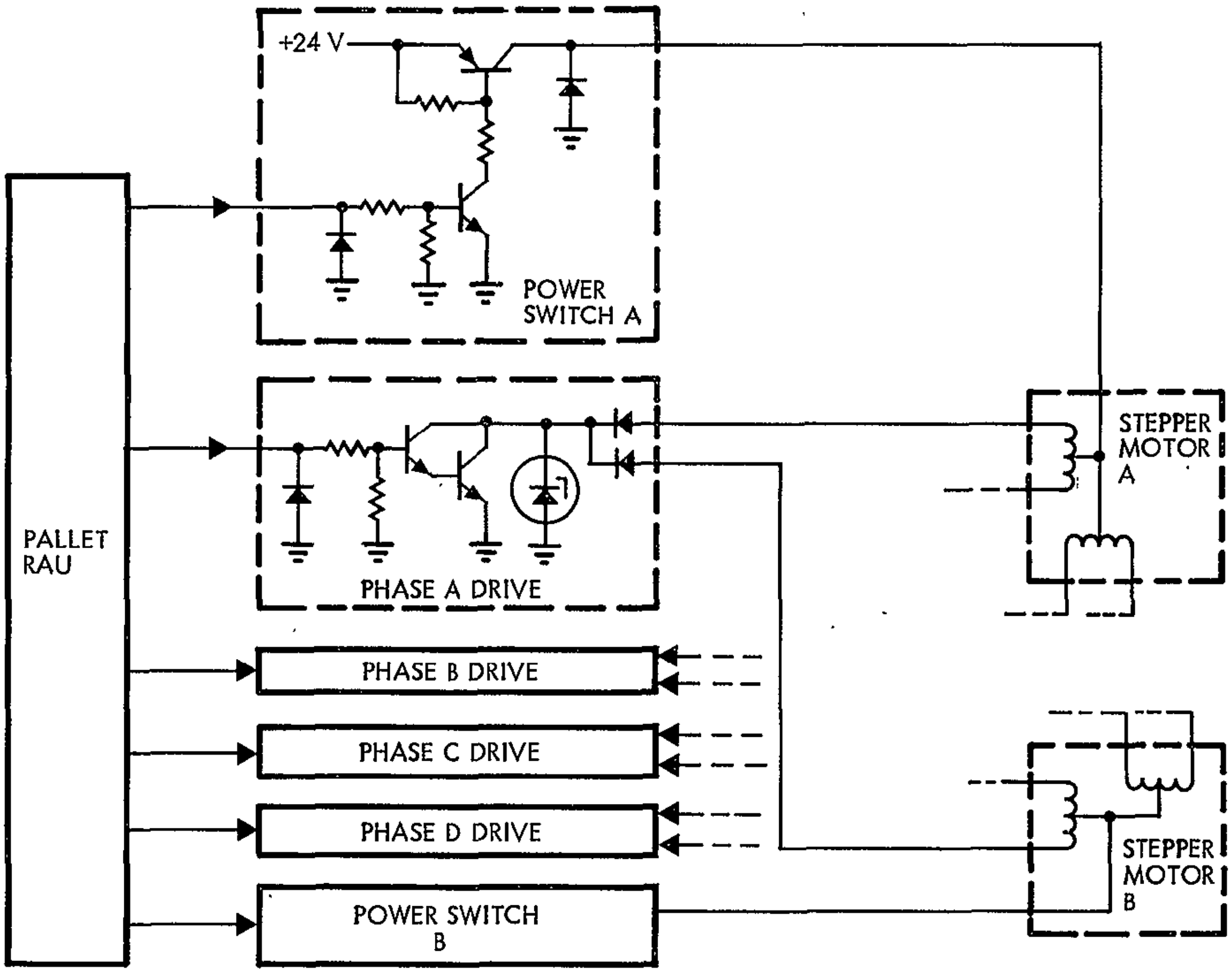


Figure 11. Stepper Motor Driver.

by RAU discrete outputs and generated automatically by commands from a stored computer subroutine. The subroutine, which is called up by the Payload Specialist, will cause the appropriate Faraday cup arm to move through the ion stream in five degree steps. The movements will occur at programmed time intervals (probably 25 seconds). The computer will read the motor positions before and after each movement to insure the proper motor response by sampling the encoders with the RAU discrete inputs.

The solenoids will also be controlled by RAU discrete signals and driven by circuits similar to the motor select power switches. It is planned to have the solenoid discrete commands generated from the Digital Display System keyboard by the Payload Specialist.

3.5.1.5 Instrumentation Power Supply

The power supply uses the standard Spacelab +28 volt bus and converts it to the required low and high voltages used by the instrumentation. The power required by the instrumentation is 9.6 watts average. These numbers do not include any power required by the thruster or thermal control heaters.

The supply consists of an input filter, switching regulator, and an isolated winding DC/DC converter. The input filter is a two-stage LC filter which allows the instrumentation to meet the EMI requirements imposed on Spacelab payloads (i.e., MIL-S-461A). The switching regulator converts the unregulated +28V spacecraft bus to a stable +20V. The switching regulator is turned on/off with a RAU discrete command. The command signal is optically isolated from the Spacelab power bus to maintain signal/power ground isolation.

The DC/DC converter runs from the regulated +20V and produces the required instrumentation voltages. They are ± 12V for the signal processing and low level electronics, +24V for the motors and solenoids, $-V_1$ and $-V_3$ (presently undefined but between -10V and -20V) for the Faraday cup grids G1 and G3 and a four-level commandable (0V, +50V, +100V, +200V) voltage for Faraday cup grid G2. The grid voltage windings are isolated from the signal winding since they are referenced to structure ground at the Faraday cups. The grid voltage G2 is selected by computer command through two RAU discrete command outputs.

3.5.1.6 Instrumentation Data Handling

The data required by the instrumentation is all collected under computer control through the pallet RAU. The data to be acquired includes both analog and discrete measurement points. There are 39 different measurements points requiring various sample rates. The instrumentation measurement list is shown in Table 13.

A typical experiment will last approximately 15 minutes (from the instrumentation viewpoint) with the selected Faraday cup located in 37 different positions. The 37 positions represent five degree steps through the 180 degree arc. Measurements 35 through 39 in Table 13 need only be measured once for each experiment run since they are not position sensitive. Measurements 18 through 34 must be measured once at each of the 37 Faraday cup positions. Measurements 1 through 17 must be measured four times (once for each grid G2 potential, i.e., 0V, +50V, +100V and +200V) at each of the 37 positions. These data samples result in a total data output of 19K-bits per experiment run, or an average data rate during operation of 21-bits per second. This rate does not include data required by the thruster.

This low data rate can best be handled by the 65K-bit/sec engineering link between the Spacelab experiment computer and the Pulse Code Modulated Measurement Unit (PCMMU). This would allow the data to be transmitted down the S-band telemetry link to the payload Operation Control Center.

The data sampling will be controlled by a stored program in the Spacelab experiment computer. At the beginning of an experiment the Payload Specialist will call up the programs from the mass memory. He will then operate the instrumentation solenoids through individual commands from the keyboard. The proper thruster commands will then be given and when the ion thruster is operating, the instrumentation subroutine will be initiated. This subroutine will proceed to run measurement cycles and step the Faraday cups through the ion stream. The data collected will then be formatted and telemetered down. Other experiments can be performed (such as monitoring the thruster start-up or shut-down) by calling up other stored programs from the mass memory.

Table 13. Data Measurement List

No.	Measurement	Type	Data Bits
1	Ion Current Outer Ring Range 1	Analog	8
2	" " " " Range 2		
3	" " " " Range 3		
4	" " " " Range 4		
5	" " " " Range 5		
6	" " " " Range 6		
7	" " " " Range 7		
8	" " " " Range 8		
9	Ion Current Inner Plate Range 1	Analog	8
10	" " " " Range 2		
11	" " " " Range 3		
12	" " " " Range 4		
13	" " " " Range 5		
14	" " " " Range 6		
15	" " " " Range 7		
16	" " " " Range 8		
17	Faraday Cup Grid G2 Voltage	Analog	8
18	Plasma Potential Voltage	Analog	8
19	Motor A Encoder Bit 1	Discrete	1
20	" " " Bit 2		
21	" " " Bit 3		
22	" " " Bit 4		
23	" " " Bit 5		
24	" " " Bit 6		
25	" " " Bit 7		
26	" " " Bit 8		
27	Motor B Encoder Bit 1	Discrete	1
28	" " " Bit 2		
29	" " " Bit 3		
30	" " " Bit 4		
31	" " " Bit 5		
32	" " " Bit 6		
33	" " " Bit 7		
34	" " " Bit 8		
35	Faraday Cup Grid 1 Voltage	Analog	8
36	Faraday Cup Grid 2 Voltage	Analog	8
37	Instrumentaion Electronics Voltage (+12V)	Analog	8
38	" " " (-12V)	Analog	8
39	" " " (+24V)	Analog	8

3.5.1.7 CAMAC Handling of Instrumentation Data

The configuration for the instrumentation electronics presented previously is very dependent upon the Spacelab computer. An alternate configuration could be designed having a computer (or micro computer) as part of the instrumentation. With this system, the majority of the software and control would reside within the on-board computer. The Spacelab computer would then just be used for executive level sequencing and data routing to the PCMMU. The executive level sequencing would consist of simple on/off commands such as "Start Experiment Number One." This configuration would be particularly attractive if the system were interfacing directly into the Orbiter. As presently defined, it will probably be very difficult to get any significant experiment payload software to run on the Orbiter GPC.

The problem with this configuration is the high cost of the instrumentation hardware if it is designed and built with the standard unmanned spacecraft methodology. The majority of the added hardware costs can be minimized if NASA Standard equipment can be used. Although most equipment developed so far at NASA are not applicable to instrumentation, Spacelab Payload Standard Modular Electronics (SPSME) are in the process of being developed out of NASA/MSFC. This SPSME equipment is a space hardened version of commercial CAMAC (Computer Automated Measurement and Control) instrumentation.

CAMAC is a nonproprietary, standard modular instrumentation and interface system for digital data acquisition and control. The CAMAC standard (IEEE-583) affords excellent benefits for this particular experiment. CAMAC is composed of individual plug-in modules that are contained in a rack-mountable structure called a crate. A multiwire bus mounted on the rear of the crate called the CAMAC dataway allows bi-directional communications between the modules and the controlling computer or between the modules themselves. The dataway also supplies regulated voltages to the modules.

The CAMAC crate accepts up to 25 modules; 23 are available for ADC's, scalers, output registers, motor drivers and other modules, and two are used by a crate controller. Modules within a crate are controlled via the crate controller which interfaces to the central processor in use. Except for the computer interface module, all components of the system are computer independent and interchangable. The CAMAC standards also define protocols for digital communications within the system.

NASA interest in CAMAC (Table 14) has been increasing since the early 1970's when payload definition for the Shuttle began. NASA is on the threshold of adopting a Spacelab payload control and data management standard based on CAMAC. MSFC is sponsoring development of this standard as well as actual hardware for Spacelab 1, 2 and 3.

A configuration for the Instrumentation Electronics using the CAMAC standard is shown in Figure 12. The interface to the CDMS is through the RAU serial PCM command/data channels. The dedicated experiment computer is a microprocessor (μP) module. A μP was chosen over a larger computer due to the simple sequencing and data collection requirements of the system. If more complex sequencing or data manipulation evolves, the μP module could be replaced by an NSSC-I or NSSC-II computer.

The modules would perform the same function as the circuits in the baseline circuitry. The analog-to-digital conversion would be done with a self-scanning 12-bit ADC. This higher resolution ADC would reduce the number of range channels required for the ion current measurement to three. With only three ranges required, μP controlled range switching was selected over separate range channels. The programmable High Voltage Power Supply (HVPS) module can supply any voltage between 0 and 400 to a resolution of 1V. This capability plus the inherent growth flexibility of CAMAC allows the system to change or expand to meet almost any new system requirement.

This instrument configuration would produce 5.3K-bits of data for each experiment or an average data rate of 6-bits/sec. The power required would be 26.6 watts peak on 17 watts average.

3.5.2 Ion Thruster Command and Data Management Requirements

The command of the ion thruster is carried out by the Digital Control Unit (DCU). All software required for operating the thruster and its gimbal assembly, the propellant reservoir, the Power Electronics Unit, and the Digita Interface Unit are prestored in the DCU and no memory allocation is required by Orbiter or Spacelab computers. The DCU accepts five command inputs.

Table 14. Summary of CAMAC Activities for Spacelab

NASA/ HEADQUARTERS	Support and coordination of standard space-qualified CAMAC development activities by low cost systems office
NASA/MSFC	Spacelab payload standard modular electronics project to:
	1) Define CAMAC system architecture and Spacelab interfaces (in-house)
	2) Develop CAMAC packaging design standard for Spacelab use (TRW)
	3) Determine CAMAC requirements of Spacelab 1, 2, and 3 payloads and develop module functional specifications (TRW)
	4) Develop space-qualified CAMAC hardware for Spacelab 1, 2, and 3 payloads (contract*)
NASA/GSFC	Suitability for astrophysics payloads (in-house)
	Studies by CAMAC manufacturers of space-qualified modules (BIRA, Kinetic Systems, ORTEC)
	Development of three prototype CAMAC modules for Shuttle use (BIRA, Kinetic Systems, Le Croy)
	Development of a prototype CAMAC crate and power supply for Shuttle use (ELDEC)
	Development of a μP based crate controller (in-house)
	Study of CAMAC utilization for smart pallet (TRW)
	Study of NIM/CAMAC for SIPS-pointed instrument (contract*)
NASA/JSC	Feasibility study of NIM/CAMAC use for Spacelab payloads (Bendix)
	Analysis of cost and utility of NIM/CAMAC for Spacelab payloads (TRW)
	Vibration and thermal tests of CAMAC equipment (in-house)
	NIM/CAMAC demonstration for Spacelab cosmic ray experiment (in-house)

*Contractor not yet selected

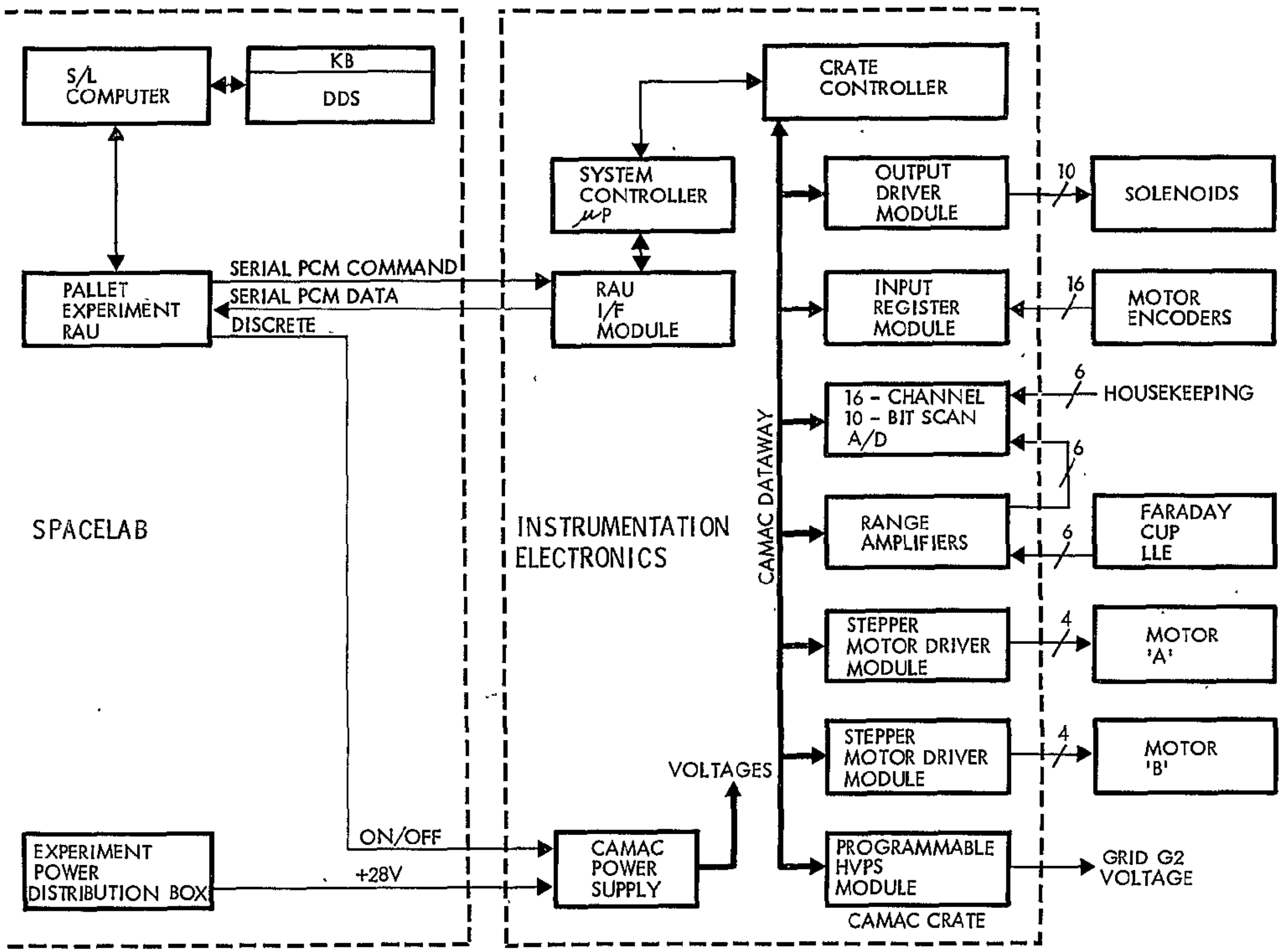


Figure 12. Alternate Electronics Configuration.

The thruster subsystem accepts discrete commands of IDLE, ON, and OFF. GIMBAL θ_1 and GIMBAL θ_2, for actuating the gimbal motors in the θ_1 and θ_2 directions respectively, are serial commands with a minimum 7-bit resolution. For the Shuttle flight test of the ion thruster, these commands would be directed into the thruster DCU by the payload specialist using the appropriat Digital Display Unit and Keyboard.

The data required of the thruster during the measurements includes measurements of thruster currents, voltages, and temperatures. The measured quantities are:

1) J_B, Beam Current,
2) V_1, Net Accelerating Voltage,
3) J_E, Emission Current
4) AV_1, Discharge Voltage,
5) J_A, Accelerator Current,
6) V_{nk}, Neutralizer Keeper Voltage,
7) P_r, Reservoir Pressure,
8) V_A, Accelerator Voltage,
9) J_{nk}, Neutralizer Keeper Current,
10) I_T, Total Thruster System Current,
11) T_{cv}, Cathode Vaporizer Temperature,
12) T_{nv}, Neutralizer Vaporizer Temperature,
13) T_p, Propellant Temperature,
14) J_{ck}, Cathode Keeper Current,
15) V_{ck}, Cathode Keeper Voltage,
16) J_{nh}, Neutralizer Tip Heater Current,
17) J_{ch}, Cathode Tip Heater Current.

The required readout accuracy is consistent with an 8-bit word with ~160 bits per thruster measurement cycle. The thruster measurement cycle need not be carried out for each data point measurement in the RPA/FC's, or in the FPP and, because of its comparatively infrequent requirement for measurement, leads to a condition that the principal data storage needs will be from the diagnostic array with only minor additions from the thruster operational measurements.

3.5.3 Self-Contained Data Acquisition Unit Requirements

From the discussion of Sections 3.5.1 and 3.5.2, it may be concluded that the data storage requirements of the complete ion thruster flight experiment (including both ion thruster data and diagnostic array data) may be satisfied by presently available tape recording units. Such a self-contained (but executively controlled) tape recorder may be considered as another (and potentially valuable) option in the CDMS system. The operational value of such a unit will be determined by the total data acquisition requirements of all elements of the Orbiter payload on a given flight. In the absence of a defined total payload and defined flight operation, access to the Orbiter computers and data storage units cannot be determined. If such access should, however, be determined to be difficult because of competing demands of other payload elements, the ion thruster flight experiment should consider the use of a self-contained data storage unit.

4. FLIGHT EXPERIMENT CONFIGURATION

4.1 FLIGHT EXPERIMENT DESIGN FACTORS

4.1.1 Experiment Mounting Options

Section 2, FLIGHT EXPERIMENT PLANNING FACTORS, has described a series of opportunities and constraints that are relevant to an ion thruster flight test on the Shuttle Orbiter. Two of the principal factors in the thruster flight test planning for a test on the Orbiter are, 1) the recoverability of the payload, and 2) the limited period of flight operation. Both of these factors point toward the design of a multiple-flight test whose serial experiments iterate and expand upon preceding flight experience The concept of a multiple flight test, in turn, requires that the flight test hardware be capable of integration into the Orbiter under at least several possible differing conditions. These conditions include the following possibilities:

1) The Orbiter payload bay contains a Spacelab pallet which has experiment mounting space upon it in a variety of locations, thus permitting the thruster flight test to locate in the pallet position it considers most advantageous for its in-flight execution, or,

2) a Spacelab pallet is present within the bay and has available mounting space, but on a more limited basis than in Item 1, above, or,

3) alternative versions of the Spacelab pallet (such as the proposed NASA/GSFC "smart pallet") have been fabricated and are present in the Orbiter bay and are available for mounting of the thruster flight hardware, or,

4) Spacelab pallets (or other versions of the Spacelab pallet) are present but have no available mounting space for the thruster flight experiment, or,

5) the Orbiter bay has no pallets for a given, specific flight.

A flight experiment design goal that the thruster flight hardware be capable of integration under all of the possible conditions above leads to two requirements. These are:

1) The thruster flight hardware should be capable of multiple location mounting on a Spacelab pallet (or similarly configured pallets), or,

2) the thruster flight hardware should be capable of connection to the Orbiter through another, as yet undefined, interconnect fixture.

To answer the two requirements above, the flight experiment configuration will describe two differing Orbiter interconnect approaches. The first of these approaches will be described in Section 4.2, SPACELAB PALLET MOUNTED FLIGHT EXPERIMENT. This first approach may be considered as a conservative, and comparatively low risk, approach. The second approach to an interconnect with the Orbiter will be described in Section 4.3, CONCEPTUAL "MICROPALLET" MOUNTED FLIGHT EXPERIMENT. This second approach is less conservative than the Spacelab pallet mounted flight experiment and does involve integration with flight hardware fixtures that are only conceptual designs at present. There are, however, reasons (which will be discussed more fully in Section 4.3) for the development of small volume and light weight interconnect fixture to the Orbiter. This light weight interconnect (or "micropallet") would allow the operation of "active" payloads (those requiring Orbiter power, or Orbiter cooling loops, or Orbiter data interconnects or various combinations of all of these interconnects) of small volume and light weight on Orbiter flights which may have available payload space and payload launch weight but which may not, for specific flight configuration reasons, possess Spacelab pallets or similarly configured derivatives of the Spacelab pallet.

4.1.2 Experiment Location Options

Irrespective of the mounting fixture (the Spacelab pallet or the "micropallet") utilized by the thruster flight experiment in its interconnect to the Orbiter, there are design considerations on the experiment location relative to the payload bay. These location possibilities may be divided into two approaches. These are:

1) An out-of-bay location of the flight experiment, or,

2) an in-bay location of the experiment.

In the first experiment location, the closure of the Orbiter payload bay doors during launch and re-entry requires that the mounting fixture be capable of movement from its in-bay launch and re-entry condition to its out-of-bay flight condition. The in-bay location of the experiment require only a single experiment position during the entirety of the flight.

The motion of the thruster flight package from in-bay to out-of-bay for the Item (1) location approach above places several burdens upon the flight hardware that can be avoided by in-bay location. These additional burdens for an out-of-bay deployment are:

1) Additional weight in the overall hardware package because of the required movement of the payload,

2) additional experiment "space" volume to permit the thruster flight experiment to move from one to another point in the Orbiter system,

3) additional system safety considerations in that electrical lines and cooling loop lines must be capable of flexure as the payload moves from one to another point in the Orbiter system,

4) additional failure mode considerations of the now out-of-bay deployed payload (requiring breakaway features for all mechanical, electrical, and thermal cooling loop connections to permit the Orbiter payload bay doors to close during re-entry) in the event of an inability of the thruster flight experiment package to re-deploy after experiment completion, and

5) additional hardware fabrication and hardware integration costs generated by all of the above items.

In view of these additional burdens on an out-of-bay flight experiment, it is worthwhile to review those experiment factors which can be influenced by payload location.

Two principal factors must be considered relative to payload location. These factors are:

1) Possible material deposition from the thruster flight operation upon other Orbiter payload elements, and,

2) location of the thruster plasma beam and the thruster diagnostic package relative to the space plasma.

A discussion of material deposition processes and the post-flight diagnosis of these material transports has been given in Section 3.2.2.3. It has been demonstrated there that, even for prolonged operation ($\sim$ 100 hours) of the thruster during the flight experiment, and even for the location of the deposition plates at positions near the thruster ($\sim$ 30 centimeters), material depositions are of such minute levels ($\sim 10^{17}$ atoms/square centimeter) that sophisticated post-flight analyses will be required to determine the extent and the species of the thruster operation created material build-ups. This low level thruster material transport build-up suggests that impact of the thruster operation on other payload elements is not likely. An absolute judgement here cannot be made in the absence of a clearly defined series of requirements for these other payload elements, and, after a given Orbiter flight has been configured, these material transport processes should be re-examined. For the present study, however, it should be concluded that material transport impact on other payload elements is not likely and should not be a governing factor in the location of the thruster flight experiment package.

The second factor in the location of the thruster experiment package relates to the placement of the thruster plasma beam and the diagnostic package relative to the space plasma. Sections 3.2 and 3.4 have discussed the coupling of the plasma thrust beam to the space plasma and the presence of the ambient space plasma and the "wake" in the space plasma generated by the Orbiter body as these features affect the execution of the various charged particle and electrical equilibration measurements. The discussions there have indicated that it is desirable to have a space plasma wake and that the thruster diagnostic package be capable of being positioned within that wake region. It is desirable, on the other hand, that there be (on occasion) an effective electrical coupling between the ambient space plasma and the plasma thrust beam. However, both the plasma electrical coupling requirement and the plasma wake requirements can be satisfied with an in-bay location of the thruster flight experiment package, and a considered optimal location of the thruster flight payload is <u>within</u> the payload bay and at the <u>edge</u> of the payload bay.

Reviewing the possible experiment costs for out-of-bay deployment as compared to costs for in-bay deployment, and reviewing the material transport considerations and space plasma location and coupling considerations, the recommended flight experiment configuration will be within the payload bay and utilizing a fixed location of the experiment package during the entirety of the Orbiter flight. The embodiments of this in-bay thruster experiment location to be described in Section 4.2 and Section 4.3 will also emphasize an edge-of-bay location as being a desirable flight condition.

4.2 SPACELAB PALLET MOUNTED FLIGHT EXPERIMENT

4.2.1 Ion Thruster Flight Experiment Package

Figure 13 illustrates the Baseline Configuration of the Ion Thruster Flight Experiment Package. For convenience in its inclusion in this report, reduction of the figure size has been employed. In the separately submitted drawings to NASA/LeRC, the experiment package is shown at full scale and is 23.25 inches x 23.25 inches (59.1 cm x 59.1 cm) on the upper face of the container while the remaining dimension of the box is 19.75 inches in depth (50.2 cm).

The ion thruster and the propellant reservoir are mounted on the upper face of an aluminum honeycomb plate located slightly below the mid-plane of the experiment container box. The Digital Interface Unit (DIU), the Power Electronics Unit (PEU), the Thruster Controller and the Regulator and Power Converter (for interconnect of the PEU to the Orbiter) are mounted on the underside of this aluminum plate. The honeycomb mesh dimensions and cell wall thickness (for an allocated plate weight of six pounds) are approximately .0625 inches wall thickness with a 1.0 inch x 1.0 inch cell open area. This plate provides sufficient mechanical strength to support the various thruster elements and also provides sufficient thermal conductivi for the eventual thermal equilibria of the container and its contents for the various Orbiter mounting configurations (to be discussed in the sections to follow). In the present configuration, the plate has substantial remaining unoccupied area to provide for experiment uprating on later flight experiments and/or to provide mounting areas for heaters and thermal cooling connections if subsequent thermal analyses should determine the necessity of these elements.

The upper face of the experiment container has been broached to provide a passage for the ion thruster and its sputter shield. The opening in this upper face is sufficient to permit the full range of thruster gimbaling if it should be determined that the in-flight exercise of the gimbal system is desirable under the Shuttle Flight Test Verification Concept. The upper face of the container also provides a mounting location for the stepper motors which drive the probe mounting arms, and for the probe mounting arm retention fittings. The deposition plate holders are also mounted on this upper face. To avoid possible unwanted charge-up effects which might interfere with measurements of low energy charged particle flow patterns and/or with measurements of thrust beam plasma/space plasma electrical equilibration, it is recommended that the exterior portion of the container upper face should be a conducting material suitably grounded to the container box and (ultimately) to the Orbiter frame. The remaining sides and the bottom of the container may be wrapped with polymeric film second surface mirrors or painted with suitable low emissivity paints (to assist in the thermal equilibration) without any expected adverse charge-up effects on the various flight experiments.

The Retarding Potential Analyzer/Faraday Cups are moved through the thruster beam by the stepper motors and the probe mounting arms. In the configuration illustrated in Figure 13, each of the RPA/FC's move over a total polar angle range of 180°. The Floating Potential Probes (FPP) are mounted on the rear portion of the RPA/FC and have sufficient immersion areas in the plasma thrust beam to determine the thrust beam floating potential (Test T3). The FPP's also have sufficient area to determine the space plasma floating potential (for Test T9) for the appropriate FPP position (see discussion in Section 3.4.5). Figure 13 has illustrated two of these floating potential probes in the flight experiment. Section 3.4 has reviewed the possible use of either one or two FPP's for the electrical equilibration measurements and has concluded that Test T9 can be carried with a single FPP if experiment costs and weight should so dictate. If experiment costs and weight, however, will allow the inclusion of the second FPP, the experiment procedure can be simplified, in principle, and possible important benefits will be derived in a relaxing of requirements on the Orbiter attitude during Test T9.

A remaining element of the probe mounting arm is the retention fitting. The retention fitting secures the probe arm during launch and during re-entry and is opened during the flight to permit the motion of the mounting arm.

The remaining portion of the diagnostic array in this baseline experiment configuration are the deposition plate holders. Two deposition plate holders are illustrated in Figure 13. The first plate holder is mounted to provide a measurement of metal atom release from the forward (exposed) face of the thruster sputter shield. Because of the location of the holder aperture below the plane of the thruster accelerator grid, the deposition plate cannot examine material release from this thruster electrode. The deposition plate here has, however, a clear path to the upper edge regions of the exposed face of the thruster sputter shield and, hence, can determine the removal of material from these areas (ε_{mas}, in the normalized thruster efflux notation). This first deposition plate holder also contains a monitor plate, whose aperture is opened during thruster OFF periods to determine the Orbiter contaminant background at this location on the thruster flight experiment package.

The second deposition plate holder is located on the rear facing side of the thruster sputter shield and has the deposition plates and their apertures so arranged as to view the upper edge of the rear face of the sputter shield. It is not expected that metal atom deposition will be present at this location and the purpose of the plate location at this position is to verify that those "umbra" regions are not subject to material build-ups from the sputter shield. This deposition plate holder also contains a monitor plate for the determination of the Orbiter generated contaminants.

4.2.2 Flight Experiment Mounting Configurations

4.2.2.1 Edge-of-Pallet (Edge-of-Bay) Mounting

Baseline Experiment Configuration/Preferred Mounting Location. Figure 14 illustrates the Baseline Configuration of the Ion Thruster Flight Experiment Package mounted on a Spacelab Pallet. For convenience in its inclusion in this report, a reduction of the figure has been made. Quarter-scale drawings of the experiment package and its mounting arrangement have been submitted separately to NASA/LeRC. The scale in Figure 14 can also be determined from the stated box dimensions in Section 4.2.1.

The location of the ion thruster at the edge of the pallet in the manner shown in Figure 14 is considered to be the preferred location for the ion thruster experiment in the Orbiter payload bay. Experiment mounting on the pallet at this position leads to an edge-of-bay location for the plasma thrust beam which permits an effective space plasma wake generation condition (for the appropriate Orbiter attitude) and also permits a wide

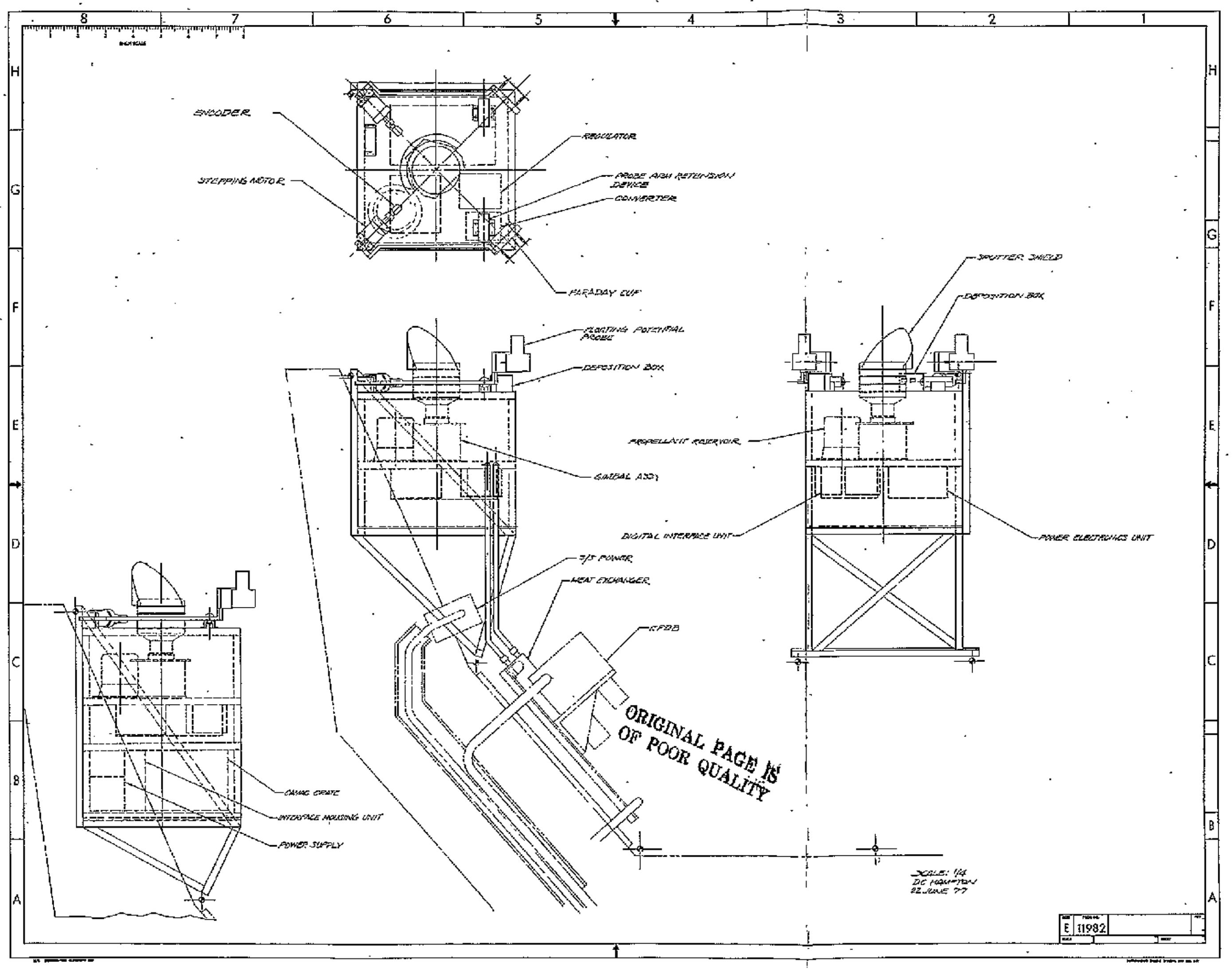


Figure 14. Ion Thruster Flight Experiment Package Mounted on Spacelab Pallet (Edge-of-Bay Mounting).

variation in the plasma beam-to-space plasma coupling condition (see discussion in Section 3.4.5.2 relative to both requirements areas). Edge-of-bay mounting, it should be emphasized, is a preferred condition but is not a required condition, and other experiment mounting locations will be discussed (Section 4.2.2.2) in this report.

Figure 14 (lower left corner) also illustrates an altered configuration of the Baseline Experiment Package in which a CAMAC crate has been added in the CDMS (Command and Data Management System). This CAMAC option will be discussed in Section 4.2.2.3.

Supporting Frame Structural Concept. The ion thruster and its associated components are structurally integrated into a module that is attached at four points to the supporting structure. The module is basically a rectangul box comprised of edge members and sheet metal panels. Individual components are supported by beams and plate elements that carry the loads to the basic module structure which then transmits them to the four support points. This arrangement is conceptually shown in Figure 14. The actual design would incorporate stiffening elements on the panels as required to obtain the required strength and to minimize acoustic resonances.

The support structure is the base upon which the module is mounted and transmits the loads to three attachment points on the pallet. It is configured to provide direct load paths in a straightforward efficient manner. As shown conceptually in Figure 14, this structure is in the nature of a rigid box, triangular in cross section, with the apex extending to the lower pallet attachments and with the module mounted on the opposite flat side. At one end of this box, two truss members extend upward with their apex at the upper pallet attachment. This truss reacts moments about the X axis through the lower attachments. All other forces are carried directly from the module to the lower attachments through the box structure. Diagonal truss members are incorporated on each of three sides to provide both torsional and shear strength.

In the actual layout and detail design, the center line of the members will be positioned to minimize eccentricities and the resulting induced moments. Where such eccentricities are unavoidable, such as at the lower two attachment areas, members will be incorporated and designed to carry the imposed moments. The resulting structure is simple and efficient. Sizing of the members to meet the required design conditions can be easily accomplished. Response to the vibration environment will probably produce the highest loads. These loads can be determined by a structural dynamic model of the structure, including the module.

Power and Data Line Attachment. The attachment of power and data lines for the ion thruster and the diagnostic array have not been illustrated in Figure 14 because of the anticipated pallet payload specific arrangements of these elements. For reference use, the location of the S/S power and the Electrical Power Distribution Box of the Spacelab Pallet have been illustrated in Figure 14 in the view along the Orbiter X axis.

Cooling Loop Line Attachment. The view along the Orbiter X axis in Figure 14 does illustrate the passage of cooling lines from the Ion Thruster Flight Experiment Package to a Heat Exchanger. In this conceptual configuration, the Heat Exchanger has been mounted on the Spacelab Pallet Cold Plate.

The present illustration is for reference use only and does not indicate a positive requirement for an active cooling of the Thruster Flight Experiment Package. In Section 4.2.2.1 (subheading: Experiment/Pallet/Orbiter Thermal Equilibrium Conditions) it will be shown that, for the anticipated heat loads in the 8-cm thruster experiment, heat rejection from the container box may require only the passive approaches of radiation from the box sides and conduction along the box structural members. It should be noted, however, that re-use of the experiment hardware is a desirable condition for the Orbiter flight series and that subsequent thruster tests may occur with other (and, perhaps, larger) ion thrusters whose heat loads into the experiment container cannot be adequately removed by either the radiation or conduction mechanisms. In these latter possible flight conditions, active thermal withdrawal with a cooling loop may be required utilizing either the Spacelab heat exchanger or the Spacelab cold plate (or, possibly, both of these elements).

The estimations of weight (Section 4.2.2.1, subheading: Baseline Experiment and Support Structure Weight) of the Ion Thruster Flight Experiment have not included the weight of either the fluid cooling lines or the fluid pump. This non-inclusion of these elements has been for the following reasons:

1) As described above, the 8-cm thruster experiment may not require active cooling,
2) the estimated weights of presently available fluid pumps for Orbiter service would add significantly to the thruster experiment package weight for even the smallest of the available series of pumps, and
3) the addition of the fluid pump is expected to add significantly to experiment hardware costs and to experiment integration costs.

The approach, thus, to the 8-cm thruster flight experiment thermal planning has been to appeal to passive cooling methods (which appear, at present, to be adequate). If conditions in the thruster experiment should be altered, or if other nearby Orbiter payload temperatures should impact on the thruster flight experiment, or, if more refined thermal analyses should alter the present program findings, then the inclusion of active thermal cooling should be re-examined with the possible use of the fluid pump and the fluid cooling lines.

Baseline Experiment and Support Structure Weight. The weight of the Ion Thruster Flight Experiment Package and the Support Structure illustrated in Figure 14 has been estimated at approximately 100 pounds. This weight estimate includes:

1) All box structural elements,
2) all diagnostic probes, probe arms, stepper motors, and the associated electronics package,
3) the 8-cm thruster package (including the thruster, thruster sputter shield, gimbal mount and propellant reservoir) and the thruster electronics units (DIU, PEU, Regulator and Power Converter) and,
4) the Support Structure connecting the Thruster Experiment Package to the Spacelab pallet.

To save weight in the flight experiment, the propellant reservoir contains a small (2 pound) mercury load. The fluid pumps and fluid cooling lines have not been included in this weight estimate for reasons discussed in the preceding section. Table 15 provides a summary of the estimated weights for the various elements of the Baseline Flight Experiment Package including the support structure.

Table 15. Thruster Flight Experiment Element Weights

Item	Weight (kg)	Weight (lb)
Thruster and Gimbal Assembly	3.4	7.5
Propellant Reservoir (dry)	1.5	3.3
Propellant (10% loading)	0.9	2.0
Power Electronics Unit	6.7	14.8
Digital Interface Unit	2.3	5.0
Thruster Controller	2.3	5.0
Converter (28 v)	0.8	1.8
Boost Regulator	1.4	3.0
Stepper Motor/Encoder (2)	2.4	5.2
Probes, Deposition Plate Enclosures	2.3	5.0
Instrumentation Electronics	2.3	5.0
Thruster Enclosure Support Rods	3.3	7.3
Lower and Upper Box Plates	2.7	6.0
Box Mid-Plane Plate	2.7	6.0
Box Side Panels	4.5	10.0
Box Framing Elements	1.7	3.7
Cabling Interconnects to Pallet	1.8	4.0
TOTAL	43.0	94.6

Experiment/Pallet/Orbiter Thermal Equilibrium Conditions. The conditions of the thermal equilibration of the experiment package with the Spacelab pallet and the Orbiter payload bay will be examined for four cases. These are:

Case 1: The thruster is operating and the Orbiter bay is exposed to sunlight.

Case 2: The thruster is not operating and the Orbiter bay is exposed to sunlight.

Case 3: The thruster is operating and the Orbiter bay is exposed to dark space.

Case 4: The thruster is not operating and the Orbiter bay is exposed to dark space.

Figure 15 illustrates the various heat transport terms and specifies the temperatures for the Orbiter bay utilized in the various calculations.

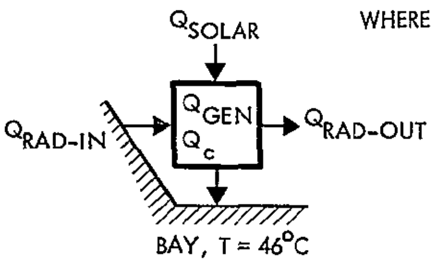
In all of the examined cases, steady state conditions have been assumed. This method of calculation is considered as a conservative approach. For example, in Case 4, if the time span in which the box temperature drops to the non-operating temperature limit (here assumed at $T = -50°C$) exceeds the actual duration of Case 4 (for a low Earth low inclination orbit), then the heater utilized in Case 4 is not required. In order to investigate the various possibilities in Case 4, however, the weights of all elements and heat capacities of all elements must be known, together with the (orbit specific) durations of Case 4 conditions. In the absence of known values on several of these parameters, the conservative approach of a steady state solution has been adopted.

The solutions to the thermal equilibrium have utilized a solar absorptivity, α, of 0.1 and an infrared emissivity, ε, of 0.1 for all surfaces of the thruster experiment package box. The conducted heat, Q_c, will be assigned parametrically. The thruster heat input into the experiment box during thruster operation (including all losses in thruster electronics packages and in the diagnostic array electronics) has been set at 70 watts.

Figure 16 illustrates the experiment box surface temperature as a function of the passive heat conduction rate (Q_c) for Cases 1, 2, and 3 and also lists the temperature limits for the various elements and conditions.

CASE 1

$$Q_{RAD-OUT} = Q_{RAD-IN} + Q_{SOLAR} + Q_{GEN} - Q_c$$



WHERE $Q_{RAD-OUT}$ — HEAT FLUX RADIATED BY BOX INTO SPACE

Q_{RAD-IN} — HEAT FLUX RADIATED BY ORBITER BAY TO BOX

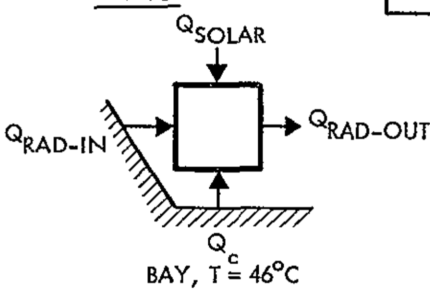
Q_{SOLAR} — SOLAR RADIATION ABSORBED BY BOX

Q_{GEN} — HEAT FLUX GENERATED BY POWER ELECTRONIC UNIT, DIGITAL INTERFACE UNIT AND CONTROLLER IN THE BOX

Q_c — PASSIVE CONDUCTION COOLING RATE

CASE 2

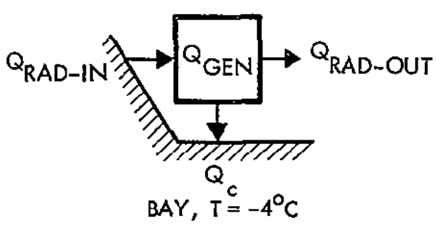
$$Q_{RAD-OUT} = Q_{RAD-IN} + Q_{SOLAR} + Q_c$$



NOTE THAT THE PASSIVE CONDUCTION COOLING REVERSES ITS FLOW DIRECTION.

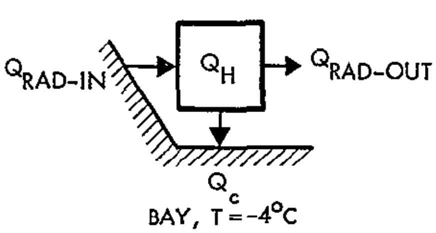
CASE 3

$$Q_{RAD-OUT} = Q_{RAD-IN} + Q_{GEN} - Q_c$$



CASE 4

$$Q_H = Q_{RAD-OUT} + Q_c - Q_{RAD-IN}$$



WHERE Q_H — THE AMOUNT OF HEAT FLUX GENERATED BY A HEATER.

Figure 15. Heat Flow Conditions and Orbiter Bay Temperatures for Thermal Operational Cases 1, 2, 3, and 4.

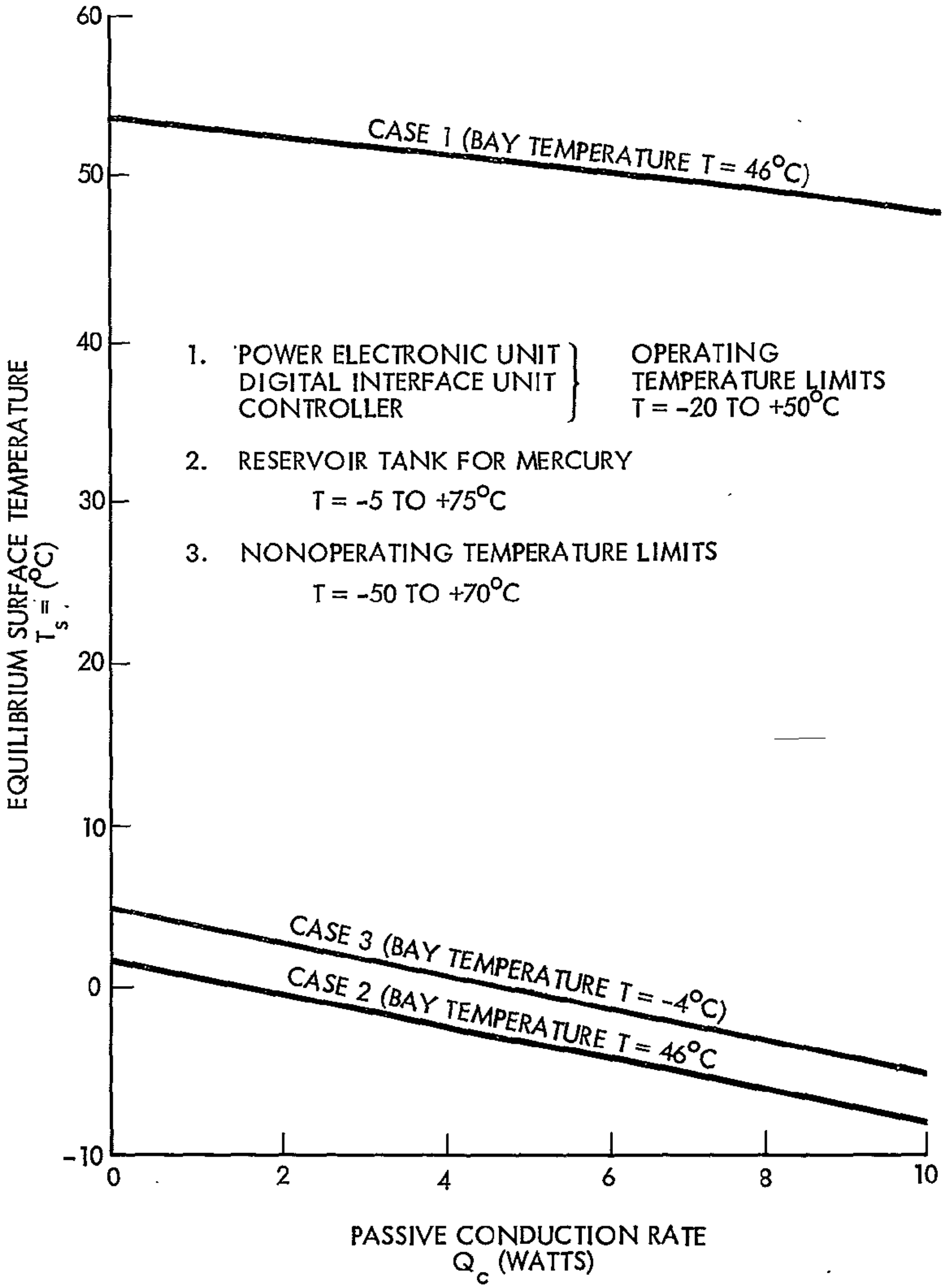


Figure 16. Equilibrium Surface Temperature as a Function of Passive Conduction Cooling Rate.

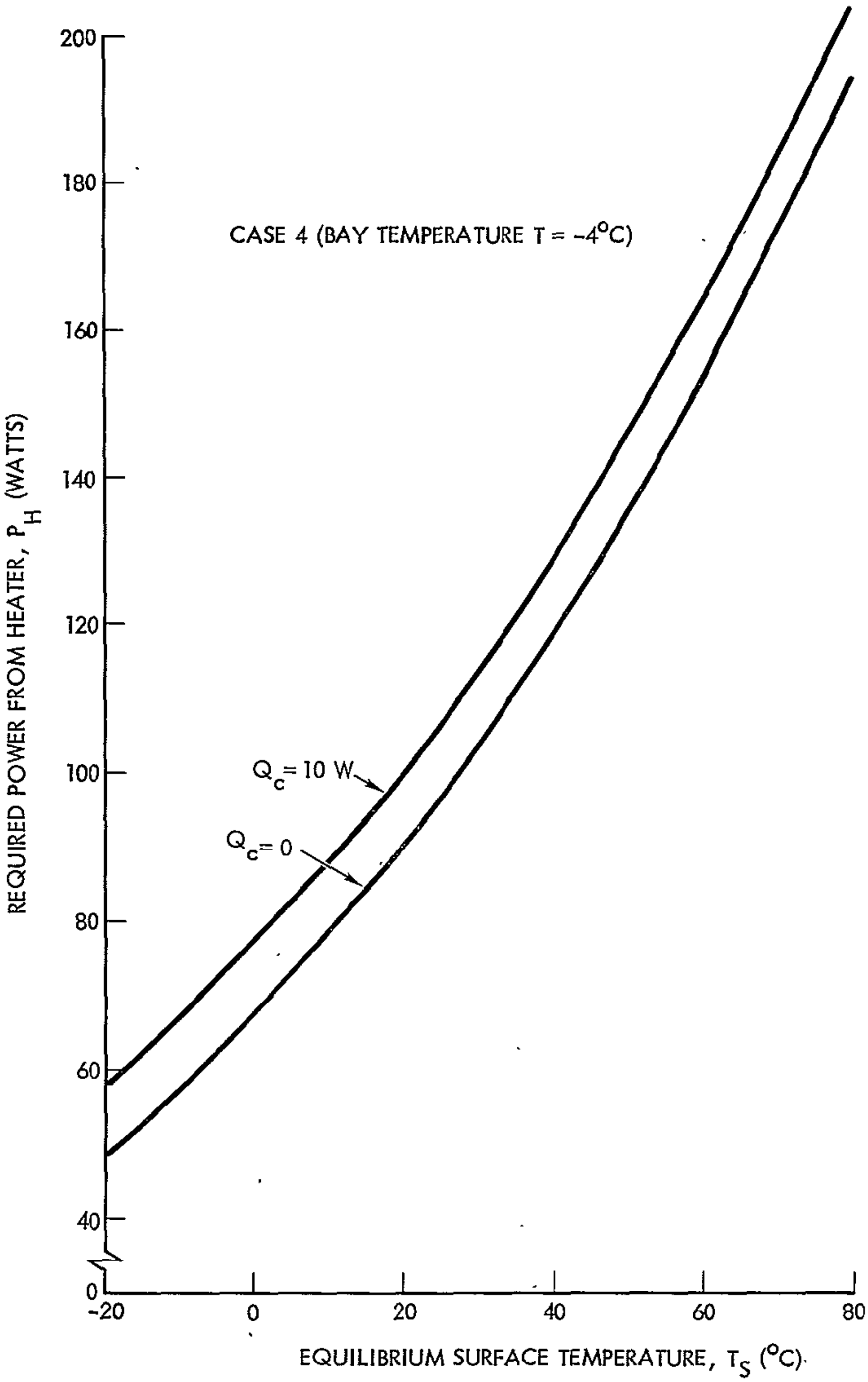


Figure 17. Heater Power as a Function of Surface Temperature.

The surface temperature values illustrated there appear to satisfy, in general, the operating temperature requirements. For small Q_c and Case 1, the maximum temperature on the thruster electronics is exceeded by small temperature increments, but only modest alterations in the ε values along the box surfaces could create the necessary lowering of the surface temperatures. For Case 3 and $Q_c \sim 10$ watts, temperature conditions inside the box might not be sufficiently high for the mercury propellant reservoir. This problem can be alleviated by small amounts of power into the thruster reservoir heater or into an additional box heater to be discussed in Case 4.

Figure 17 illustrates the required power into a box heater to preserve a given box surface temperature for values of $Q_c = 0$ watts and $Q_c = 10$ watts for a Case 4 condition. Only modest values of heater power ($\sim$ 70 watts) are required if this thruster OFF/Orbiter bay exposed to dark space condition is encountered.

While the thermal analysis undertaken here is only a qualitative analysis, the results are sufficiently encouraging that provisions for active cooling of the 8-cm thruster experiments have not been included in the experiment package. It will be assumed that a heater with modest power capabilities (less than 100 watts) will be included in the experiment box. All of these analyses and assumptions should be re-examined after a specific Orbiter flight and the specific Orbiter payload have been assigned and determined. Because of hardware costs and integration costs, it will be particularly desirable to avoid, if possible, the use of active thermal cooling via the fluid loops.

4.2.2.2 Central Pallet (Mid-Bay) Mounting

The edge-of-bay mounting in Figure 14 has been designated as the preferred location. However, if other payload considerations should prevent an edge-of-pallet mount, it is possible for the ion thruster flight experiment package to be mounted in other locations and for a successful flight experiment to be carried out. Figure 18 illustrates such a central pallet (mid-bay) mounting for the Baseline Configuration Experiment Package. The mounting requirements here are, if anything, simpler than those for the edge-of-bay mount. The coupling conditions between the space plasma and the thrust beam plasma are not as widely variable for this mid-bay location

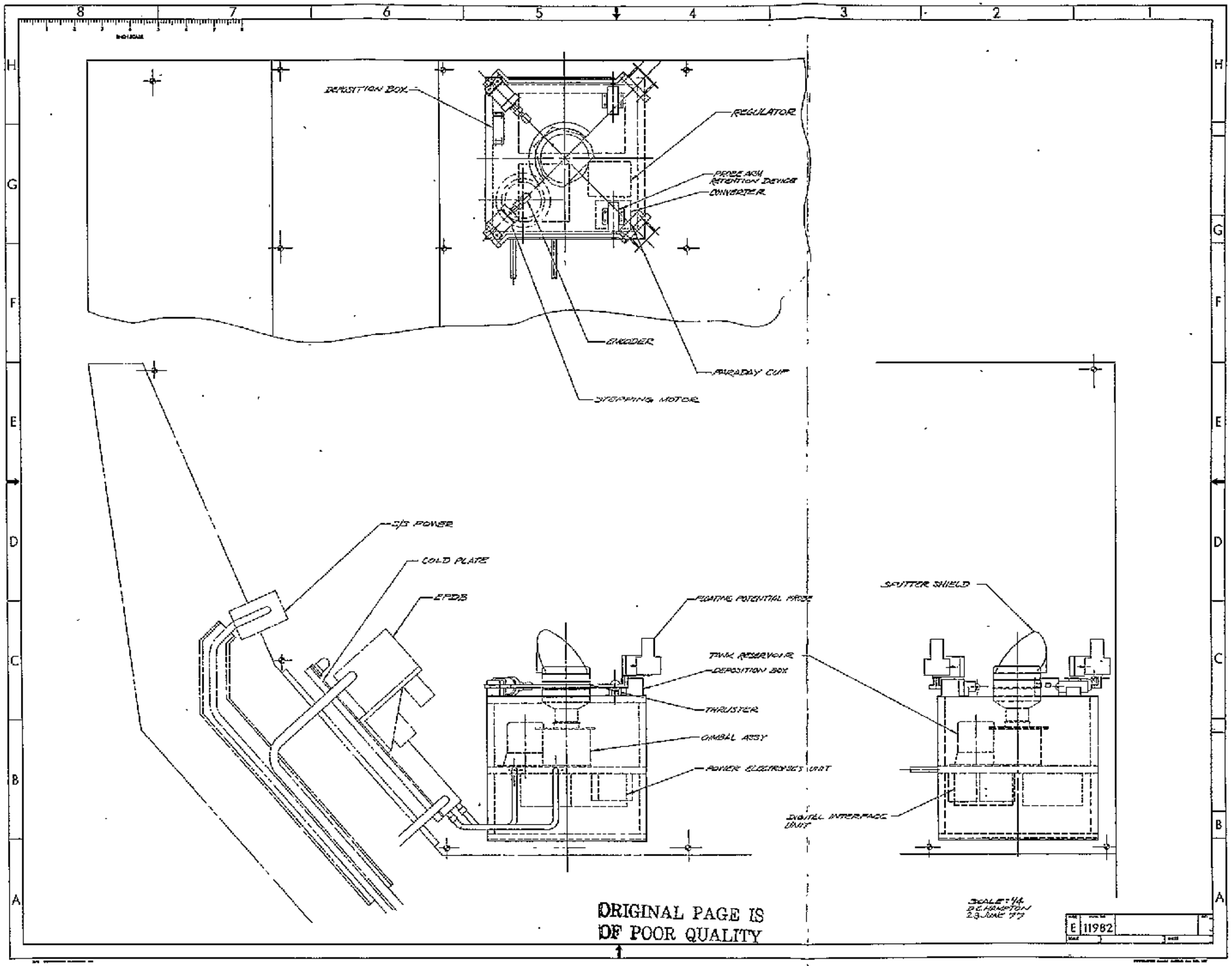


Figure 18. Ion Thruster Flight Experiment Package Mounted on Spacelab Pallet (Mid-Bay Mounting).

as for the edge-of-bay location. Nevertheless, Test T9 can be carried out successfully. A final condition to note for this mid-bay mounting location is that other payload elements should not enter the cone whose axis is the thrust beam axis and whose half angle is $\sim 30°$ because of the thrust ion current density within this region. Thermal analyses have not been carried out for this mid-bay location mounting. The thermal balance conditions for this mounting are, however, expected to be similar to those conditions encountered for the edge-of-bay location. The thermal cooling loops shown in Figure 18 are, as before, for reference use only and it is expected that the 8-cm thruster can operate under all of the thermal cases examined without the use of active cooling by the loops.

4.2.2.3 Add-On CAMAC CDMS Package Configuration

Section 3.5 has discussed possible advantages to the use of the CAMAC system by the thruster flight experiment. Figure 14 (lower left corner) has illustrated an alteration of the Baseline Configuration Experiment Package to include the CAMAC unit. The required modification to the experiment box structure and to the experiment support structure is comparatively minor. Because of added weight and power requirements, however, the structural and thermal analyses of the Baseline Configuration should be re-examined for the CAMAC present case. This analysis cannot be done with accuracy at the present because of remaining definition and development work in the flight-worthy CAMAC system. When completed units are available, it is recommended that a complete cost, power, thermal, and structural analysis be carried out for this add-on system.

4.3 CONCEPTUAL "MICROPALLET" MOUNTED FLIGHT EXPERIMENT

4.3.1 "Passive" and "Active" Orbiter Wall Mounted Payloads

Section 4.1, FLIGHT EXPERIMENT DESIGN FACTORS, has discussed a variety of payload mounting circumstances including those in which a pallet (either of the Spacelab configuration or of the proposed hybrid configuration) cannot be utilized by the ion thruster flight experiment. Under these circumstances the thruster flight experiment (in view of its comparatively small volume and weight requirements) could apply for a mounting on the Orbiter bay wall. There are, however, several difficulties against such a wall mounting arrangement because of the Orbiter services (in the power, data, and perhaps, thermal areas) required by the thruster experiment package.

The accommodation policy of the Orbiter, as it is presently evolving, will permit the attachment of "small" payloads of the order of several cubic feet in volume and of several hundred pounds in weight) to the Orbiter wall for those payloads which are completely passive (that is, not requiring power, data, and thermal cooling services). The number of payloads which can conform to the weight and volume requirements and to the passive operations requirements remains to be determined. A likely condition, however, is that a significantly larger number of payload applicants may appear within the weight and volume requirements but which also may require either data channels or Orbiter power or, perhaps, cooling loops to the Orbiter. Of these active area requirements, the most likely to occur is a need for command and data management channels and the next most likely requirement is in Orbiter power (probably at comparatively low levels). Requirements for thermal cooling of these small payloads may not appear as frequently as either the data or power requirements, because of the possible use of passive cooling techniques for the payloads (for example, see the discussion relative to the ion thruster flight experiment in Section 4.2.2.1).

The total number of payload applicants for wall mounting locations which satisfy the weight and volume requirements but will require some level of "active" remains to be determined. If significant numbers of such applicants should arise, it is possible that additional hardware interconnects between these payloads and the Orbiter will be designed and fabricated to meet these needs. These additional hardware interconnects would provide Orbiter services similar to those available on the present Spacelab pallets but at considerably less weight and volume. Section 4.3.2, which follows will discuss a concept for such a small pallet.

4.3.2 Conceptual "Micropallet" Design

Figure 19 illustrates a design concept for a small pallet (or micropallet) for mounting on the Orbiter wall. The micropallet utilizes (primarily) a mounting to two hard points along the Orbiter longerons (using the more common 59 inch spacing). The two hardpoint mountings

FOLDOUT FRAME 1

FOLDOUT FRAME 2

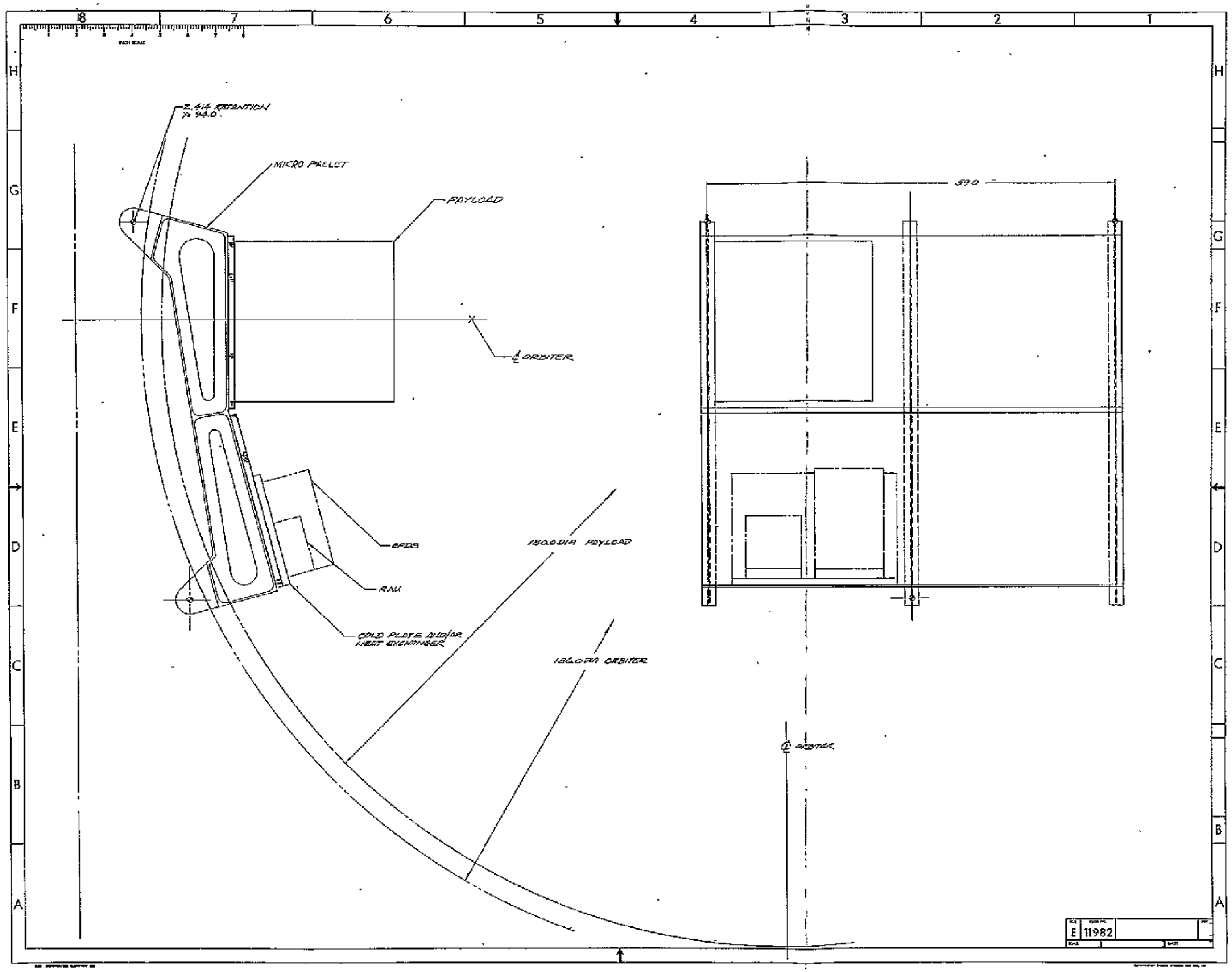


Figure 19. Ion Thruster Flight Experiment Mounted on Conceptual "Micropallet."

react against the major portion of the pallet loads. A third mounting point into the Orbiter fuselage (at the bottom middle of the pallet) provides for only compression or tension loads at that point. The hard point retention fittings and the Orbiter fuselage retention fitting in Figure 18 are not to detail and are described in the drawing there only as to retention fitting location.

The micropallet in Figure 18 is intended to provide a mounting space for several small payloads. The ion thruster flight experiment container is illustrated as one of these possible small payloads. The location of the Remote Acquisition Unit (for command and data links) and the Electrical Power Distribution Box could be either beneath the pallet exterior surface or along the lower bottom edge of the pallet. Because the power demands of the small payloads on this pallet are not expected to be large, the sizing of the electrical power distribution fixtures may permit these elements to be placed within the pallet.

The micropallet may consider the inclusion of a thermal cooling capability in its available services. As noted previously, however, the low power levels of the small payloads mounted on this pallet may not require active cooling.

The micropallet illustrated in Figure 19 is, as has been pointed out, only a conceptual fixture. In view of the possibility of a growing number of applicants for mounting on the Orbiter wall and requiring some levels of active Orbiter servicing, it is recommended that this conceptual fixture be given additional design study.

5. FLIGHT EXPERIMENT PROGRAM PLAN

5.1 MISSION PLANNING FACTORS

The program plan for conducting the 8-cm thruster plume and efflux characterization on seven-day Shuttle Orbiter sortie missions includes preflight, in-flight, and post-landing activities. The plan is consistent with an initial mission as early as first-quarter 1981, with subsequent refurbishment and additional sortie missions to follow as required. Program activities include hardware and software development, experiment integration, ground support, payload specialist support, post flight operations, and data analysis. Ground support requirements have been identified and are presented separately below, as are the requirements for in-flight payload specialist support. Cost estimates have been made for preflight development and testing, and for post flight analysis and data reduction. Other costs chargeable to the experiment have also been identified.

5.2 PREFLIGHT, IN-FLIGHT AND POST LANDING ACTIVITIES

5.2.1 Program Schedule

The overall program schedule for the flight experiment program is shown in Figure 20. The experiment definition was performed under the present contract in the first half of 1977. Following preliminary flight assignment, preflight activities can begin, followed by Shuttle Orbiter integration and flight, and then post-flight activities. About seven months overlap in preflight and Orbiter integration activities is possible because the preflight development has a design verification test phase before entering into proto-flight model fabrication, qualification, and and test. The refurbishable proto-flight model hardware approach has been selected to take advantage of Shuttle's ability to retrieve flight hardware, and thus, to minimize the number of thrusters, power processors, and experimental packages that have to be built to support the program from development through design verification testing, qualification, integration and flight tests. Accordingly, the program plan calls for fabrication, assembly, and test of one development model and one proto-flight model. The former is used for engineering model development through design verification tests. The proto-flight model is employed for qualifi-

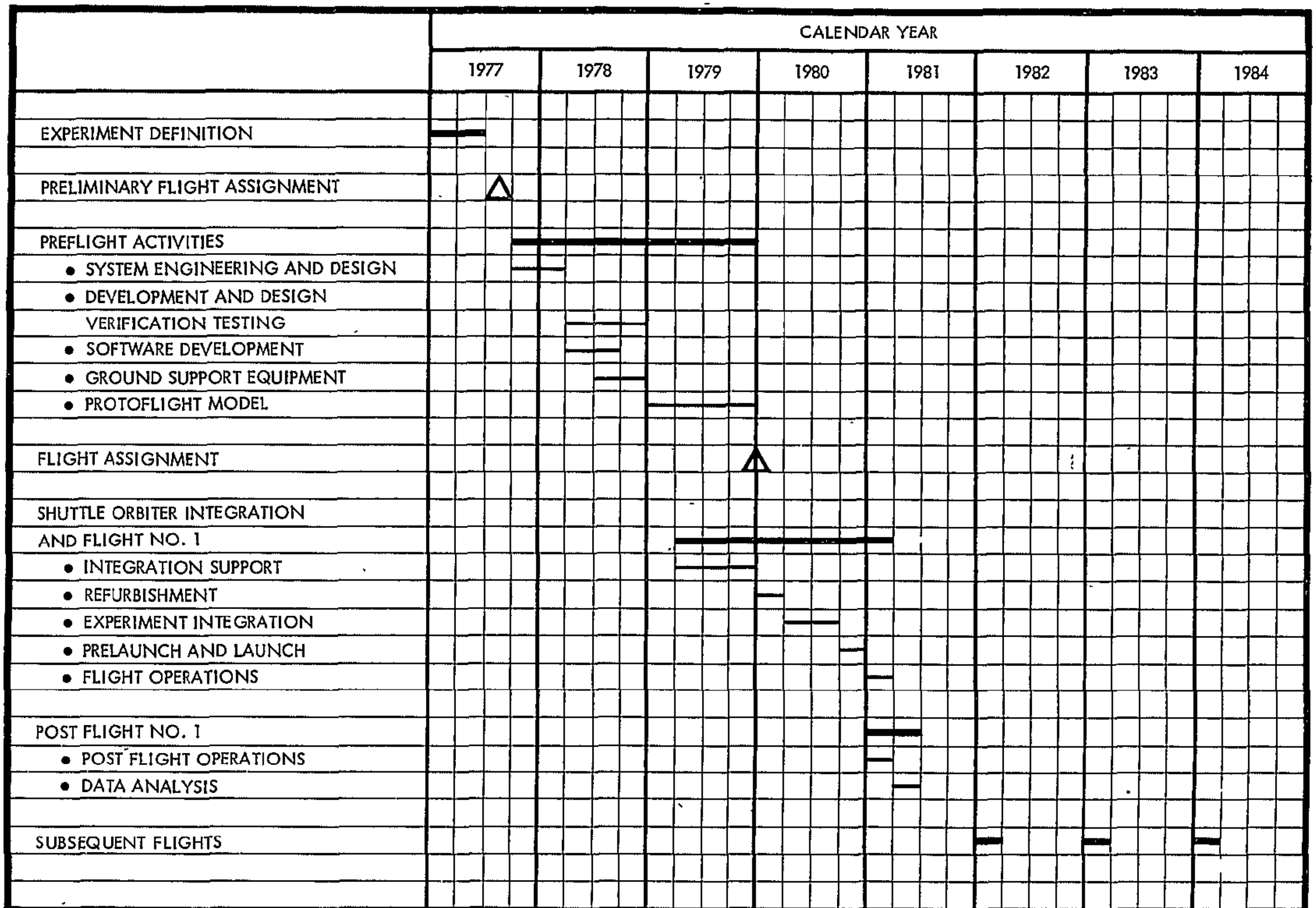


Figure 20. Ion Beam Plume and Efflux Characterization Flight Experiment Program Schedule.

cation, integration, flight, refurbishment, and re-use.

Preflight activities span a period from the third-quarter 1977 through 1979. It begins with the system engineering and design activity that is described in more detail later. Upon completion of design review, development hardware is fabricated and tested, while software development and ground support equipment needs are being implemented. The development hardware undergoes design verification testing prior to initiation of proto-flight model development, which goes through flight qualification testing. After qualification, the hardware is sufficiently well developed and documented to enable a firm flight assignment to be made.

Integration activities may be initiated in anticipation of successful qualification, and are appropriately started upon completion of design verification testing. Integration interface information must be provided to both the pallet integrating contractor and the Shuttle Orbiter integrating contractor before the hardware is actually delivered for integration. After qualification testing, the hardware is refurbished and acceptance tested before delivery to the integrating contractor. The ground support equipmetn is also delivered and is used during integrated system testing. The experiment package is first integrated onto the pallet, which in turn is integrated into the Orbiter bay. The Shuttle is then taken to the launch pad where it is readied for its seven-day sortie mission. It is anticipated that Shuttle Orbiter integration activities, plus a seven-day flight, will take place from the second quarter in 1979 through the first quarter of 1981. Subsequent integration time spans will be shorter, because non-recurring efforts are associated with the initial integration only. Subsequent integration time spans should be about one year in duration.

Post-flight acitvities include equipment checkout during post-flight operations, data reduction and analysis. These activities should be complete by the middle of 1981 for the first flight, and span a six-month period per flight.

5.2.2 Activity Flow Diagram

Figure 21 shows the program activity flow diagram. It serves to show the logical progression from one major block of activity to another during the course of the program. It also shows how flight test data are fed back

FOLDOUT FRAME 1

FOLDOUT FRAME 2

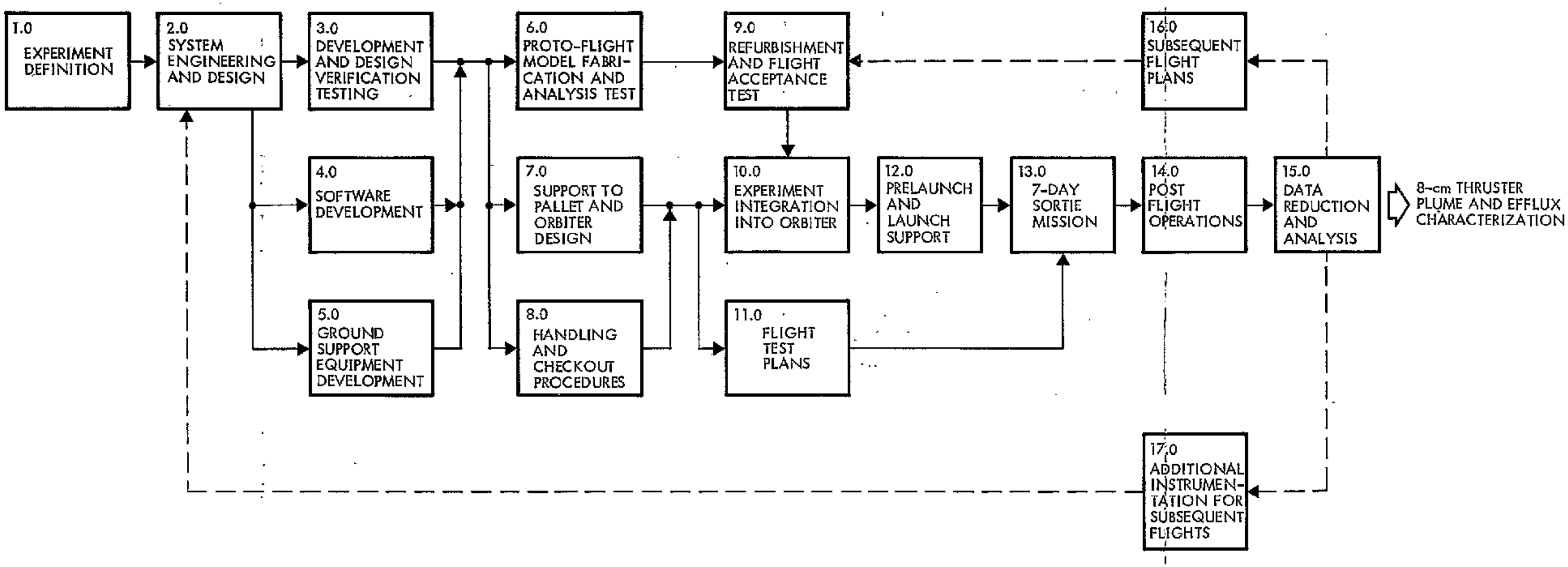


Figure 21. Program Activity Flow Diagram.

into the program in preparation for subsequent flights. The data are used in flight planning and in determining additional instrumentation needs, not provided by the initial experiment package, but deemed desirable for future flights in view of the test data obtained.

The specific activities associated with blocks 2.0 through 15.0 on the flow diagram are discussed individually below. Figure 22 is a detailed schedule for Block 2.0, the System Engineering and Design Task. This figure is included for illustrative purposes to identify the next level of activity definition below that shown on the flow diagram. Task 1.0, Experiment Definition, was performed under the present contract. Tasks 16.0 and 17.0 require feedback from the flight tests to identify specific needs for subsequent flights.

Task 2.0: System Engineering and Design. The objective of this task is to generate the experiment design, system specification, and detailed interface planning documents. It is anticipated that this effort will take six months as shown in Figure 22.

It is first necessary to identify Shuttle Orbiter requirements for the specific flight target, and to prepare a system specification describing the 8-cm thruster plume and efflux experimental package, including the test article, structure, instrumentation, and controls. Also, key inputs to interface control documents have to be prepared to assure that provisions will be made for necessary Shuttle services, including power, refrigerant cooling (if necessary), commands, and data management. When these preparations have been made, a conceptual design review is held with NASA-LeRC to review the system specifications and interface requirements, thereby providing a detailed definition of the work to follow.

After the conceptual design review, design drawings and schematics are prepared for the development model of the experimental package. In parallel with this effort, software requirements are identified and ground support equipment needs are defined in detail. Also, equipment handling and checkout procedures are conceptualized to provide for later planning needs. Equipment tradeoff studies are also performed at this time before hardware commitments have been made. One of these trade studies will investigate the suitability of commercial hardware, such as CAMAC (computer-

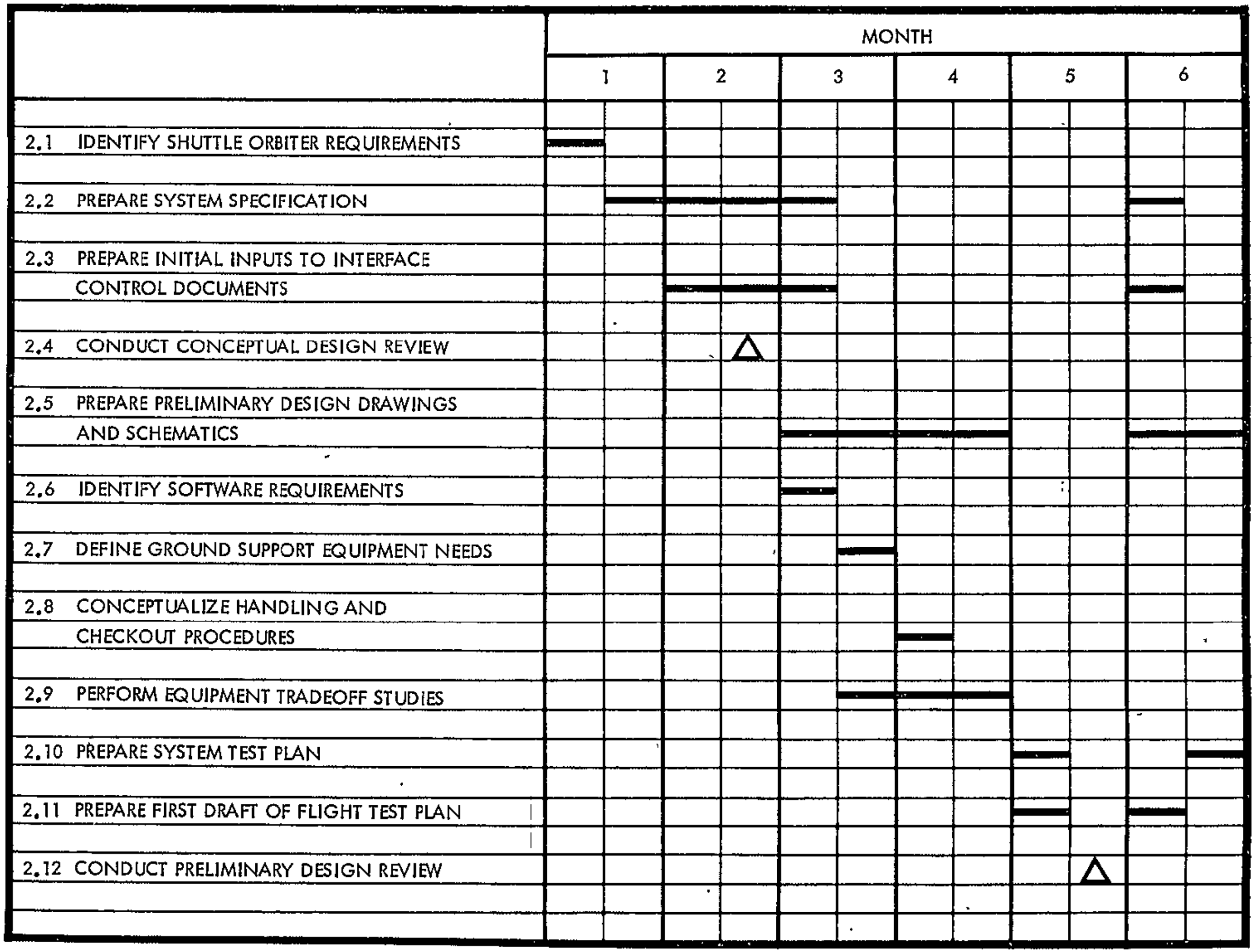


Figure 22. System Engineering and Design Task Schedule.

assisted measurement and control) for application on Shuttle, in order to determine the most cost effective approach. Another tradeoff involves the use of Shuttle avionics versus built-in experiment electronics.

Upon completion of the design effort and trade studies, an overall system test plan can be prepared that defines development, design verification, acceptance, qualification, integrated system, and flight test needs. Also, the first draft of the flight test plan can be prepared. At this time, enough information has been assembled to conduct the preliminary design review, which is conducted to see how the design is intended to meet specification requirements, and how the testing program will verify compliance. After review, appropriate modifications are incorporated into the design drawings, specifications, interface control documents, and test plans to enable further development efforts to proceed.

Task 3.0: Development and Design Verification Testing. In this task, the experimental package hardware for the development model is purchased, fabricated, and assembled in accordance with the preliminary design. Development tests are conducted and the hardware is modified as required to meet performance objectives. At this stage, the development model is subjected to a series of design verification tests, as delineated in the system test plan, to demonstrate that design goals have been achieved.

Task 4.0: Software Development. Algorithms and codes for the Shuttle computer are developed in this task in accordance with the test plans. Also, procurement specifications are prepared for commercial equipment to be purchased for the proto-flight model. The data management system will be selected, and specific items of commercial equipment will be subjected to environmental testing (under Task 3.0) to ensure Shuttle compatibility and to validate the equipment selection.

Task 5.0: Ground Support Equipment Development. The objective of this task is to develop the ground support equipment that will be used with the proto-flight model during acceptance, qualification, and integrated system tests. Specific ground support equipment needs are defined in Task 2.0. These needs will include:

- Ground checkout console to operate and record data from the experimental package.

- Instrument simulator that can be used in conjunction with the ground checkout console during atmospheric testing of the experimental package or during vacuum testing without an active thruster.

The ground support equipment is designed, fabricated, and checked out functionally before being used with the development model. Development tests will be specifically tailored to exercise the ground support equipment in order to validate the equipment for subsequent use with the proto-flight model.

<u>Task 6.0: Proto-Flight Model Fabrication and Qualification Test</u>. An interim design review is held upon completion of design verification testing to verify that performance specifications have been met and that interface requirements fare correctly defined. The preliminary design drawings, together with the development model data, then form the basis for preparation of flight model design drawings. Detailed structural, thermal, EMI (electromagnetic interference) worst-case, FMECA (failure modes, effects and criticality analysis) and performance analyses are made at this time. Also, procurement of long lead items is initiated. Qualification and flight acceptance test plans are prepared.

Upon completion of design and analysis, all the data accumulated to date are assembled for the critical design review which precedes fabrication and assembly of the proto-flight model. Thruster interface control documents as negotiated from task 7.0, are also reviewed, as are the qualification and flight acceptance test plans.

Following the critical design review, proto-flight model fabrication and assembly takes place. At the same time, test facilities are set-up, test instrumentation is calibrated to flight production standards, and detailed test procedures are prepared. A functional check of the proto-flight model is made using the previously validated ground support equipment. Then a Government-furnished 8-cm thruster and power processor is integrated into the experimental package in preparation for qualification testing. These tests of the proto-flight model include:

- Functional checkout
- Environmental

- EMI
- Thermal-vacuum

Any deviation from specification requirements from this time forward requires formal material review board action, including implementation of failure analysis and corrective actions, when required.

During the thermal-vacuum test, a modified flight sequence will be performed in which the diagnostic sensors are deployed through their full range of movement. Qualification tests will be conducted after successful completion of a flight acceptance test sequence, and will obtain data at the extreme specification limits (with suitable margins) where the proto-flight model is expected to operate in service.

Task 7.0: Support to Pallet and Orbiter Design. Following design verificati testing, and in parallel with proto-flight model activities, support must be provided to the pallet integrating contractor and the Orbiter integrating contractor. Pallet and Orbiter layouts, drawings, and interface control documents are reviewed, and interface problems are resolved. The pallet specifications are reviewed for experimental package compliance. Pertinent deviations are identified. Likewise, the Shuttle Orbiter specifications are reviewed for compliance, and pertinent deviations are identified.

The integrating contractors have to be furnished with existing drawings, design analyses, and equipment specifications. Pallet and Orbiter design activities are supported by furnishing additional new or modified drawings, analyses, and specifications as the designs mature. Also, participation is necessary in briefings and design reviews with NASA-LeRC, the pallet integrating contractor, and the Orbiter integrating contractor. Inputs are provided to the interface control documents while they are being negotiated.

Task 8.0: Handling and Checkout Procedures. Handling and checkout procedure for the proto-flight experimental package have to be defined for the integrating contractors. Also, the necessary fixtures, tooling, and handling equipment have to be designed. The areas of handling and checkout include:

- Cleaning
- Shipping

- Receiving Checkout
- Ground Handling
- Installation and Removal
- Electrical Checkout
- Calibration
- Integrated Pallet Tests
- Integrated Orbiter Tests
- Safety
- Storage

Task 9.0: Refurbishment and Flight Acceptance Test. Following qualification test, or for re-use after flight test, the proto-flight model is refurbished to a flight-ready condition. It is then flight acceptance tested and cleaned after test. During this phase of the program, it is desirable to have the payload specialist in attendance so that he can become familiar with the thruster and its diagnostic instrumentation. It is anticipated that a significant portion of the payload specialist training for this experiment can take place during flight acceptance testing.

After test and cleaning, the proto-flight model and its ground support equipment are packaged for shipment to the integrating contractor.

Task 10.0: Experiment Integration Into Orbiter. Because the experimental hardware is neither large nor complex, it is expected that Level IV integration will be performed at Kennedy Space Center (KSC) after the hardware has been delivered. Level IV integration onto the pallet includes experiment installation, connection, and interface verification. A special test will be performed to assure correct operation of the equipment. The ground support equipment will be needed to perform a modified experiment sequence during which the sensors deployment will be checked in addition to noise levels of the sensor electronics.

During Level III integration, the various pallets are mated to each other and the experiment interfaces are checked out.

Level II integration includes connecting the pallet support systems to the experiment and a check of system operation using the actual payload specialist control panels. Computer program and test routines will be checked out and system validations completed. Specifically, EMI testing will be carried out and the mission sequence checked.

Other integration activities include fabrication and validation of the handling and checkout fixtures and tooling used above. Also, handling and checkout procedures have to be validated, as well as those for in-flight control, diagnostics, and operational work-around.

Task 11.0: Flight Test Plans. In addition to the support provided to the integrating contractors, a flight test plan is prepared for implementation during flight operations. Data have to be generated for safety certification of both flight and ground support equipment. Support requirements from launch facilities and personnel are identified. Procedures are defined for in-flight control, diagnostics, and operational work-around. Also, participation in flight readiness reviews assures that adequate flight preparations have been made.

Task 12.0: Prelaunch and Launch Support. Final operations occur during Level I integration when the Spacelab is mated to the Orbiter and a final checkout of system interfaces is completed. In-flight procedure simulations are also performed at this time. On the launch pad, support is provided, as needed, for the system tests conducted prior to lift-off.

Task 13.0: Seven-Day Sortie Mission. Once the Shuttle has attained its prescribed orbit and the experimental period has commenced, the diagnostics will be checked out to assure correct operation. This will entail measuring the static and dynamic voltages applied to the Retarding Potential Analyzer/Faraday Cups and the cold probe and measuring the noise level of the collector outputs. The fixed material deposition monitors will be briefly exposed to assure the retraction of their covers. Each of the movable Faraday cups will be rotated through its full angular range while recording the background current levels produced by the Shuttle/Spacelab and the ambient plasma. Finally, the ion engine will be started and its operational parameters will be checked to assure that the engine is operating in a nominal manner.

After the ion engine has been checked out and the major portion of the Shuttle outgassing has occurred, the monitoring deposition plates are exposed for about an eight-hour period. These plates are only exposed when the ion engine is not operating and they are exposed for time-periods equivalent to those of the acitve plates that are exposed during ion engine operation.

The ion engine should be operational for at least a fifty-hour period over the seven days. This time period should allow meaningful data to be obtained from the deposition and erosion monitors and provide adequate information on ion engine performance.

The first ion engine experiment is nominally programmed to start on the second day. The ion engine is started with the Faraday cups operating in the T1 mode. Once stabilization is achieved, the Faraday cup is electronically programmed to perform the additional T2 and T4 experiments measurements at this angular position. The Faraday cup is then stepped through each angular position recording data on the T1 through T4 modes at each position. The active deposition plates will be exposed for the full period of engine operation (except when Faraday cup probes are in the ion beam) and measurements of the Orbiter electrical equilibration level will be monitored during this period. It is proposed that the ion engine will operate for approximately eight hours each day, and in between these operational periods the monitoring deposition plates will be exposed to the local environment. The following days will essentially be repeats of the first. On one day, the Faraday cups will be maintained in a fixed position and will monitor the ion engine as it is cycled on and off at regular intervals to test its multistart capability. Essentially, day five or six will be a repeat of an earlier day except the ion engine will be operated and the measurements taken in the dark. Day six or five will be a repeat of earlier experiments and the experimental period will end with a final exposure of the deposition monitors.

The final day before re-entry will be occupied with assuring that the various diagnostics and the engine itself are stowed in a manner that protects the equipment and exposured samples during the re-entry phase.

Initial data analysis will be performed while the mission is in progress. If the data indicate that changes in the flight plan are necessa: these changes will be implemented, if possible, in real-time by coordinatio: with the payload specialist and ground support crew.

Task 14.0: Post-Flight Operations. After the Shuttle has been returned to the Eastern Test Range (ETR), the experimental package will be checked out to assure correct system operation and to resolve any in-flight anomali Then the package will be removed from its pallet segment and, in particular the erosion and deposition samples will be removed for subsequent measurement of erosion depth and deposited material accumulation, respectively.

The proto-flight model is then returned for examination and refurbishment for its next flight.

Task 15.0: Data Reduction and Analysis. The data obtained from the seven-day sortie mission are reduced for subsequent use and analysis. Ground test data are also reviewed for comparison. The data are analyzed with respect to:

- Validity of prior ground test results
- 8-cm thruster interactions on operational flight missions
- Subsequent flight test needs.

A final test report is then prepared.

5.3 Ground Support Requirements

Ground support will be required during preflight, in-flight, and post-landing activities. Preflight support will be needed from the pallet integrating contractor and the Orbiter integrating contractor. The pallet integrating contractor conducts Level IV integration, and participates in the subsequent integration activities. The Orbiter integrating contractor conducts Levels I to III integration.

Ground support will be needed from the flight operations crew at ETR during prelaunch check out, in addition to support during the seven-day mission.

Post flight operations will involve the flight operations crews, the

Orbiter integrating contractor, and the pallet integrating contractor in order to check out the experimental package interfaces, assure that it is still functioning properly, and to remove the erosion and deposition samples for post-flight analysis. The package is then removed and returned for refurbishment.

5.4 In-Flight Payload Specialist Support

The payload specialist will be required to command the thruster on and off during flight, to verify that it has achieved steady-state operation within predetermined limits, and to activate the probe mechanisms. His participation in the experiment will be spelled out in detail in the flight test plans, and he will perform his functions in accordance with the scheduled sortie mission time line. The specialist training program should include a session during flight acceptance ground testing of the experimental package, so that he can become familiar with thruster and instrument operation.

5.5 Program Cost Estimates

Estimates have been made, in terms of 1977 dollars, for all the program activities described above. On the basis of these estimates, the total pre-flight development and testing is estimated to cost $400,000, including $43,000 of software development. Non-recurring experiment costs total $431,000, while recurring costs total $83,000. Other non-recurring costs chargeable to the experiment total $176,000, while post flight analysis and data reduction are estimated at $49,000. These cost estimates are all summarized in Table 3-1.

If it is assumed that the initial configuration and flight plans are substantially the same for subsequent flights, then the non-recurring costs per flight are the total of items A, B-1 and C in Table 16, while the recurring costs are the total of items B-2 and D. Thus, the cost per flight is derived as shown in Table 17. For 4 or more flights, the cost of providing and supporting an experimental package for 8-cm ion engine testing in space is less than $400,000 per flight.

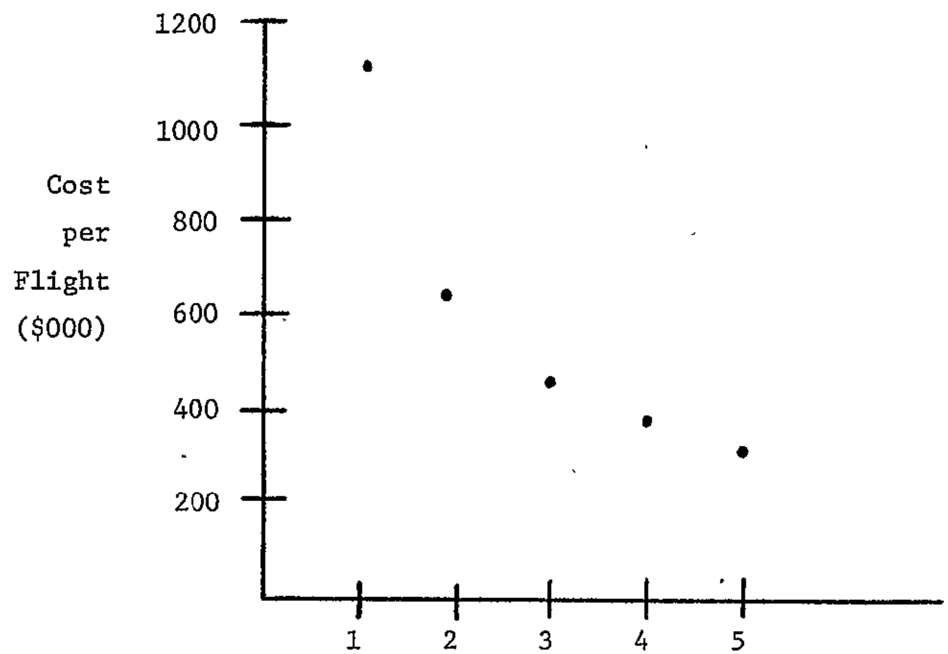
Table 16. Estimated Costs, Ion Beam Plume and Efflux Characterization Experiment

Task	Description	Estimate (Thousands of Dollars)	
	A. PRE-FLIGHT DEVELOPMENT AND TESTING		
2.0	System Engineering and Design	192	
3.0	Development and Design Verification Testing	165	
4.0	Software Development	43	
-	Total Pre-Flight Development and Testing		400
	B. HARDWARE, INTEGRATION, AND FLIGHT OPERATIONS SUPPORT		
	B-1. Non-recurring		
6.0	Proto-Flight Model Fabrication and Qualification Test	411	
10.0*	Experiment Integration Into Orbiter (non-recurring)	20	
-	Total Non-Recurring Experiment Costs		431
	B-2. Recurring		
9.0	Refurbishment and Flight Acceptance Test	28	
10.0*	Experiment Integration Into Orbiter (recurring)	33	
12.0	Prelaunch and Launch Support	8	
13.0	7-Day Sortie Mission Support	6	
14.0	Post-Flight Operations	8	
-	Total Recurring Experiments Costs		83
	C. OTHER COSTS CHARGEABLE TO THE EXPERIMENT		
5.0	Ground Support Equipment Development	54	
7.0	Support to Pallet and Orbiter Design	53	
8.0	Handling and Checkout Procedures	52	
11.0	Flight Test Plans	17	
-	Total Other Non-Recurring Costs		176
	D. POST FLIGHT ANALYSIS AND DATA REDUCTION		
15.0	Data Reduction and Analysis	49	
-	Total Flight Analysis		49

* Task 10.0 totals $53,000; the non-recurring and recurring portions of the task are separated on the table.

Table 17. Estimated Cost Per Experimental Flight

No. of Flights	Total Cost	Cost Per Flight
	(Thousands of Dollars)	
1	1139	1139
2	1271	636
3	1403	468
4	1535	383
5	1667	333



6. SUMMARY

6.1 METHOD OF SUMMARY

This section will summarize, in brief and on a section-by-section basis, the areas examined in this ion thruster flight test study, the principal factors in these examined areas, and the results and conclusions obtained.

6.2 SUMMARY OF THE FLIGHT EXPERIMENT PLANNING FACTORS

Section 2 has examined the FLIGHT EXPERIMENT PLANNING FACTORS with initial emphasis on OVERALL FLIGHT TEST RATIONALE AND GOALS. The goals for the ion thruster flight experiment divide into two groups. The first group, TECHNOLOGY GOALS, is the acquisition of material transport data (for both charged and neutral particles) and spacecraft electrical equilibration data which cannot be obtained in the presence of the material boundaries of conventional (ground based) laboratory testing facilities. The second group of goals comprise the SHUTTLE FLIGHT TEST VERIFICATION CONCEPT, which, utilizing both laboratory and flight experiment data, provides a verification of flight worthiness of ion thrusters for other spacecraft applications. Specific desirable properties of the space testing condition have been identified. These properties are the zero gravity condition, the absence of material boundaries, and the presence of the ambient space plasma. In addition to these general properties of space, the Shuttle Orbiter flight test will possess several, Orbiter specific, opportunities. These opportunities are payload recoverability, payload power, weight, and volume capabilities, manned participation, and Orbiter facilities utilization. The principal constraints in the use of the Shuttle Orbiter for an ion thruster flight test are total operational time and Orbiter orientation. These factors above and the possibility of serial flight experimentation indicate a required flexibility in flight experiment planning. Two forms of flexibility are specifically desirable. These forms are experiment mounting flexibility and experiment operational period flexibility. To approach this operational period flexibility, the planned experiments are designated in three levels. These levels are: Level I experiments, which can be conducted in comparatively brief periods and whose function is to assure that the thruster is operating under nominal conditions, Level II experiments, which can also be conducted in comparatively brief periods and whose function is to examine short term behavior in space which cannot be effectively duplicated in laboratory facilities, and Level III experiments, which also utilize the specific operational conditions of the space environment but which require longer periods of thruster operation. The initial set of flight test experiments will be in all of these indicated levels. In addition, and because of serial flight test possibilities, growth modes of the flight test have been identified. Two specific growth modes of interest are flight experiments involving multiple thrusters (cluster effect studies) which can utilize modular add-ons to the initial single thruster test package, and flight experiments involving substitution of other ion thrusters (perhaps of varying engine diameter) within the original thruster test fixture. In all of these Shuttle Orbiter flight tests, compatibility of the ion thruster with the Orbiter and with other payload elements is required. Such compatibility should be examined on a flight by flight basis. Present indications from ground based thruster testing indicate that it is unlikely that the operation of the ion thruster will adversely affect the operation of the Orbiter or of other payload elements. Finally, Section 2 has described elements of the SHUTTLE FLIGHT TEST VERIFICATION CONCEPT. These elements are the demonstration of total thruster system integrity through spacecraft launch, total system start-up and operational capability under space conditions, total system restart capability through a predetermined set of thruster close-downs and restarts, and thruster operational compatibility with the host spacecraft and with remaining payload elements. The recoverability of the thruster and its post flight examination are a valuable element in this verification of thruster flight readiness. The use of the Shuttle makes possible a simplified, two component, testing approach, utilizing both laboratory measurements and Shuttle Orbiter measurements, which may be able to reduce the total resources required for flight readiness verification compared to the resources required utilizing only a single means (either ground or space) for verification testing.

6.3 SUMMARY OF THE FLIGHT EXPERIMENT DEFINITION

6.3.1 Summary of Ion Beam Plume and Efflux Documentation Tests

Section 3 has examined the FLIGHT EXPERIMENT DEFINITION. The ion beam plume and efflux documentation tests to be defined are a logical continuation and extension of ground based laboratory measurements. These laboratory measurements are contained in two reference works, "Beam Efflux Measurements" (NASA CR-135038, 1 June 1976) and "Ion Engine Auxiliary Propulsion Applications and Integration Study" (to be published Fall of 1977). These reference works have introduced the notation of normalized thruster effluxes and have defined, generally, the permissible normalized efflux levels for specific spacecraft missions. These permissible efflux levels become an implicit portion of the experiment planning for the Shuttle Orbiter flight test.

The THRUSTER TEST DEFINITION PACKAGE (reviewed in Section 3 and presented in a more complete form in Appendix A) has defined a series of ten flight experiments and has described the test objective, the sensor requirements, the instrumentation requirements, the in-flight procedure, the test duration, and the requirements of the Orbiter including possible post-flight activities. These ten tests may also be stated as test groups in ion plume measurements, ion efflux and deposition effects measurements, charged particle drainage measurements, sputter shield effectiveness measurements, thruster internal erosion measurements, and electrical equilibration measurements. Of these ten flight tests a sub-group has been selected for an initial flight experiment. Applicable reasons for the selection or de-selection of a given test have been given. The selected experiments include ion plume measurements for Group I, Group II, and Group IV ions, ion thrust beam neutralization measurements, fixed position deposition plates (analyzed post-flight), thruster internal erosion measurements, and thrust beam/space plasma/Orbiter electrical equilibration measurements. Also included in the initial flight test are Shuttle Flight Test Verification Concept Experiments, including a series of start-restart thruster exercises. All of these flight tests have then been placed into a Flight Schedule for a seven day Shuttle Orbiter sortie. In addition, shortened versions of these flight tests have been examined at the four day and two day mission duration level.

6.3.2 Summary of Flight Experiment Sensor Description

The flight experiment sensors have been examined in detail for those sensors to be employed in the initial flight test. The sensors on this test include Retarding Potential Analyzer/Faraday Cups (2), Floating Potential Probe (1 or 2), and Deposition Plates (2, Fixed Position). In addition, the required sensitivities of all sensors employed in the more general test series have been stated. The sensor configuration for the initial flight test has been defined, including the positioning of the deposition plates and their holders, and the movement of the Retarding Potential Analyzer/Faraday Cups and the Floating Potential Probes.

6.3.3 Summary of the Thruster Internal Erosion Measurements

The thruster internal erosion meausrements have been examined and defined. This test definition includes the location of the internal erosion/deposition samples and possible methods of post-flight analysis of these samples. The study has also discussed conceptual facility effects in ion thruster internal erosion processes and has examined the possible presence of Shuttle Orbiter facility effects and possible resulting requirements for a flight log on Shuttle Orbiter material releases.

6.3.4 Summary of the Ion Thruster System Operation Requirements

The ion thruster system operation requirements have been identified. These requirements have been placed into requirement groups. A first requirement group contains experiment power, experiment energy, and experiment power/time requirements. Stated approximately, the power requirement is 200 watts and the energy requirement (for a 100 hour operation in the flight) is 20 kilowatt hours. Experiment power as a function of time does not appear as a significant requirement area. A second requirement group contains experiment weight, experiment volume, experiment volume location, and experiment volume orientation. For the flight test package designed for an initial flight, experiment weight is approximately 94 pounds (43 kilograms), and experiment volume is approximately 8 cubic feet (0.2 cubic meters). The experiment volume location and orientation are flight experiment configuration dependent and will be described in the review of Section 4. The experiment thermal requirements (requirement group C) are also treated in the review of Section 4.

Experiment propellant for the 100 hour flight test is 0.16 pounds and has been included in the experiment weight list at a level of 2 pounds, for a factor of ten excess loading. Experiment operation requirements for the completion of Level I, Level II, and Level III tests is set in the 50 to 100 hour time range with desired operation at the upper end point. Completion of the Level I and Level II experiments is set at 25 hours. Requirements on Orbiter Daylight/Darkness are not mandatory, but operation under both conditions, if possible, is desirable. Orientation of the Orbiter to create both ambient space plasma wake and ram conditions is required. Orbiter orbit altitude and orbit plane inclination do not appear as significant requirements. The support of the Payload Specialist is required on a limited time basis (with requirements in this area being driven by the degree to which computer-stored experiment programs can be provided). Specific experiment requirements during the Orbiter re-entry period and during post-flight payload handling have been identified and discussed.

6.3.5 Summary of the Command and Data Management Systems Requirements

The Command and Data Management Systems (CDMS) requirements have been examined for several possible experiment configurations. The baseline configuration examined employs experiment specific electronics and interface units which link to the Remote Acquisition Unit (RAU) of the Spacelab system. The experiment specific electronics provides the necessary grid bias voltages for the Retarding Potential Analyzer/Faraday Cups, reads the ion collector plate signals, measures the Floating Potential Probe floating potential, advances the probe mounting arm stepper motors, reads the probe arm angular position encoders, and closes and opens the deposition plate holder shutters. These various operations and measurements require 1 Serial PCM Command Channel, 23 Analog and 16 Discrete Flexible Inputs, and 22 Discrete Commands. These requirements would utilize only a fraction (approximately one third) of the present RAU capability. In addition to the experiment specific requirements, there are requirements for the command of the ion thruster and the measurement of the various thruster currents, voltages, and temperature. The study has utilized an ion thruster DCU which accepts and requires only five commands for all phases of the thruster operation. A total measurement of thruster operation parameters involves approximately 17 measurements.

The measurement cycle for the thruster is, however, more infrequent than for the experiment sensors and the major data storage and transmission will be for the ion beam plume current and floating potential measurements. Even here, the total data storage (approximately 20 kilobits per experiment run) is comparatively minor by flight experiment standards. The measurements of the ion currents, on the other hand, do require additional effort in the electronics package design because of the wide dynamic range (six orders of magnitude) as the current measuring probe moves from the thrust beam axis to positions near 90° from this axis. This dynamic range consideration has been approached through the use of multiple amplifiers (8) on each ion collector element which provide current measurement accuracies of five percent over the total dynamic range of six orders of magnitude. Accurate measurements (one percent) of the floating potentials are achieved with only a single voltage measurement unit because of the reduced range in variation of this parameter.

In addition to the baseline CDMS configuration described above, the study also examined alternative configurations including the hybrid pallet, use of the Orbiter General Purpose Computer, and the CAMAC handling of instrumentation data. The hybrid pallet interfaces with a NASA standard spacecraft computer, NSSC-1, which differs from the Spacelab computer. Because the NSSC-1 does not have a high order language compiler, a significant impact would exist for software originally written for experiment operation on the Spacelab computer. In the case of the use of the Orbiter General Purpose Computer, a majority of the software generated in the (Spacelab oriented) CDMS baseline would be transferable. The thruster electronics interface to the GPC will, however, be altered because of differing methods in the transmission of the serial command link. The CAMAC option for the handling of instrumentation data, a final element in the CDMS study and in this review of Section 3, FLIGHT EXPERIMENT DEFINITION, has a series of attractive features which should be reviewed and updated as the development and requalification of CAMAC units from laboratory systems to flight systems proceeds.

6.4 SUMMARY OF THE FLIGHT EXPERIMENT CONFIGURATION

6.4.1 Summary of the Flight Experiment Design Factors

Section 4 has examined the FLIGHT EXPERIMENT CONFIGURATION. In determining the flight experiment configurations to be studied in greater detail, and because of the possibilities of a serial flight experiment, the study emphasis was directed initially at FLIGHT EXPERIMENT DESIGN FACTORS, including Experiment Mounting Options and Experiment Location Options. Several possible conditions for experiment mounting were considered. The conditions were that a Spacelab pallet was present and had available upon it a variety of mounting locations, that a Spacelab pallet was present but had mounting positions on a more limited basis, that alternative versions of the Spacelab pallet were present in the bay and had available mounting space, and, finally, that mounting space on pallets was not available (either because of prior mounting commitments on pallets that were present or because of a total absence of pallets). The design goal of this study was that the thruster test package could adapt to all of the conditions outlined above. An immediate consequence of this goal is the required definition of some other, as yet unspecified, interconnect element between the ion thruster flight test package and the Shuttle Orbiter. That interconnect element will be described later in further detail and will be designated as an Orbiter "Micropallet."

The experiment mounting locations include two possibilities. These are the mounting of the experiment in the Orbiter bay with in-bay operation, or in-bay mounting with a subsequent out-of-bay deployment for the flight operation. A variety of factors relative to out-of-bay deployment have been identified including additional weight and volume requirements, additional systems safety and failure mode considerations, and additional hardware fabrication and integration costs, all of which counter indicate the out-of-bay deployment option. Other factors involved in the experiment location involve material deposition on other payload elements as a result of thruster operation and the relative placements of the thruster plasma beam and the space plasma. Because material deposition levels are at extremely small levels, and because the plasma wake and plasma ram conditions are generated more easily and distinctly with an in-bay mounting location, and because the costs of an in-bay mounting appear to be significantly less than an out-of-bay deployed experiment, the in-bay mounting and operating option has been chosen. An optimum configuration, in the context of such an in-bay location is within the payload bay and at the edge of the bay. Experiment value is retained at high levels, however, for other locations within the payload bay, and both edge-of-bay and mid-bay mounting configurations have been detailed in the study.

6.4.2 Summary of the Spacelab Pallet Mounted Flight Experiment

The baseline configuration of the flight experiment will be an ion thruster flight experiment package in an edge-of-pallet (also edge-of-bay) mounting on a Spacelab pallet. The ion thruster flight test package is a rectangular box containing the ion thruster and its several components (DCU, Digital Interface Unit, Power Electronics Unit, and the Regulator and Power Converter) and the diagnostic array and its elements (Retarding Potential Analyzers/Faraday Cups, Floating Potential Probes, Deposition Plate Holders, Probe Mounting Arms, Stepper Motors and Encoders, and Diagnostic Array Electronics). The mounting plates within the test package are capable of either heating or cooling if thermal equilibration conditions demand such actions. The preliminary thermal analysis indicates, however, that cooling of the package will not be required although heating may be required for sunlight absent/thruster OFF conditions. In the baseline configuration (edge-of-pallet), both thermal and electrical connections to the pallet are illustrated, although, as described, the most likely condition is that only power and data lines will require a connection to the Spacelab pallet.

In addition to the baseline configuration the study has carried out a description of a mid-bay (mid-pallet) mounting of the experiment and, for the edge-of-bay configuration, an alteration of the basic thruster flight test package to include a CAMAC unit.

6.4.3 Summary of the Conceptual "Micropallet" Mounted Flight Experiment

Section 4 has also described a "Micropallet" for an edge-of-bay mounting of comparatively small payloads, such as the ion thruster experiment, which are also active payloads in that they require either power or data or thermal cooling loops from the Orbiter. The number of such small and active payloads that will apply for Orbiter flights and for which conventional pallet mounting will not be possible, for whatever reasons, remains to be determined. If the number of such payloads should become significant, consideration should then be given to some form of a smaller interconnect between the Orbiter and these payloads, such as the conceptual micropallet.

6.5 SUMMARY OF THE FLIGHT EXPERIMENT PROGRAM PLAN

A final section, Section 5, of this flight experiment study has examined the FLIGHT EXPERIMENT PROGRAM PLAN. The program schedule for the 8-cm thruster flight test has utilized this present study as an experiment definition stage and has continued this program activity through a launch and flight test on the Shuttle Orbiter in the mid-1981 time frame with recurring (yearly) flight tests thereafter with the developed flight hardware. A refurbishable proto-flight model hardware approach has been selected to take advantage of the payload recoverability feature of the Shuttle Orbiter. This approach minimizes the number of thrusters, power processors, and experimental packages that have to be built to support the program from development through design verification testing, qualification, integration, and flight tests. Accordingly, the program plan calls for the fabrication, assembly, and test of one development model and one proto-flight model. The development model is used for engineering model development through design verification tests. The proto-flight model is employed for qualification, integration, flight, refurbishment, and re-use.

Pre-flight activities in the program plan span a period from third quarter 1977 through 1979. The system engineering and design activity in this pre-flight period is described in task-by-task detail in the body of the report. The system engineering and design phase is the second element of a seventeen element total activity flow which extends from the experiment definition phase (element 1, and the present study), through the flight test period and post-flight data analysis to the flight experiment iteration for serially continuing flight experiments. The activity during each of these program elements is described in the study text.

A final element in the flight experiment study and in the program plan is the estimation of program costs. These estimates have been made, in terms of 1977 dollars, for all of the program activities described above, and have been further separated into recurring and non-recurring cost items. On the basis of these estimates, the total pre-flight development and testing is estimated to cost $400,000 including $43,000 of software development. Non-recurring experiment costs total $431,000, while recurring costs total $83,000. Other non-recurring costs chargeable to the experiment total $176,000, while post-flight analysis and data reduction are estimated at $49,000.

If it is assumed that the initial configuration and flight plans are substantially the same for subsequent flights, the non-recurring costs are $1,007,000 while the recurring costs are $132,000 per flight. The cost per flight for a single flight is $1,139,000, and, for increasing numbers of flights diminishes to less than $400,000 per flight for four or more flight tests.

In summary for the total flight test study, a flight experiment for an 8-cm ion thruster has been defined. This flight test connects logically with previous ground based measurements and carries out measurements which cannot be obtained in conventional testing facilities. The ion thruster flight test is also consistent with and makes appropriate use of the various opportunities and constraints of the Shuttle Orbiter. Because of payload recoverability, the Shuttle Orbiter ion thruster flight test provides a cost effective method for the serial testing of thrusters in space.

7. EXECUTIVE SUMMARY

The Ion Beam Plume and Efflux Characterization Flight Experiment Study has examined the definition and configuration of a flight experiment and flight experiment package for a Shuttle-borne flight test of an 8-cm mercury ion thruster. The principal emphasis in the flight experiment is to obtain charged particle and neutral particle material transport data that cannot be obtained in conventional ground based laboratory testing facilities. The principal features of the space environment to be utilized here are the absence of material boundaries and the presence of the ambient space plasma. A second objective of the Shuttle thruster flight test is the Shuttle flight test verification concept through which, by the use of both ground and space testing of ion thrusters, the flight worthiness of these ion thrusters, for other spacecraft applications, may be demonstrated.

A principal advantage of a Shuttle flight test is the recoverability of the payload. This recoverability has important implications in terms of the use of the payload hardware for serial flight testing and in terms of reduced per flight testing costs. A series of growth mode flight experiments for the thruster flight tests has been described, including the modular build-up of multi-thruster tests to examine "cluster" effects in the combined plumes and including the substitution of other thrusters in the flight test package.

A principal limitation in the Shuttle flight test of an ion thruster is in the test duration. The range of available test time lies between 10^2 and 10^3 hours with the latter figure as a possibility only after the development of the prolonged mission (40 day) Shuttle Orbiter capability. Because of these test time limitations, endurance testing of ion thrusters will continue to be a ground based laboratory test.

The flight experiment definition for the ion thruster has initially defined a broadly ranging series of flight experiments and flight test sensors. From this larger test series and sensor list, an initial flight test configuration has been selected with measurements in charged particle material transport, condensible neutral material transport, thruster internal erosion, ion beam neutralization, and ion thrust beam/space plasma electrical equilibration. These measurement areas may all be examined for a seven day Shuttle sortie mission and for available test time in the 50-100 hour period.

The ion thruster flight test package is a comparatively small (.6 meters x .6 meters x .6 meters), comparatively light ($\sim$ 45 kilograms) experiment capable of many different mounting options on a Spacelab pallet. A preferred mounting location is a fixed mount (non-deployed) at the edge of the Spacelab pallet (edge-of-bay for the Orbiter). If required, a mid-bay mount is also satisfactory for experiment purposes. The flight test package is presently configured to accept the 8-cm mercury ion thruster. If desired, and for later flight test, this flight test package can be modified to accept the 30-cm ion thruster. Electrical power lines and experiment command lines from the Shuttle Orbiter are required. Data transmission lines may also be used, or, if required, data storage can be internal to the thruster flight experiment. Thermal cooling loops from the Orbiter may or may not be required, depending upon specific flight configurations, orientations, and other payload elements.

The Ion Beam Plume and Efflux Characterization Flight Experiment Study has also defined an overall program plan for the flight experiment and has estimated the costs for the flight test package (exclusive of the thruster system costs). The estimated costs for a single flight test are slightly in excess of $1,000,000. For a serial flight test, and including both recurring and non-recurring tests, the per flight test costs are somewhat less than $400,000 at the five flight test point, leading to a cost effective space flight test procedure and utilizing the principal feature in Space Shuttle flight testing of payload recoverability.

APPENDIX A

TEST DEFINITION PACKAGE

Table 1. Ion Beam Plume and Efflux Characterization Flight Test Titles and Designations

Test Designation	Test Title
T1	Group I (Thrust) Ion Plume Measurements
T1A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T1B	Transverse Plane Measurements
T2	Group II (High Energy High Angle) Ion Plume Measurements
T2A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T2B	Transverse Plane Measurements
T3	Ion Thrust Beam Neutralization Measurements
T3A	Thrust Beam Plasma Potential Measurements
T3B	Thrust Beam Neutralizing Electron Temperature Measurements
T4	Group IV (Charge Exchange) Ion Plume Measurements
T4A	Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements
T4B	Transverse Plane Measurements
T5	Condensible Neutral Efflux Measurements
T5A	Deposition Plate Measurements
T5A1	Fixed Position Deposition Plates
T5A1a	In-Flight Deposition Effects Measurements
T5A1b	Post-Flight Deposition Effects Measurements
T5A2	Movable Position Deposition Plates
T5A2a	In-Flight Deposition Effects Measurements
T5A2b	Post-Flight Deposition Effects Measurements
T5B	Quartz Crystal Microbalance Measurements
T6	Non-Condensible Neutral Effects Measurements
T6A	Ionization Gauge Measurements
T6A1	Fixed Position Ionization Gauge
T6A2	Movable Position Ionization Gauge
T6B	Residual Gas Analyzer
T7	Thruster Internal Erosion Measurements
T8	Charged Particle Drainage to Electrically Biased Surface Measurements
T9	Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurements
T10	Multiply Charged Ion Production Measurements

Table 2. Ion Beam Plume and Efflux Flight Test Measurement Areas and Associated Flight Test Designations

MEASUREMENT AREA	OVERALL TEST DESIGNATION									
	T1	T2	T3	T4	T5	T6	T7	T8	T9	T10
PLUME MEASUREMENTS	●	●	●	●						●
EFFLUX AND DEPOSITION EFFECTS MEASUREMENTS					●	●				
CHARGED PARTICLE DRAINAGE MEASUREMENTS				●				●		
SPUTTER SHIELD EFFECTIVENESS MEASUREMENTS	●	●		●	●	●				
THRUSTER INTERNAL EROSION MEASUREMENTS							●			
ELECTRICAL EQUILIBRATION MEASUREMENTS			●						●	

Table 3. Test Titles and Objectives

T1: Group I (Thrust) Ion Plume Measurements

Objective: The objective of the Group I Ion Plume Measurements is the determination of Hg^+ thrust ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T2: Group II (High Energy High Angle) Ion Plume Measurements

Objective: The objective of the Group II Ion Plume Measurements is the determination of high energy high angle Hg^+ ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T3: Ion Thrust Beam Neutralization Measurements

Objective: The objective of the Ion Thrust Beam Neutralization Measurements is the determination of the thrust beam plasma potential and the thrust beam plasma neutralizing electron temperature as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" plane.

T4: Group IV (Charge Exchange) Ion Plume Measurements

Objective: The objective of the Group IV Plume Measurements is the determination of low energy, high angle, charge exchange Hg^+ ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

T5: Condensible Neutral Efflux Measurements

Objective: The objective of the Condensible Neutral Efflux Measurements is the determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster and the surface properties effects of such effluxes at selected locations in the thruster system coordinate space.

T6: Non-Condensible Neutral Efflux Measurements

Objective: The objective of the Non-Condensible Neutral Efflux Measurements is a determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster at selected locations in the thruster system coordinate space.

T7: Thruster Internal Erosion Measurements

Objective: The objective of the Thruster Internal Erosion Measurement is the determination of the rate of material loss at specified internal locations of the ion thruster during in-flight operation.

T8: Charged Particle Drainage to Electrically Biased Surfaces Measurements

Objective: The objective of the Charged Particle Drainage to Electrically Biased Surfaces Measurement is the determination of the charged particle flow from the ion thruster exhaust plume to specified surfaces at varying levels of electrical bias and under varying degrees of insulating encapsulation.

T9: Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurements

Objective: The objective of the Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurement is the determination of the Orbiter electrical potential relative to the potential of the space plasma for varying orientations between the thrust beam vector, $\vec{v}_+$, and the Earth's magnetic field vector, $\vec{B}_e$, and for varying configurations of the ionosperic plasma wake (created by Orbiter motion through the space plasma) and the ion thruster beam plasma.

T10: Multiply-Charged Ion Production Measurements

Objective: The objective of the Multiply Charged Ion Production Measurements is to determine the ratio of doubly charged thrust ions to singly charged thrust ions (Hg^{++}/Hg^+) as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" plane.

Table 4. Selected and De-Selected Tests for an Initial Orbiter Ion Thruster Flight Experiment

Selected Experiments	T1, T2, T3, T4, (Both A, B) T5A1b T7 T9
De-Selected Experiments	**Reasons for De-Selection**
T4A1a, T5A2a, T5A2b	Experiment cost and complexity Possible Orbiter contaminants Possible cross-contaminant generation
T5B, T6A1, T6A2, T6B	Experiment cost and complexity Possible Orbiter contaminants
T8	Competing effects of space plasma
T10	Experiment costs and complexity Laboratory measurements may be sufficient

Table 5. Ion Beam Plume and Efflux Characterization Flight Test Sensors and Sensor Designation

Sensor	Designation
Retarding Potential Analyzer/Faraday Cup (Neutralizer/Sputter-Shield Mid-Line/Thrust Beam Axis Plane)	RPA/FC1
Retarding Potential Analyzer/Faraday Cup (Transverse Plane)	RPA/FC2
Floating (Cold) Potential Probe (Transverse Plane)	FPP
Deposition Plate (Fixed Position)	DPF
Deposition Plate (Movable)	DPM
Quartz Crystal Microbalance	QCM
Ionization Gauge (Fixed Position)	IGF
Ionization Gauge (Movable)	IGM
Residual Gas Analyzer	RGA
In-Flight Optical Properties Analyzer	IOA
Internal Erosion Sample	IES
Electrically Biasable Surface	EBS
Orbiter Floating Potential Probe	OFP
Multiply-Charged Ion Probe	MIP

Table 6. Ion Beam Plume and Efflux Charcterization Flight Test Sensors and Associated Test Designations and Required Test Fixtures for Ion Beam Plume and Efflux Characterization Flight Test, Test Fixture Designation, and Associated In-Flight Test Designations

Sensor	Test Designation
RPA/FC1	T1A, T2A, T4A, T8, T9
RPA/FC2	T1B, T2B, T3B, T4B, T8, T9
FPP	T3A, T3B, T8, T9
DPF	T5A1a, T5A1b
DPM	T5A2a, T5A2b
QCM	T5B
IGF	T6A1
IGM	T6A2
RGA	T6B
IOA	T5A1a, T5A2a
IES	T7
EBS	T8
OFP	T9
MIP	T10

Required Test Fixtures

Fixture	Designation	Test
Thruster Test Fixture	TTF	T1, T2, T3, T4, T5, T6, T7, T9, T10
Remote Test Fixture	RTF	T8

Table 7. Ion Thruster Flight Test Sensor and Sensitivity Requirements

Sensor	Sensitivity Requirement
Retarding Potential Analyzer/ Faraday Cup (RPA/FC)	Lower End Current Density Sensitivity, 10^{-8} A/cm^2, for Ion Group I, II, IV
Floating Potential Probe (FPP)	1 Volt in Plasma Floating Potential
Deposition Plates (DPF and DPM)	Lower End Deposition Level Sensitivity, $5(10)^{16}$ particles/cm^2
Quartz Crystal Microbalance	Lower End Deposition Level Sensitivity, 10^{15} particles/cm^2
Ionization Gauge (IGF and IGM)	Lower End Flux Density Sensitivity, $3(10)^{11}$ particles/cm^2/sec
Residual Gas Analyzer (RGA)	Lower End Flux Density Sensitivity, $3(10)^{11}$ particles/AMU/cm^2/sec
In-Flight Optical Properties Analyzer (IOA)	Lower End Deposition Level Sensitivity, 10^{16} particles/cm^2
Internal Erosion Sample (IES)	Lower End Erosion Level Sensitivity, 100 Angstroms
Electrically Biasable Surface (EBS)	Requirements are Mission Specific
Orbiter Floating Potential Probe (OFP)	1 Volt in Plasma Floating Potential
Multiply-Charged Ion Probe (MIP)	Lower End Current Density Sensitivity, 10^{-6} A/cm^2 for Hg^+, 10^{-8} A/cm^2 for Hg^{++}

Title: Group I (Thrust) Ion Plume Measurements

Overall Test Designation: T1

Sub-Test Designations:

T1A: Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements

T1B: Transverse Plane Measurements

Objective: The objective of the Group I Ion Plume Measurements is the determination of Hg^+ thrust ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

Sensor Requirements: The required sensors are two movable position, multigridded, Faraday Cup/Retarding Potential Analyzers (RPA/FC). The Faraday Cups are located so that one cup moves in each of the two designated planes of measurement (See T1A and T1B above). Other sensor requirements are:

Angular Position Measurements Accuracy	.01 radians
Angular Range	$-90° < \theta < 90°$
Maximum Cup Entrance Angular Width	.1 radians
Lower End Current Density Sensitivity	10^{-8} A/cm^2
Grids	Three, Separately Biasable
Collectors	2
Forward and Rear Grid Potential Variation Range	± 20 volts
Middle Grid Potential Variation Range	± 200 volts
Minimum Radial Separation Distance	30 cm

Instrumentation Requirements

Element	Number	Measurement
Angular Position Drive	2	Probe Angular Position
Forward Grid Potential Supply	2	Forward Grid Potential
Middle Grid Potential Supply	2	Middle Grid Potential
Rear Grid Potential Supply	2	Rear Grid Potential
Ion Current Collector	4	Ion Current

Procedure: The procedure for Group I ion current density measurement in each of the measurement planes is:

Step	Action
1)	Set Probe Position at Designated Angle
2)	Read Probe Position
3)	Set Forward and Rear Grid Potentials at Designated Levels
4)	Read Forward and Rear Grid Voltages
5)	Set Middle Grid Potential at Designated Lower Level
6)	Read Middle Grid Lower Level Potential
7)	Read Ion Collector Current (Groups I, II and IV)
8)	Set Middle Grid Potential at Designated Upper Level
9)	Read Ion Collector Current (Groups I, II)
10)	Advance Probe Position and Recycle Procedure to Step 5) Above

The procedure for Group I and Group II ion current separation is based upon angular range and rate of current dropoff for increasing θ. Group IV ion current is identified as the variance in Collector Signal between Steps 7) and 9) in the procedure above.

Duration

15 minutes per 180° Probe Sweep

30 minutes per complete (2-plane) scan

Orbiter Requirements: Very high θ values (low current density levels) should be examined for possible photoemission current signals (sunlight/dark signal variance).

Post-Flight Activities: None

Title: Group II (High Energy High Angle) Ion Plume Measurements

Overall Test Designation: T2

Sub-Test Designations:

T2A: Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements

T2B: Transverse Plane Measurements

Objective: The objective of the Group II Ion Plume Measurements is the determination of high energy high angle Hg^{+} ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

Sensor Requirements: The required sensors are two movable position, multi-gridded, Faraday Cup/Retarding Potential Analyzers (RPA/FC). The Faraday Cups are located so that one cup moves in each of the two designated planes of measurement (See T2A and T2B above). Other sensor requirements are:

Angular Position Measurement Accuracy	.01 radians
Angular Range	$-90° < \theta < 90°$
Maximum Cup Entrance Angular Width	.1 radians
Lower End Current Density Sensitivity	10^{-8} A/cm^2
Grids	Three, Separately Biasable
Collectors	2
Forward and Rear Grid Potential Variation Range	± 20 volts
Middle Grid Potential Variation Range	± 200 volts
Minimum Radial Separation Distance	30 cm

Instrumentation Requirements

Element	Number	Measurement
Angular Position Drive	2	Probe Angular Position
Forward Grid Potential Supply	2	Forward Grid Potential
Middle Grid Potential Supply	2	Middle Grid Potential
Rear Grid Potential Supply	2	Rear Grid Potential
Ion Current Collector	4	Ion Current

Procedure: The procedure for Group II ion current density measurement in each of the measurement plans is:

Step	Action
1)	Set Probe Position at Designated Angle
2)	Read Probe Position
3)	Set Forward and Rear Grid Potentials at Designated Levels
4)	Read Forward and Rear Grid Voltages
5)	Set Middle Grid Potential at Designated Lower Level
6)	Read Middle Grid Lower Level Potential
7)	Read Ion Collector Current (Groups I, II and IV)
8)	Set Middle Grid Potential at Designated Upper Level
9)	Read Ion Collector Current (Groups I, II)
10)	Advance Probe Position and Recycle Procedure to Step 5) Above

The procedure for Group I and Group II ion current separation is based upon angular range and rate of current dropoff for increasing θ. Group IV ion current is identified as the variance in Collector Signal between Steps 7) and 9) in the procedure above.

Duration

15 minutes per 180° Probe Sweep

30 minutes per complete (2-plane) scan

Orbiter Requirements: Very high θ values (low current density levels) should be examined for possible photoemission current signals (sunlight/dark signal variance).

Post-Flight Activities: None

Title: Ion Thrust Beam Neutralization Measurements

Overall Test Designation: T3

Sub-Test Designations:

T3A: Thrust Beam Plasma Potential Measurements

T3B: Thrust Beam Neutralizing Electron Temperature Measurements

Objective: The objective of the Ion Thrust Beam Neutralization Measurements is the determination of the thrust beam plasma potential and the thrust beam plasma neutralizing electron temperature as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" plane.

Sensor Requirements: The required sensor is a movable position (floating) cold probe (FPP) so located that its passage through the thrust beam plasma follows the same path as the Faraday Cup Retarding Potential Analyzer in the designated "Transverse" plane. Other sensor requirements are:

Angular Position Measurement Accuracy	.01 radians
Angular Range	$-90° < \theta < 90°$
Maximum Probe Surface Angular Width	.1 radians
Cold Probe Minimum Floating Impedance	10 megohms
Minimum Probe Surface Area Exposed to Ion Flow	10 cm^2
Minimum Radial Separation Distance	30 cm

Instrumentation Requirements

Element	Number	Measurement
Angular Position Drive	1	Probe Angular Position
Floating Cold Probe	1	Probe Floating Potential

Procedure: The procedure for the thrust beam neutralization measurements is:

Step	Action
1)	Set Probe Position at Designated Angle
2)	Read Probe Position
3)	Read Probe Floating Potential
4)	Advance Probe Position and Recycled Procesure to Step 1) Above

The measurement above directly determines the thrust beam plasma floating potential (T3A). The determination of thrust beam neutralization electron temperature (T3B) follows from the electrostatic "barometric" equations, using the plot of probe floating potential (V_{FP}) against the logarithm of the thrust beam plasma density as determined by thrust ion current density measurements (T1B) and known thrust ion velocity.

Duration:

15 minutes per 180° Probe Sweep

Orbiter Requirements: High θ values should be examined for possible ambient ionospheric plasma effects (using Orbiter orientation/ionospheric plasma wake signal variance). The Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Test (T9) also requires use of the Floating Cold Probe and will require Orbiter orientation to create and to eliminate ionospheric plasma wakes and to create varying $\vec{v}_+$ and $\vec{B}_e$ configurations (where $\vec{v}_+$ is the thrust ion velocity vector and $\vec{B}_e$ is the Earth's magnetic field vector).

Post-Flight Activities: None

Title: Group IV (Charge Exchange) Ion Plume Measurements

Overall Test Designation: T4

Sub-Test Designations:

T4A: Neutralizer/Sputter Shield Mid-Line/Thrust Beam Axis Plane Measurements

T4B: Transverse Plane Measurement

Objective: The objective of the Group IV Plume Measurements is the determinations of low energy, high angle, charge exchange Hg^{+} ion current density as a function of polar angle, θ, at fixed radial distance, R, in each of two mutually orthogonal planes.

Sensor Requirements: The required sensors are two movable position, multigridded, Faraday Cup/Retarding Potential Analyzers (RPA/FC). The Faraday Cups are located so that one cup moves in each of the two designated planes of measurement (See T1A and T1B above). Other sensor requirements are:

Angular Position Measurement Accuracy	.01 radians
Angular Range	$-90° < \theta < 90°$
Maximum Cup Entrance Angular Width	.1 radians
Lower End Current Density Sensitivity	10^{-8} A/cm^2
Grids	Three, Separately Biasable
Collectors	2
Forward and Rear Grid Potential Variation Range	± 20 volts
Middle Grid Potential Variation Range	± 200 volts
Minimum Radial Separation Distance	30 cm

Instrumentation Requirements

Element	Number	Measurement
Angular Position Drive	2	Probe Angular Position
Forward Grid Potential Supply	2	Forward Grid Potential
Middle Grid Potential Supply	2	Middle Grid Potential
Rear Grid Potential Supply	2	Rear Grid Potential
Ion Current Collector	4	Ion Current

Procedure: The procedure for Group IV ion current density measurements in each of the measurement planes is:

Step	Action
1)	Set Prove Position at Designated Angle
2)	Read Probe Position
3)	Set Forward and Rear Grid Potentials at Designated Levels
4)	Read Forward and Rear Grid Voltages
5)	Set Middle Grid Potential at Designated Lower Level
6)	Read Middle Grid Lower Level Potential
7)	Read Ion Collector Current (Groups I, II and IV)
8)	Set Middle Grid Potential at Designated Upper Level
9)	Read Ion Collector Current (Groups I, II)
10)	Advance Probe Position and Recycle Procedure to Step 5) Above

The procedure for Group I and Group II ion current separation is based upon angular range and rate of current dropoff for increasing θ. Group IV ion current is identified as the variance in Collector Signal between Steps 7) and 9) in the procedure above.

Duration

15 minutes per 180° Probe Sweep

30 minutes per complete (2-plane) scan

Orbiter Requirements: Very high θ values (low current density levels) should be examined for possible photoemission current signals (sunlight/dark signal variance) and for possible ambient ionospheric plasma signals (Orbiter orientation/ionospheric plasma wake signal variance).

Post-Flight Activities: None

Title: Condensible Neutral Efflux Measurements

Overall Test Designation: T5

Sub-Test Designations:

T5A: Deposition Plate Measurements

T5A1: Fixed Position Deposition Plates

T5A1a: In-Flight Deposition Effects Measurements

T5A1b: Post-Flight Deposition Effects Measurements

T5A2: Movable Position Deposition Plates

T5A2a: In-Flight Deposition Effects Measurements

T5A2b: Post-Flight Deposition Effects Measurements

T5B: Quartz Crystal Microbalance Measurements

Objective: The objective of the Condensible Neutral Efflux Measurements is the determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster and the surface properties effects of such effluxes at selected locations in the thruster system coordinate space.

Sensor Requirements: For Sub-Test T5A the required sensors are deposition plates at either fixed positions or at movable positions (DPF or DPM). For Sub-Tests T5A1a and T5A2a, the in-flight analysis of deposition effects requires the In-Flight Optical Properties Analyzer (IOA). For the deposition plates (either fixed or movable) the requirements are:

Angular Location Measurement Accuracy	.01 radians
Maximum Deposition Plate Angular Width	.1 radians
Lower End Deposition Level Sensitivity	5×10^{16} particles per square centimeter
Shielding Enclosures Required	Yes
Minimum Radial Separation Distance	30 cm

For In-Flight Analysis of deposition effects on surfaces the sensor requirement on the In-Flight Optical Properties Analysis is:

Lower End Deposition Level Sensitivity	5×10^{16} particles per square centimeter

For Sub-Test 5B, the required sensor is a Quartz Crystal Microbalance (QCM). The lower end deposition level sensitivity requirement of the QCM is 10^{15} particles per square centimeter.

Instrumentation Requirements: For the movable deposition plates the instrumentation requirements are:

Element	Number	Measurement
Angular Position Drive	2	Deposition Plate Angular Position

For In-Flight Analyses of deposition effects the instrumentation requirements are:

Element	Number	Measurement
Reference Light Source and Internal Detector	1	Reference Light Intensity
Transmitted Light Detector	1	Transmitted Light Intensity
Reflected Light Detector	1	Reflected Light Intensity

For the Quartz Crystal Microbalance the instrumentation requirements are:

Element	Number	Measurement
QCM Drive Oscillator	1	QC Frequency

Procedure: For the fixed position deposition plates, the in-flight procedure is:

Step	Action
0)	Set 8-cm Thruster at Nominal Operation Levels
1)	Open Shielding Enclosure Aperture
2)	Carry Out Deposition Plate Exposure to Thruster Efflux
3)	Close Shielding Enclosure Aperture
4)	Secure 8-cm Thruster Operation

For the movable position deposition plates the procedure steps (following Step 0) above) are:

Step	Action
0a)	Elevate Deposition Plate and Shielding Enclosure
0b)	Read Deposition Plate Angular Position

and following Step 3) is:

3a)	Return Deposition Plate and Shielding Enclosure to Stowed Position

In-flight analysis of deposition effects has procedural steps which follow Step 3) for the fixed position plates and Step 3a) for the movable deposition plates. These consist of:

Step	Action
3b)	Initiate Reference Light Source
3c)	Determine Reference Light Intensity
3d)	Determine Transmitted Light Intensity
3e)	Determine Reflected Light Intensity

If continued exposure is desired the procedure re-cycles to Step 1) for fixed position deposition plates and Step 0a) for movable position deposition plates. If no further exposure is desired, the procedure proceeds to Step 4).

For Test T5B the procedure is a continued readout of quartz cyrstal frequency with noted variations as thruster operation is initiated and terminated.

Duration: Maximum exposure duration is set at 50 hours (subject to Orbiter operational approval for more prolonged thruster running periods).

Orbiter Requirements: Possible Orbiter requirements include a record of all fluid and material release activity by Orbiter systems and possible rescheduling of fluid and material releases either pre- or post-exposure for the deposition plates. Other possible Orbiter requirements may include absence of sunlight to avoid stray light impact on the in-flight optical properties light detectors.

Post-Flight Activities: Post flight activities include recovery of deposition plates and laboratory evaluations (electron beam microprobe, ion beam microprobe, auger emission spectroscopy, optical transmission, optical reflection, solar absorptivity, infrared emissivity, chemical analyses) of exposed plates.

Title: Non-Condensible Neutral Efflux Measurements

Overall Test Designation: T6

Sub-Test Designations:

- T6A: Ionization Gauge Measurements
- T6A1: Fixed Position Ionization Gauge
- T6A2: Movable Position Ionization Gauge
- T6B: Residual Gas Analyzer

Objective: The objective of the Non-Condensible Neutral Efflux Measurements is a determination of the rate and material content of the atomic and molecular efflux from the 8-cm thruster at selected locations in the thruster system coordinate space.

Sensor Requirements: The required sensors for Test T6A, Ionization Gauge Measurements, is either a fixed position gauge (IGF), (T6A1), or movable position gauge (IGM), (T6A2). For either gauge, sensor requirements are:

Angular Position Measurement Accuracy	.01 radians
Maximum Gauge Inlet Angular Width	.1 radians
Lower End Neutral Flux Density Detection Level	3×10^{11} atoms/cm^2/sec
Minimum Radial Separation Distance	30 cm

The required sensor for Test T6B is a (fixed position) Residual Gas Analyzer (RGA). The requirements for this sensor are:

Angular Position Measurement Accuracy	.01 radians
Maximum Gauge Inlet Angular Width	.01 radians
Lower End Neutral Flux Density Detection	3×10^{11} atoms/AMU/cm^2/sec
Minimum Radial Separation Distance	30 cm

Instrumentation Requirements: For the movable ionization gauge (IGM), the instrumentation gauge requirements are:

Element	Number	Measurement
Angular Position Drive	1	Probe Angular Position
Ionization Gauge Controller	1	Ionization Gauge Current

For the fixed position gauge (IGF), the angular position drive requirement is deleted.

For the residual gas analyzer, instrumentation requirements are:

Element	Number	Measurement
RGA Controller	1	RGA Output Current as F(AMU)

Procedure: The procedure for the movable ionization gauge measurement is:

Step	Action
1)	Set Probe Position at Designated Angle
2)	Read Probe Position
3)	Activate Gauge Controller Circuitry
4)	Read Ion Gauge Output
5)	Advance Probe Position and Recycle Procedure to Step 1) Above

The procedure for the fixed ionization gauge measurements is simplified to Steps 3) and 4) of the above procedure. The procedure for the residual gas analyzer is similar to Step 3) and Step 4) above except that the RGA controller circuitry is activated and RGA output is read as a function of mass unit setting.

Duration:

15 minutes per 180° Probe Sweep (Movable Gauge)

5 minutes per fixed position ion gauge or RGA circuit activation and sensor readout

Orbiter Requirements: None

Post-Flight Activities: None

Title: Thruster Internal Erosion Measurements

Overall Test Designation: T7

Objective: The objective of the Thruster Internal Erosion Measurement is the determination of the rate of material loss at specified internal locations of the ion thruster during in-flight operation.

Sensor Requirements: The required sensors are multilayer thin film sputtering samples located at internal positions of the 8-cm thruster. The number and location of such samples shall be specified by NASA/LeRC. The sputtering depth determination accuracy shall be 100 Angstroms.

Instrumentation Requirements: None

Procedure: The sputtering samples shall be affixed at the specified internal locations of the 8-cm thruster prior to installation in the Orbiter payload. The samples shall be removed after Orbiter re-entry and payload recovery.

Duration: Maximum exposure duration is set at 50 hours (subject to Orbiter operational approval for more prolonged thruster running periods).

Orbiter Requirements: None

Post-Flight Activities: Internal erosion samples are removed after Orbiter re-entry and payload recovery and are subjected to measurement of erosion depth as determined by the total number of layers removed in the multilayer thin film samples.

Title: Charged Particle Drainage to Electrically Biased Surfaces Measurements

Overall Test Designation: T8

Objective: The objective of the Charged Particle Drainage to Electrically Biased Surfaces Measurement is the determination of the charged particle flow from the ion thruster exhaust plume to specified surfaces at varying levels of electrical bias and under varying degrees of insulating encapsulation.

Sensor Requirements: The required sensor is an electrically biasable surface (EBS). The total number of such samples, the conditions of insulating encapsulation, and the location of the sensor package on the remote test fixture (RTF) relative to the thruster test fixture (TTF) are to be determined items.

Instrumentation Requirements:

Element	Number	Measurement
Variable Bias Supply	1	Bias Voltage
Multi-Position Switch	1	Switch Position
Drainage Current Impedance	1	Charged Particle Drainage Current

Procedure: The procedure for the charged particle drainage measurements should follow the completion of Test T4 [Group IV (Charge Exchange) Ion Measurements] because the drainage to the electrically biased surfaces will result primarily from charged particle flow from the charge exchange ion plasma plume. The procedure for Test T8 is:

Step	Action
1)	Set Bias Voltage to Zero
2)	Set Multi-Position Switch to Indicated Sample
3)	Determine Zero Bias Current Drainage Signal
4)	Advance Bias Voltage Through Set ΔV
5)	Determine Charged Particle Drainage Signal at Bias Setting V
6)	Read Ion Thruster Neutralizer Current
7)	Recycle to Procedure to Step 4) above for Bias Voltage Within Specified Bias Voltage Range
8)	For Bias Voltage at Limit of Bias Voltage Range Recycle Procedure to Step 1) above and Advance Multi-Position Switch to Next Indicated Sample

Duration: 5 minutes per Bias Voltage Sweep per electrically biasable surface sample.

Orbiter Requirements: Charged particle drainage from the ambient ionospheric plasma may be significant compared to drainage currents from the ion thruster plume. Orbiter requirements may include Orbiter orientation such that the biasable surface is located within the ionospheric plasma wake region created by Orbiter motion through the space plasma.

Post-Flight Activities: Electrically biasable surface samples are removed after Orbiter re-entry and payload recovery for laboratory investigation of dielectric micro-property alterations (solar absorptivity, infrared emissivity, material bulk resistivity) and macro-property alterations (pin-hole formation and growth, large scale material removal or deposition).

Title: Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurements

Overall Test Designation: T9

Objective: The objective of the Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration Measurement is the determination of the Orbiter electrical potential relative to the potential of the space plasma for varying orientations between the thrust beam vector, $\vec{v}_+$, and the Earth's magnetic field vector, $\vec{B}_e$, and for varying configurations of the ionospheric plasma wake (created by Orbiter motion through the space plasma) and the ion thruster beam plasma.

Sensor Requirements: The required sensor is a cold floating probe, designated as the Orbiter Floating Potential Prove (OFP) and located on the Remote Test Fixture (RTF). Other sensor requirements are:

Floating Potential Measurement Accuracy	1 volt
Minimum Probe Surface Area	100 cm^2
Minimum Probe Floating Impedance	10 megohms

Instrumentation Requirements:

Element	Number	Measurement
Orbiter Floating Potential Probe	1	Probe Floating Potential

Procedure: The procedure for an electrical equilibration measurement is:

Step	Action
1)	Set Orbiter to required orientation relative to Orbiter velocity vector, $\vec{v}_o$, and set thrust beam vector, $\vec{v}_+$, at required orientation relative to Earth's magnetic field vector, $\vec{B}_e$.
2)	Measure Orbiter Floating Potential Probe Voltage
3)	Recycle Procedure to Step 1) to Continue Test Matrix in $\vec{v}_o$, $\vec{v}_+$, $\vec{B}_e$.

The procedure for Thrust Beam Plasma/Space Plasma/Orbiter Electrical Equilibration also requires the completion of Test T3, Ion Thrust Beam Neutralization Measurements.

Duration: 1 minute per floating potential measurement, following setup of required, $\vec{v}_o$, $\vec{v}_+$ $\vec{B}_e$ orientation.

Orbiter Requirements: Orbiter orientation to required attitudes and velocity vector relative angles for the complete $\vec{v}_o$, $\vec{v}_+$, $\vec{B}_e$ matrix.

Post-Flight Activities: None

Title: Multiply-Charged Ion Production Measurements

Overall Test Designation: T10

Objective: The objective of the Multiply-Charged Ion Production Measurements is to determine the ratio of doubly charged thrust ions to singly charged thrust ions (Hg^{++}/Hg^{+}) as a function of polar angle, θ, at fixed radial distance, R, in the "Transverse" Plane.

Sensor Requirements: The required sensor is a movable position (magnetic field) ion velocity analyzer (charge-to-mass analyzer) (MIP) which may be scanned through the thrust beam in the designated "Transverse" Plane. Other sensor requirements are:

Angular Position Measurement Accuracy	.01 radians
Angular Range	$-90° < \theta < 90°$
Maximum Cup Entrance Width	.02 radians
Lower End Current Density Sensitivity (Hg^{+})	10^{-6} A/cm^2
Lower End Current Density Sensitivity (Hg^{++})	10^{-8} A/cm^2
Entrance Grids	2, Separately Biasable
Ion Current Collectors	2
Hg^{++}/Hg^{+} Current Separation Capability	10^{-3}
Ion Velocity Separation	Magnetic
Minimum Raidal Separation Distance	30 cm

Instrumentation Requirements

Element	Number	Measurement
Angular Positive Drive	1	Probe Angular Position
Forward Grid Potential Supply	1	Forward Grid Potential
Rear Grid Potential Supply	1	Rear Grid Potential
Magnetic Separation Field	1	Magnetic Separation Field Current
Ion Current Collectors	2	Ion Current (Hg^{+}/Hg^{++})

Procedure: The procedure for Hg^{++}/Hg^{+} ion current density ratio measurements is:

Step	Action
1)	Set Probe Position at Designated Angle
2)	Read Probe Position
3)	Set Forward Grid Potential at Designated Level
4)	Read Forward Grid Potential
5)	Set Rear Grid Potential at Designated Level
6)	Read Rear Grid Potential
7)	Set Magnetic Field Separation Current at Designated Level
8)	Read Magnetic Field Separation Current
9)	Read Hg^{+} Ion Current Collector Signal
10)	Read Hg^{++} Ion Current Collector Signal
11)	Advance Probe Position and Recycle Procedure to Step 1) Above

The procedure for the separation of Hg^{++} thrust ions from Hg^{+} thrust ions is based upon trajectory variation following the $\vec{v}_+ \times \vec{B}_{sep}$ interaction [where $\vec{v}_+$ is thrust ion velocity (Hg^{+} or Hg^{++}) and $\vec{B}_{sep}$ is the magnetic field in the ion velocity separation region]. For a separation field provided by permanent magnets, Steps 7) and 8) above are deleted and the Magnetic Field Separation element is deleted from the Instrumentation Requirements.

Duration:

15 minutes per 180° Probe Sweep

Orbiter Requirements: None

Post-Flight Activities: None