(NASA-CR-135281) EXTENDED PERFORMANCE SOLAR ELECTRIC PROPULSION THRUST SYSTEM STUDY. VOLUME 1: EXECUTIVE SUMMARY Final Report (Hughes Research Labs.) 106 p HC A06/MF A01 CSCL 21C G3/20

N78-10205

Unclas
15090

EXTENDED PERFORMANCE SOLAR ELECTRIC PROPULSION THRUST SYSTEM STUDY

Final Report

Volume I — Executive Summary

September 1977

By

Ion Physics Department Staff

Hughes Research Laboratories

and

Technology Division Staff

Space and Communications Group

of

Hughes Aircraft Company

Prepared For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA Lewis Research Center

Contract NAS 3—20395

TECHNICAL REPORT STANDARD TITLE PAGE

1. Report No. CR-135281	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle EXTENDED PERFORMANCE SOLAR ELECTRIC PROPULSION THRUST SYSTEM STUDY VOLUME I — EXECUTIVE SUMMARY		5. Report Date September 1977
		6. Performing Organization Code
7. Author(s) R.L. Poeschel and E.I. Hawthorne, et al.		8. Performing Organization Report No.
9. Performing Organization Name and Address Hughes Aircraft Company Hughes Research Laboratories 3011 Malibu Canyon Road Malibu, California 90265 Hughes Space & Com.Group P.O. Box 92919 Los Angeles, CA 90009		10. Work Unit No.
		11. Contract or Grant No. NAS 3-20395
		13. Type of Report and Period Covered 14 Feb 1977–29 Aug 1977
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Lewis Research Center 21000 Brookpark Road Cleveland, Ohio 44135		14. Sponsoring Agency Code

15. Supplementary Notes

Project Manager: James Cake, NASA-Lewis Research Center, Cleveland, Ohio

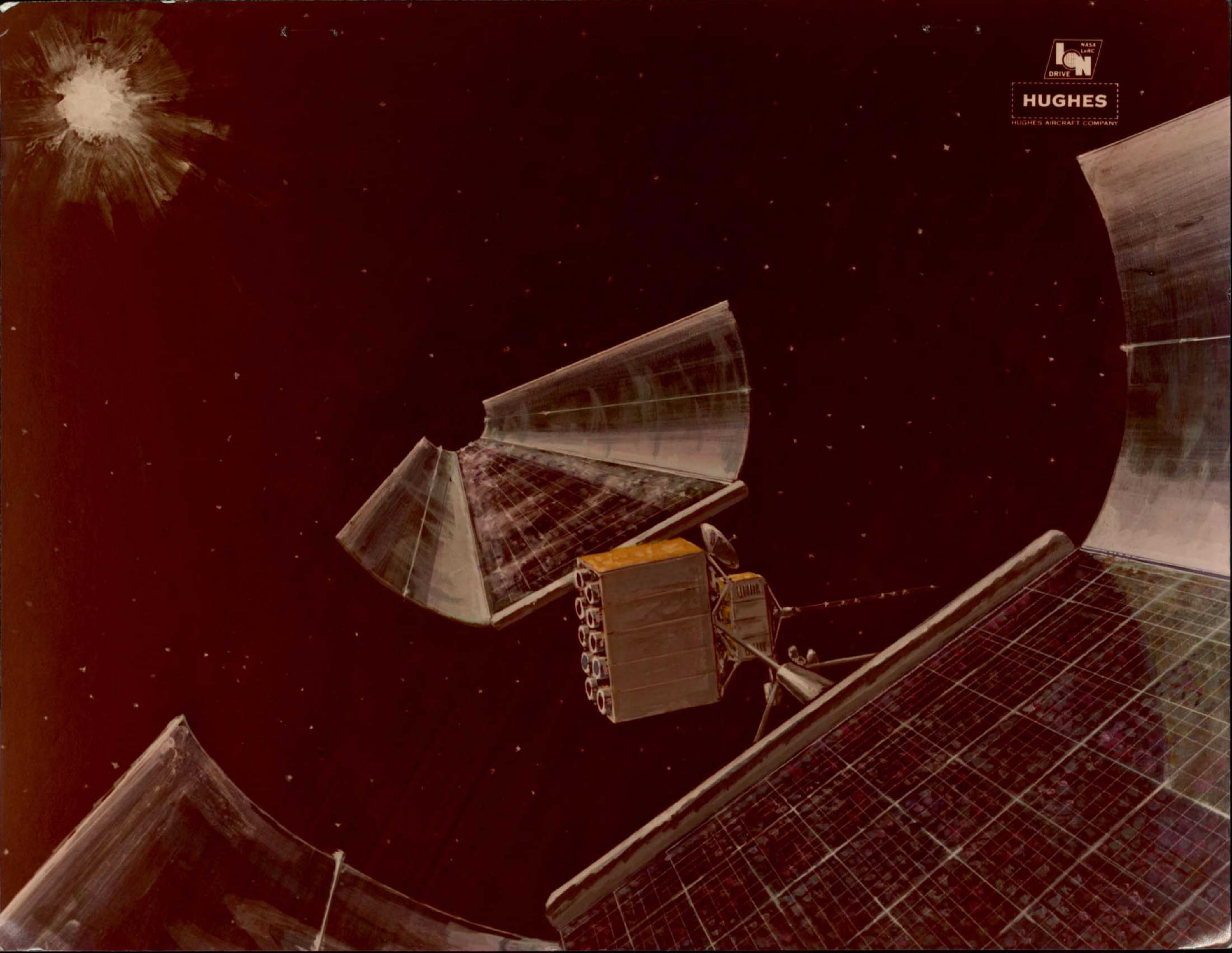
16. Abstract

Ion-thruster technology has progressed during the past decade to the point that it is considered ready for application. During this study, several thrust system design concepts were evaluated and compared using the specifications of the most advanced 30-cm engineering model thruster as the technology base. Emphasis was placed on relatively high-power missions (60 to 100 kW) such as a Halley's comet rendezvous. The extensions in thruster performance required for the Halley's comet mission were defined and alternative thrust system concepts were designed in sufficient detail for comparing mass, efficiency, reliability, structure, and thermal characteristics. Confirmation testing and analysis of thruster and power-processing components were performed, and the feasibility of satisfying extended performance requirements was verified. A baseline design was selected from the alternatives considered, and the design analysis and documentation were refined. The baseline thrust system design features modular construction, "conventional" power processing, and a "concentrator" solar array concept and is designed to interface with the Space Shuttle. A program development plan was formulated that outlines the work structure considered necessary for developing, qualifying, and fabricating the flight hardware for the baseline thrust system within the time frame of a project to rendezvous with Halley's comet during December 1985. An assessment was made of the costs and risks associated with a baseline thrust system as provided to the mission project under this plan. Critical procurements and interfaces were identified and defined. The results of this study are presented in the five volumes of this report.

17. Key Words (Selected by Author(s))	18. Distribution Statement
Solar Electric Propulsion Thrust System Ion Propulsion Ion Thruster	Unclassified-Unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
UNCLASSIFIED	UNCLASSIFIED	113	

*For sale by the Clearinghouse for Federal Scientific and Technical Information, Springfield, Virginia 22151.


NASA
LeRC
ION
DRIVE
HUGHES
HUGHES AIRCRAFT COMPANY

FOREWORD

The work described herein was performed by the coordinated efforts of personnel within two divisions of the Hughes Aircraft Company. Responsibility for the study resided in the Ion Physics Department of the Research Laboratories Division. This department is managed by Mr. J.H. Molitor. A major portion of the thrust system design activity was performed by a team of individuals assembled from the Technology Division of the Space and Communications Group and coordinated and directed by Dr. E.I. Hawthorne. The work was funded under contract NAS3-20395 and monitored by Mr. James E. Cake of the NASA Lewis Research Center. The key technical contributors were

R.L. Poeschel	- Study manager for the final phases of the study and project engineer for the approach confirmation task
E.I. Hawthorne	- Manager of all thrust system design and program development activities
Y.C. Weisman	- Project engineer for structural design
M. Frisman	- Project engineer for structural design
G.C. Benson	- Project engineer for power management and control design
R.J. McGrath	- Project engineer for thermal control design
R.M. Martinelli	- Project engineer for capacitor diode voltage multiplier development and evaluation
T.L. Linsenbardt	- Thermal analysis
J.R. Beattie	- Thruster evaluation

SUMMARY

The primary objective of this study was to provide a data base for a program plan for the development of the ion-propulsion thrust system for the Halley's comet mission spacecraft. This data base was to include: the definition of a design concept, selected from among alternate candidate configurations; the identification of required supporting technology, including the definition of critical areas and potential technical risks; the definition of a program development plan, including a development schedule and an assessment of potential schedule risks; and a preliminary estimate of yearly and total program costs.

A concurrent objective of the study was to conduct a hardware "approach confirmation" technology effort to evaluate the ion thruster's performance and lifetime at the power level required for the Halley's comet mission, to design and evaluate the thruster isolator required for operation at the higher power level, and to evaluate the design of a capacitor-diode voltage multiplier.

A thrust system baseline configuration was identified for the 30-cm extended-performance mercury ion thruster that can perform the Halley's comet rendezvous mission. The configuration is comprised of 10 thrusters configured with a power management and control system and a structure and thermal control system in a modular thrust system design. The power management and control system uses conventional power processing. Power is provided to the thrust system with an 85 kW concentrating solar array. The thrust system mass is 1010 kg (including 15% contingency), the average system efficiency is 70%, and the estimated reliability upper bound is 72%.

Adaptability of the 900-series 30-cm thruster design to the 6 to 7 kW range required for the Halley's comet mission was demonstrated with only minor design modification required, and an acceptable high-voltage isolator design was validated by laboratory tests. The design and performance of an alternate power management and control system design approach utilizing the capacitor-diode voltage multiplier was successfully demonstrated by laboratory model tests in excess of 1 kW.

The technology efforts mentioned above assisted in the identification of the level of technical risks associated with the thrust system design. These risks have been found amenable to resolution through normal engineering development and, therefore, judged to be acceptable for mission application.

The program plan, which includes the procurement plan generated for the baseline configuration is a viable plan that provides for delivery in May 1981 of the flight thrust system to be integrated with the mission module and solar array. The cost of the thrust system development program is projected to be 54 million dollars (in fiscal year 1977 dollars) excluding contractor fee, of which approximately 13.5 million dollars will be required in fiscal year 1978.

In contrast to the low technical risk, the schedule risk for initiating this program development is of particular concern. Timely approval of the authorization of 13.5 million dollars for fiscal year 1978 must be granted so that the pre-project, or advanced development, activities can be initiated.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

SECTION 1

INTRODUCTION

This report summarizes the results of a six-month study to define the design, program plan, and costs of the ion-propulsion thrust system for the Halley's comet mission spacecraft. The modular characteristics of the design developed during this study also make it applicable as the prime space propulsion system for other potential missions.

This study, which is based on an initial system characterization (completed 7 February 1977) performed by the National Aeronautics and Space Administration's Lewis Research Center (NASA LeRC), was performed in three parts:

- Design tradeoff studies (14 February to 15 April 1977) to define and compare alternate design approaches.
- Conceptual design definition, program plan, and costs of a selected design approach (15 April to 15 June 1977).
- Approach confirmation of supporting technology in selected areas.

The results of this study are presented in five volumes. This volume, Volume I, summarizes the results of the entire program. Volume II discusses the conceptual design, program development plan, and cost estimates for the selected baseline thrust system design. Volume III describes the design tradeoff studies performed to compare alternate design approaches. Volume IV presents the results of the evaluation of the technology approach. Volume V presents the details of the capacitor-diode voltage multiplier (CDVM) circuit analysis and experimental evaluation. The results reported in these volumes have also been presented in briefings at NASA LeRC.

A. BACKGROUND

In the fall of 1976, the Office of Aeronautics and Space Technology (OAST) was given the responsibility of assessing the capability of the electric propulsion technology under development at NASA LeRC and of the solar array technology under development at Marshall Space Flight Center (MSFC) and the Jet Propulsion Laboratory (JPL) to perform the Halley's comet rendezvous mission proposed by JPL. OAST established an "August Project" team from members of the three organizations to develop a preliminary program plan to support a fiscal year (FY) 1979 new start.

The August Project consisted of parallel efforts by JPL, NASA LeRC, and MSFC to define the design approach, program plan, costs, and risks of the Halley's comet mission. Three areas were considered: the spacecraft (including the science payload), the ion propulsion subsystem (referred to as the thrust system in this report), and the solar array. The NASA LeRC program was conducted in two phases. First, initialization studies (completed 15 February 1977) were conducted to define requirements and to identify preliminary design characteristics. Second, during the 15 February to 15 July period, the design of the thrust system was defined, the program plan and projected costs were generated, and a risk assessment was made. The results of the second phase of the program are reported in this volume. The design selection process included tradeoff studies among alternate design approaches, followed by a refinement of the conceptual design that had been selected. Iteration with design data available from the parallel activities at JPL and MSFC, and concurrent approach confirmation tests and analyses included in this study, strengthen the conclusions of the thrust system study.

NASA directed us to begin the study by identifying two candidate solar array configurations (flat or concentrator), three candidate power management and control (PMaC) approaches (conventional, direct drive, or voltage multiplier), and two structural design approaches (modular or integrated). A comparative assessment of the various configurations possible from combinations of these design choices was desired in terms of performance, mass, efficiency, reliability, and technical and schedule risks.

The thrust systems being considered are based on the electric propulsion technology that NASA LeRC has been developing for over a decade. The technical baseline for this application is the most recent operational engineering model thruster (EMT), the 900-series 30-cm mercury ion EMT. This thruster is a scaled-up version of the 15-cm thruster developed and flight tested during the 1960-1969 period for the SERT II program. The EMT operates at a 3 kW power level with a specific impulse of 3,000 sec. By making minor modifications in the existing thruster design, extended performance at approximately 6 kW power level, 4,800 sec specific impulse, and 15,000 hr pre-wearout life (as required for a Halley's comet mission) was believed to be achievable at a low technical risk. This supposition was evaluated as part of this study.

In addition to the extended-performance thruster, the key elements of the thrust system for this extended-performance application are the PMaC subsystem, gimbal system, propellant storage and distribution system, thermal control system, and supporting structure. The background of extensive development in power-processing technology for mercury ion thrusters and technology developments in the other areas were the basis for the high level of confidence that the required extended performance levels could be achieved.

B. SCOPE

The scope of this study included: the development of conceptual designs for various candidate systems; the selection, definition, and evaluation of a baseline design concept and its critical interfaces; an evaluation of the sensitivity of the baseline design to critical data base and design parameters; the generation of a development program plan for the baseline concept; estimation of costs and fiscal year funding requirements; fabrication of a demonstration scale model; and the conduct of supporting technology studies (including fabrication and testing of critical hardware components) to estimate the physical and electrical performance and to provide a baseline for subsequent work.

The design characteristics, program plan, and costs of the baseline system were defined in parallel with the supporting technology effort. Design definition was carried out in two consecutive phases:

- Phase 1: Definition and comparison of alternate configurations, leading to baseline selection.
- Phase 2: Design definition and evaluation of the baseline configuration, culminating in the generation of a program plan and cost estimates.

The concurrent technology effort comprised thruster performance and lifetime evaluation, thruster isolator design and evaluation, and the design and evaluation of a CDVM breadboard.

The design study was necessarily limited to the conceptual definition of the key design features and characteristics. However, sufficient understanding was achieved in all important areas to provide realistic estimates of masses; power requirements, which led to efficiency calculations; complexity and parts count, which led to reliability estimates; development, procurement, fabrication, and test requirements, which led to schedule definition; potential areas of uncertainty and concern, which led to an assessment of the technical and schedule risks; the scope and nature of system interactions, which led to the definition of principal interfaces; and requirements and phasing for hardware and manpower, which led to a cost estimate.

The scope of and the approach to this study are reflected in this volume. Section 2 summarizes the Phase 1 configuration tradeoff studies (discussed in more detail in Volume III). Sections 3, 4, and 5 describe the key features of the selected baseline design, the program plan, and the estimated costs, respectively. (These are treated in more detail in Volume II.) Results of the supporting technology work are summarized in Section 6 and described in greater detail in Volumes IV and V. Section 7 presents study conclusions and an assessment of interfaces and of technical and program risks.

SECTION 2

DESIGN TRADEOFF STUDIES

The initial phase of this study considered a spectrum of alternative design concepts and approaches. The objective was to select the most promising configuration from the standpoint of performance and risk. The configuration selected, which then became the recommended baseline approach for the Halley's comet mission, was then assessed in terms of its design characteristics and performance, culminating in the preparation of a program plan and cost estimate. Results of these initial tradeoff studies are presented in this section.

A. CANDIDATE CONFIGURATIONS

A block diagram of the thrust system is shown in Figure 1. Each block contains elements that are possible parameters for tradeoff studies. The shaded blocks represent elements for which the only possible options were determined by NASA LeRC, and appropriate interface specifications were given. For the other elements, some flexibility was permitted. Table 1 lists the possible options for the elements of the block diagram that were purposely varied. Only seven combinations of these options were specified by NASA LeRC for detailed study. These combinations are given in Table 2; a coding system is included for each reference.

The configurations examined included:

- Comparison of the three PMaC design concepts in a modular thrust system design, using a flat solar array, and of conventional and direct drive discharge supplies as a subset (under direct-drive PMaC).
- A full examination of the matrix of flat versus concentrator array and modular versus integrated design approaches, using the conventional PMaC concept.

These choices, which approximately bracket the spectrum of alternate design concepts derivable from Table 1, were expected to provide a reasonable basis for selecting the baseline approach. The number of thrusters selected for each configuration (see Table 2) was derived from trajectory analysis and was not considered to be an independent design option.

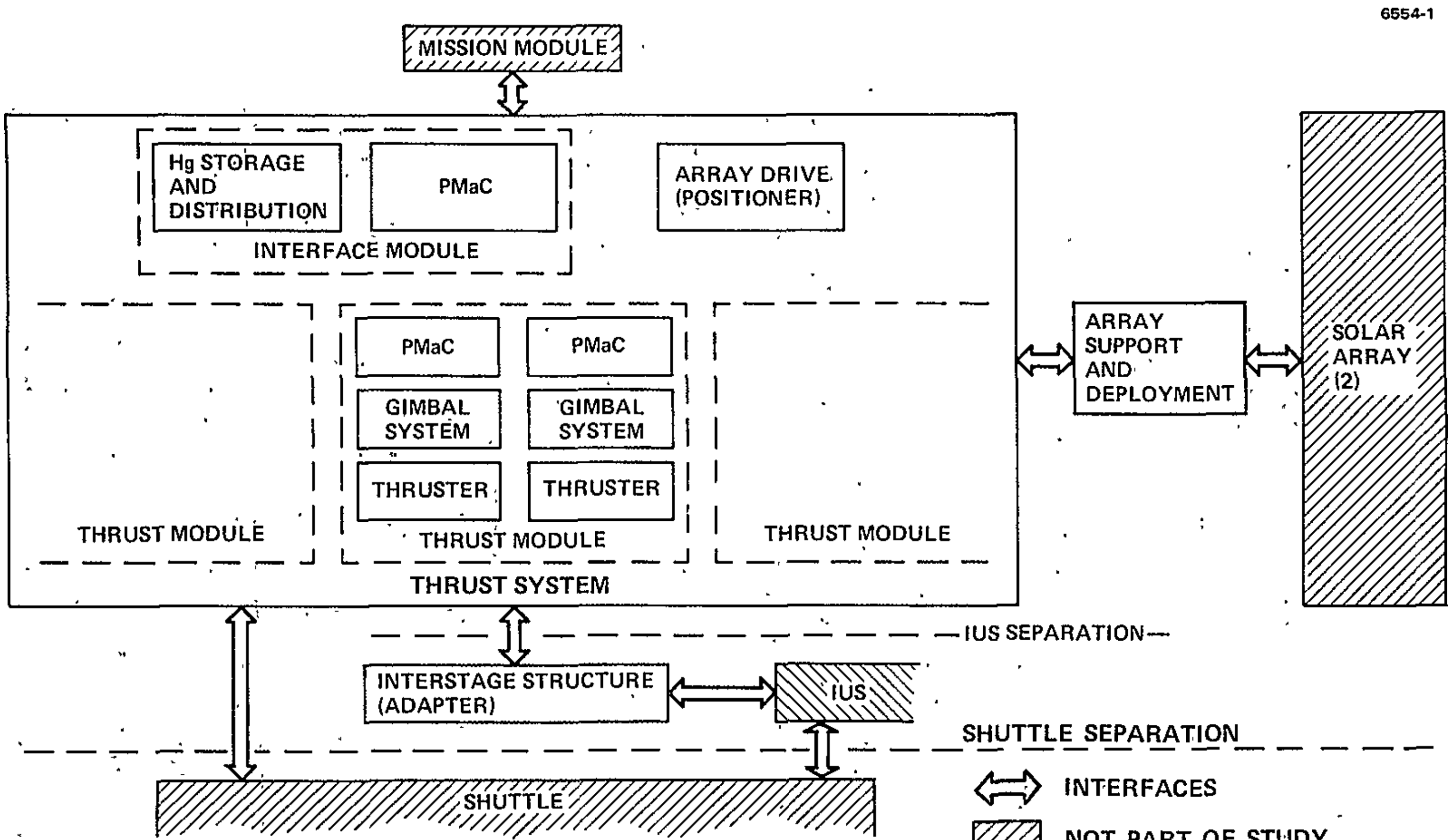


Figure 1. Thrust system block diagram showing principal interfaces.

Table 1. Study Options

Option	Code
Solar array	
Flat	A
Concentrator	B
PMaC	
Direct Drive	1
Conventional discharge supply	(none)
Direct drive discharge supply	X
Conventional	2
CDVM	3
Modularity	
Modular design	(none)
Integrated design	1
Number of thrusters (modules): determined by mission requirements	(none)

T5866

Table 2. Selected Study Configurations

Option Code	PMaC	Solar Array	Modularity	No. of Thrusters
1A	Direct drive conventional discharge supply	Flat	Modular	12
1AX	Direct drive direct discharge supply	Flat	Modular	12
2A	Conventional	Flat	Modular	10
2B	Conventional	Concentrator	Modular	10
2A/I	Conventional	Flat	Integrated	10
2B/I	Conventional	Concentrator	Integrated	10
3A	Capacitor-diode voltage multiplier	Flat	Modular	10

T5866

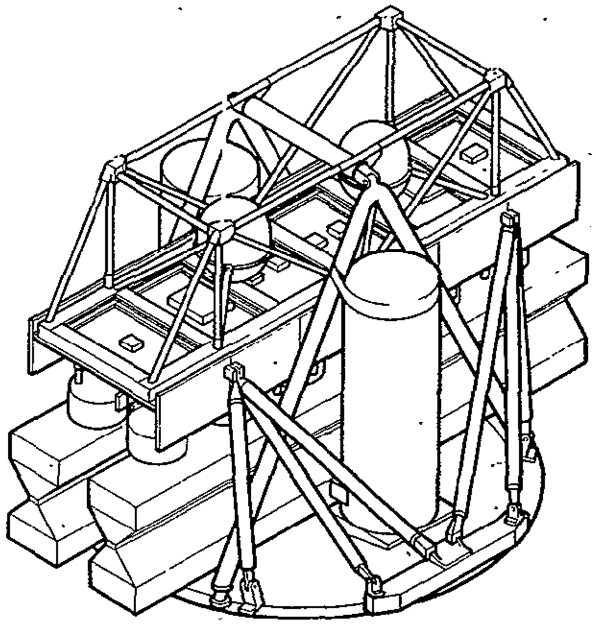
The concentrator solar array used in this design tradeoff study was not the same as that subsequently furnished by NASA-LeRC for the more detailed analysis of the selected baseline. Nevertheless, the comparison of the seven configurations is believed to have furnished a valid basis for the final choice.

B. DESIGN ANALYSIS

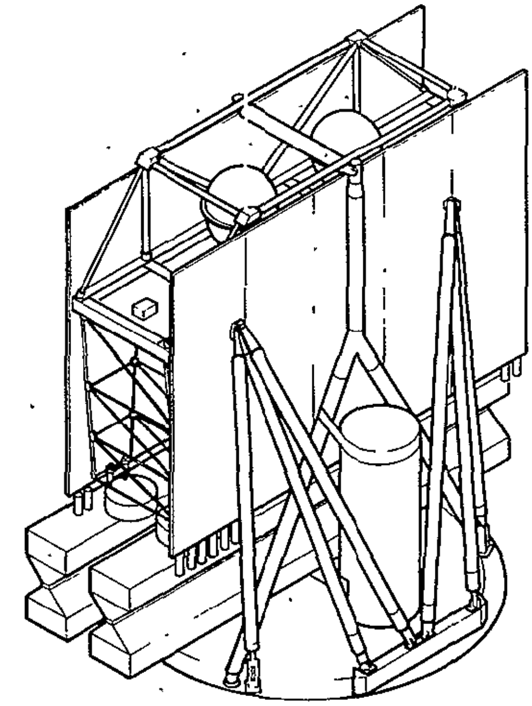
Each of the seven configurations was studied in sufficient depth to assess their design features; interfaces; performance in terms of mass, efficiency, and reliability; and technical and schedule risk. The analysis encompassed selection of thruster parameters and operations profile; PMaC design and sizing; thermal control tradeoffs and design; structural design to accommodate the stowed array and thrust system requirements accounting (loads and interfaces were taken into account in the design); and materials selection. Layouts were generated for each configuration. Since two alternate stowage concepts were examined for the flat array, a total of eight configuration layouts were developed.

Results of this analysis (see Volume II for details) are illustrated in Figure 2, which shows isometrics of representative configuration designs (the solar array is shown stowed). Figure 2(a) describes the two direct-drive PMaC configurations for one of the two flat array stowage concepts. Figure 2(b) describes the similar conventional PMaC configuration with the flat array (differs from the direct-drive configuration primarily in that larger thermal radiators are required to accommodate the larger power dissipation of conventional PMaC approaches). The voltage multiplier PMaC configuration (not shown) falls between these two designs. Figure 2(c) describes the integrated configuration. The modular design approach has been abandoned in favor of an integrated design to reduce total system mass: The thrusters are shown placed on the circumference of the circular thruster array. Figure 2(d) describes the modular configuration, which uses a concentrator array and conventional PMaC approach. The space required to stow the large array dominates this configuration. The final configuration studied was an integrated design that uses a concentrator array and conventional PMaC approach (not shown).

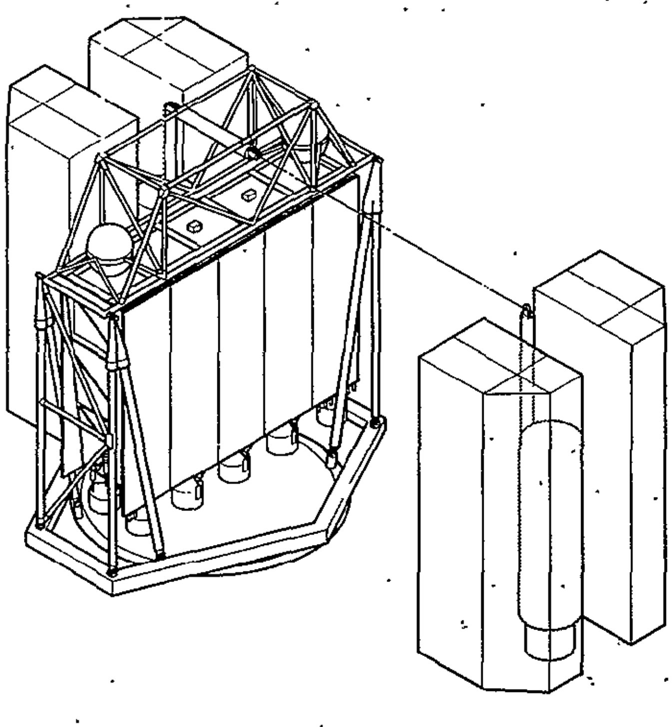
6554-10



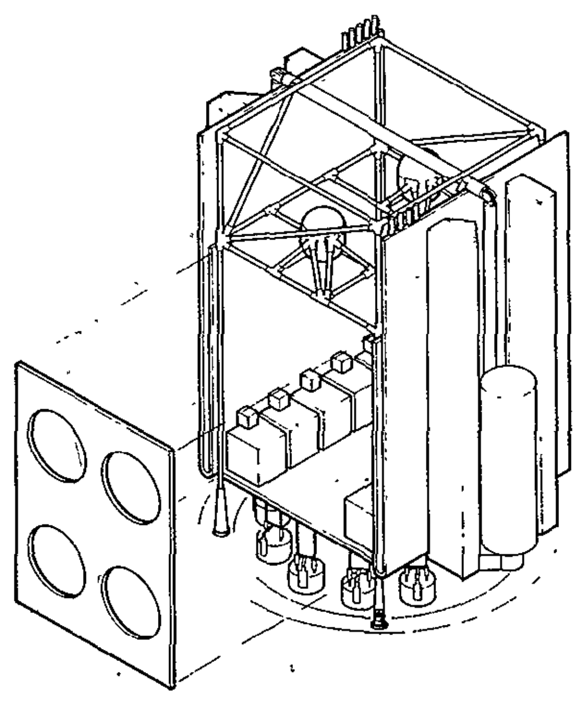
(a) Direct drive PMaC approach, flat array, modular. Configurations (1A and 1 AX).



(b) Conventional PMaC approach, flat array, modular. Configuration (2A).



(c) Conventional PMaC approach, concentrator array, modular. Configuration (2B).



(d) Conventional PMaC approach, flat array, integrated. Configuration (2A/I).

Figure 2. Isometrics of representative configurations.

These seven thrust systems differ as to mass, reliability, efficiency, and other respects. Those key characteristics that influenced the selection of the baseline configuration are summarized below.

C. COMPARATIVE ASSESSMENT AND BASELINE SELECTION

The design features and the mass, efficiency, reliability, and risk of the resulting thrust system were compared for the seven configurations. This assessment (which is summarized here) was a contributing but not the sole factor in the final selection of a baseline design by NASA LeRC. The other factors considered are also discussed here.

The seven configurations are compared in Table 3 with respect to several criteria. Differences in lengths are not considered to be dominant criteria because all configurations would fit comfortably in the shuttle bay, allowing for the length of the intermediate upper stage (IUS) and for the mission module/science payload (NASA LeRC specified 2.5 m for the mission-module payload). The other parameters in Table 3 were weighted in the final selection.

Direct drive configurations show a significant advantage with respect to mass. All conventional PMaC configurations would result in a relatively high initial mass. Comparing the integrated and modular configurations shows that an integrated configuration does not result in a significant saving in initial mass.

Direct drive configurations also have an advantage over conventional PMaC configurations; in terms of thrust-system relative reliability and efficiency, the voltage multiplier configuration is between the two.

From a risk standpoint, however, which was heavily weighted in the final selection, direct drive configurations were considered significantly less attractive. Significant risks are projected not only for the thrust system but also for the solar array configuration: thruster power levels are highest; thruster/PMaC interactions and potential high-voltage effects are not adequately known; operational flexibility in terms of parameter adjustments is limited; solar array high-voltage technology represents a novel design, with potentially detrimental high-voltage effects; full-scale ground-test validation poses significant difficulties. The

Table 3. Comparison of Candidate Thrust System Configuration.

Thrust System Characteristics	Configuration Designation						
	1A	1AX	2A	2A/I	2B	2B/I	3A
Type of configuration							
PMaC approach	Direct drive[a]	Direct-drive[b]	Conventional	Conventional	Conventional	Conventional	CDVM
Solar array	Flat	Flat	Flat	Flat	Concentrator	Concentrator	Flat
Design approach	Modular	Modular	Modular	Integrated	Modular	Integrated	Modular
Comparison criteria							
Length, m	2.9	2.9	4.4	4.6	4.9	4.6	3.7
Mass, kg							
Thrust system	650	610	1000	1060	1050	1020	840
Mission module	450	450	450	450	450	450	450
Solar array	700	700	700	700	700	700	700
Propellant	2130	2130	2240	2240	1950	1950	2250
Initial injected, after IUS separation	3930	3890	4390	4450	4150	4120	4240
Adapter	50	50	110	20	220	150	110
IUS payload	3980	3940	4500	4470	4370	4270	4350
Average thrust system efficiency, %	73	73.5	67.5	67.5	68	68	68.5
Relative reliability[c]	0.93	1.0	0.87	0.87	0.88	0.88	0.92
Technical risk	High	Highest	Low	Low	Low	Low	Medium

[a]For beam power only.

[b]For beam and discharge power.

[c]Assumes value of 1.0 for direct-drive configuration 1AX.

conventional PMaC design presents the lowest risk. The voltage multiplier design, currently under investigation, must still be considered a relatively high risk approach as compared to the conventional design. The risk, however, is considerably less than for direct drive. Current development of the voltage multiplier concept, reported in Section 6, may significantly reduce the risk.

The seven configurations were compared briefly with respect to several other criteria. Although additional differences between configurations were identified, these differences were not sufficiently important to significantly affect the final selection. All configurations are feasible from the structural and thermal standpoint, although the degree of design complexity and difficulty would vary. The configurations differed in IUS interface complexity, the difficulty of mechanizing separation, accessibility, methods by which mercury ion impingement on the solar array could be avoided, and in the difficulty of packaging and deploying the solar arrays. The packaging of the stowed concentrator solar array (for both the modular and integrated configurations with conventional PMaC) would be cumbersome and costly in mass; this originally was a deterrent to selection of the concentrator array. A modified stowage envelope having a shorter stowed length was subsequently recommended to the solar array designers by NASA LeRC; the resulting solar array envelope enabled a viable concentrator array thrust-system configuration to be designed.

The final selection by NASA, which was partly based on the acceptance of this modified concentrator design, was to adopt the modular, conventional PMaC concentrator array configuration as the baseline. The rationale for this selection may be summarized as follows:

- The direct drive PMaC approach was rejected because of high risk.
- The flat solar array was rejected because of risks associated with the thin solar array cells, and because of the high mass of the resulting configurations.

- The integrated configurations were rejected because the relatively small mass saving was not sufficient justification for abandoning the modular approach and its advantages.

- The concentrator array was considered superior, particularly because the modified design was expected to result in a lower system mass and much more manageable packaging. Both expectations were later confirmed by the study of the baseline design; the stowed array package, however, had to be quite significantly modified. The conventional PMaC approach was adopted because of its relatively low risk, and because an acceptable system mass, reliability, and efficiency were expected; this was subsequently validated during the study of the baseline design.

- The concentrator array with the CDVM PMaC approach (not explicitly studied) was considered as a potential alternative to the baseline, depending on progress in CDVM development.

During the remainder of the thrust system study (conducted after 20 April 1977), the conceptual design was refined, a program development plan was prepared, and costs were estimated for the selected baseline thrust system configuration. This work is summarized in the following sections.

SECTION 3

BASELINE DESIGN

A. DESIGN SUMMARY

The key features of the baseline thrust system design are summarized in this section. The principal characteristics of and interfaces with the other major elements of the spacecraft, mission module, and solar array are presented in the form of a design data base. The resulting performance characteristics (including mass, efficiency, and reliability) of the baseline thrust system are also presented. The design characteristics of the various subsystems that comprise the baseline thrust system are discussed in more detail in Section 3.A.1.

1. Data Base

Defining the baseline design of the thrust system from the design concepts selected during the configuration trade studies (Section 2) required making assumptions regarding principal characteristics of the mission module and solar array and of their interface with the thrust system. These assumed characteristics — the data base for the proposed design — are summarized in Table 4; supporting data is given in Figures 3, 4, and 5.

This data was used in defining the electrical, structural, and thermal design specifications and in determining the system performance; the data is also referenced in the thrust system design description. The key input to the thrust system design is the postulated power profile, shown in Figure 3. The stowed array configuration shown in Figure 5 helped in defining and sizing the thrust system structure. The length of the baseline structure is, in fact, wholly determined by the length of the stowed array.

Table 4. Data Base Summary

Solar Array Data
85 kW concentrator array 3:1 concentration ratio (max) Conventional solar cells
Power profile: 48 kW to thrusters (1.0 to 1.8 AU); see Figure 3
Voltage/current profiles provided (not shown in Figure 3): max voltage swing over trajectory: 2.6 to 1 (without reconfiguration)
Thermal characteristics (see Figure 4)
Deployed configuration (see Figure 5(a)) Side reflector angle: 45^{o} and 60^{o} (adjustable during mission) Separation distance from thrust system sufficient to ensure Hg impingement angle of 50^{o} min at 0^{o} gimbal angle Natural frequency at root of drive structure: 0.015 Hz
Stowed configuration (see Figure 5(b))
Mission Module
Weight: 450 kg
Height: 2.5 m (1.5 m above thrust-system interface plane)
Lowest lateral frequency: 30 Hz
Internal temperature: 5 to $50^{o}C$
Conductance to interface truss: 0.01 $W/^{o}C$
Emittance of multilayer insulation blanket: 0.025
Thrust system interface area: 1.13 m^2
Power requirement Thrust phase: 400 W (max) Rendezvous phase: 650 W (max)

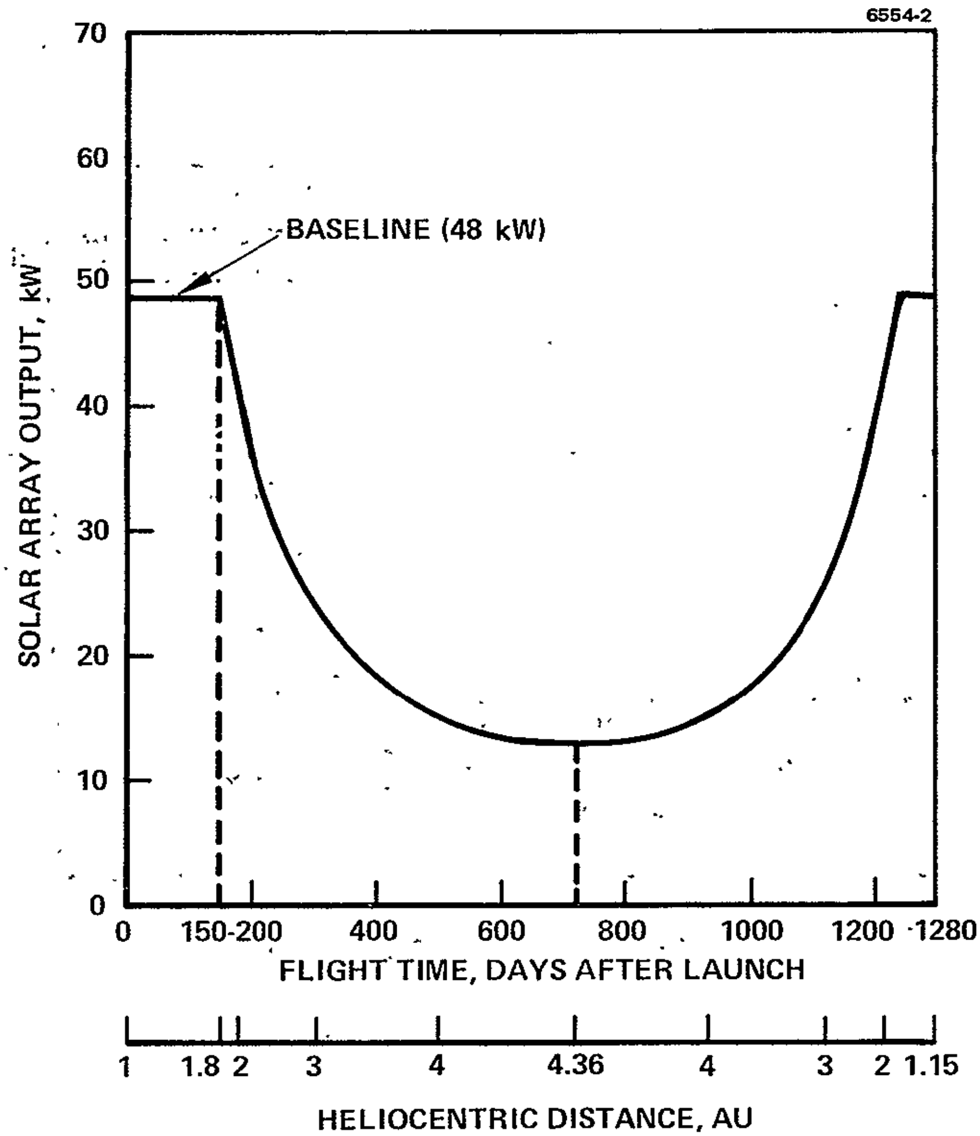


Figure 3. Power profile of the main solar array.

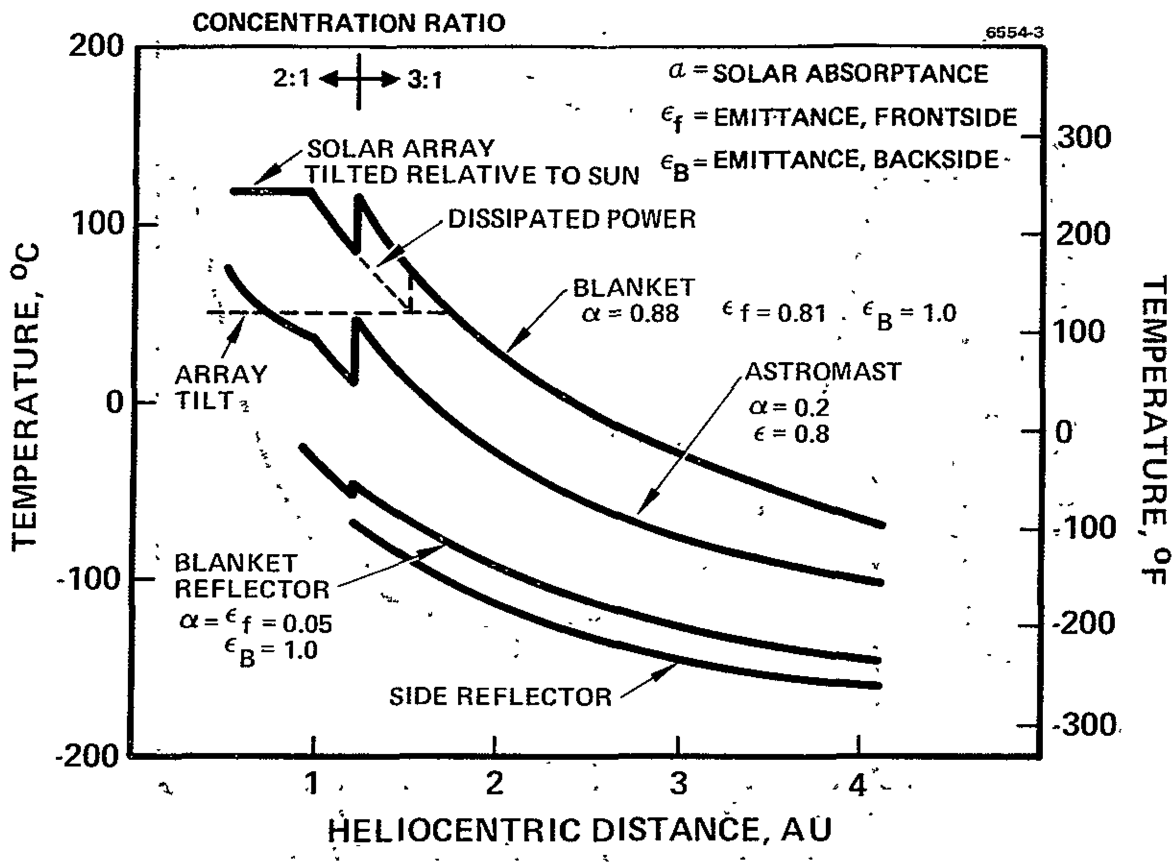
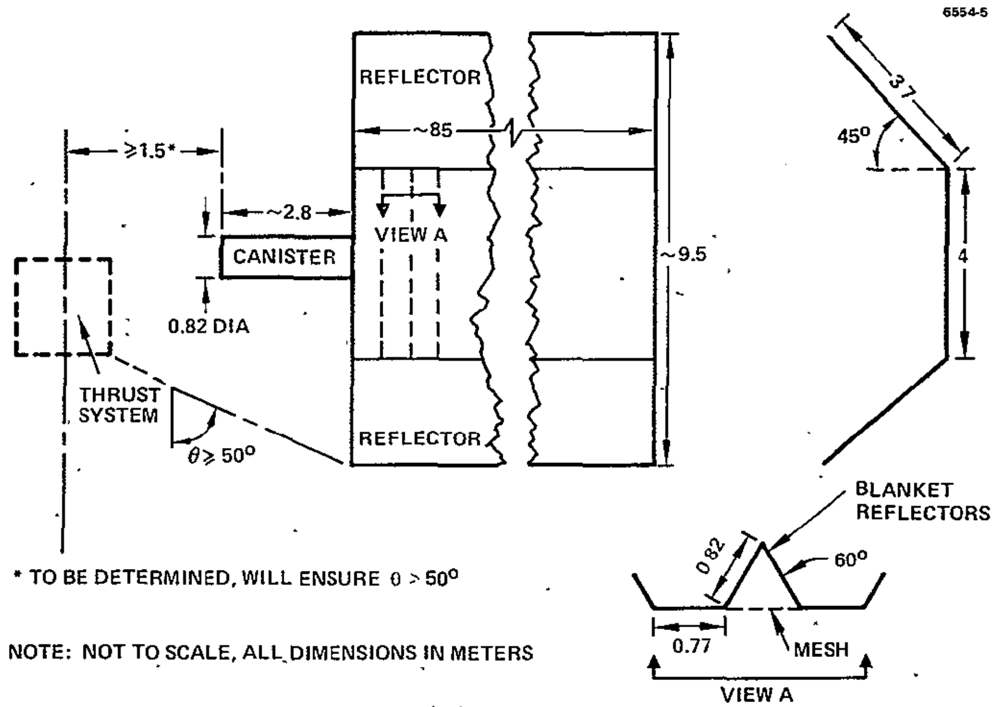
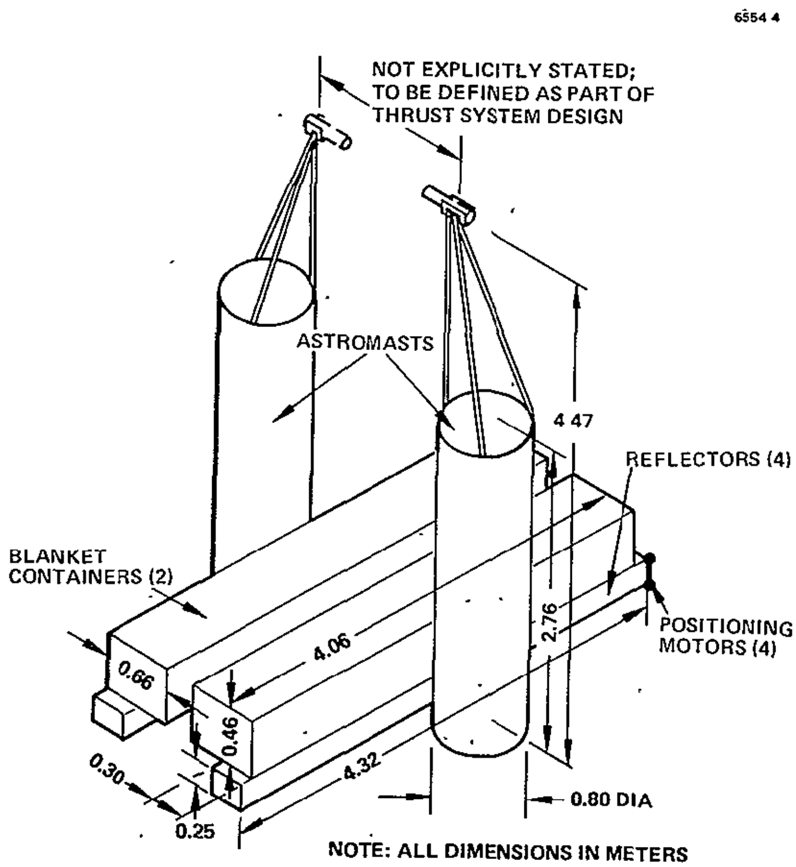


Figure 4. Thermal characteristics of the solar array.



(a) Deployed



(b) Stowed

Figure 5. Configuration of the solar array.

2. Thrust System Description

The thrust system, shown in Figure 6 in both the stowed and deployed configurations, consists of an interface module and five thrust modules. Each thrust module consists of two sets of thrusters and gimbals and of the associated PMaC beam, discharge, and low-voltage power supplies, packaged in a common modular assembly per thruster. Each module also includes a thermal control assembly of two radiators, a cold plate, and embedded heat pipes. The thrust module PMaC packages are mounted on the bottom of the cold plate. Distributed over the top of the cold plates and mounted on them are the common interface module PMaC units: power distribution units, distribution inverters, dc/dc converters, and controllers. The interface module also houses the Hg propellant reservoir system and the two solar array drives. The thrust system is designed to provide for full design modularity; the modules are essentially interchangeable. The design may be altered to decrease or increase the number of modules with relatively minor interface module modifications, and the individual module designs may be applied to other missions.

The stowed configuration in Figure 6 shows the solar array and the thrust system-IUS adapter consisting of four beryllium tripods. This structural configuration is designed to withstand the IUS loads that dominate the design requirements. On one end, the adapter is mounted to the IUS ring, and to the cross beams added to the ring; on the other end, it is attached to the interface module structure at four points. Separation from the IUS is accomplished by releasing the thrust system at these four points (which permits the members of the adapter tripods to separate and swing away) and by releasing the solar array cannisters (which are fastened directly to the cross beams) with appropriate separation mechanisms.

The length of the thrust system, 4.7 m, is largely determined by the length of the stowed solar array. Not shown in Figure 6 are the mission module and the science payload, which are mounted on top of the interface module. The overall length of the IUS/thrust-system/mission-module/payload configuration is estimated to be well within the shuttle bay length of 18.3 m (60 ft). When installed in the shuttle bay, the thrust system

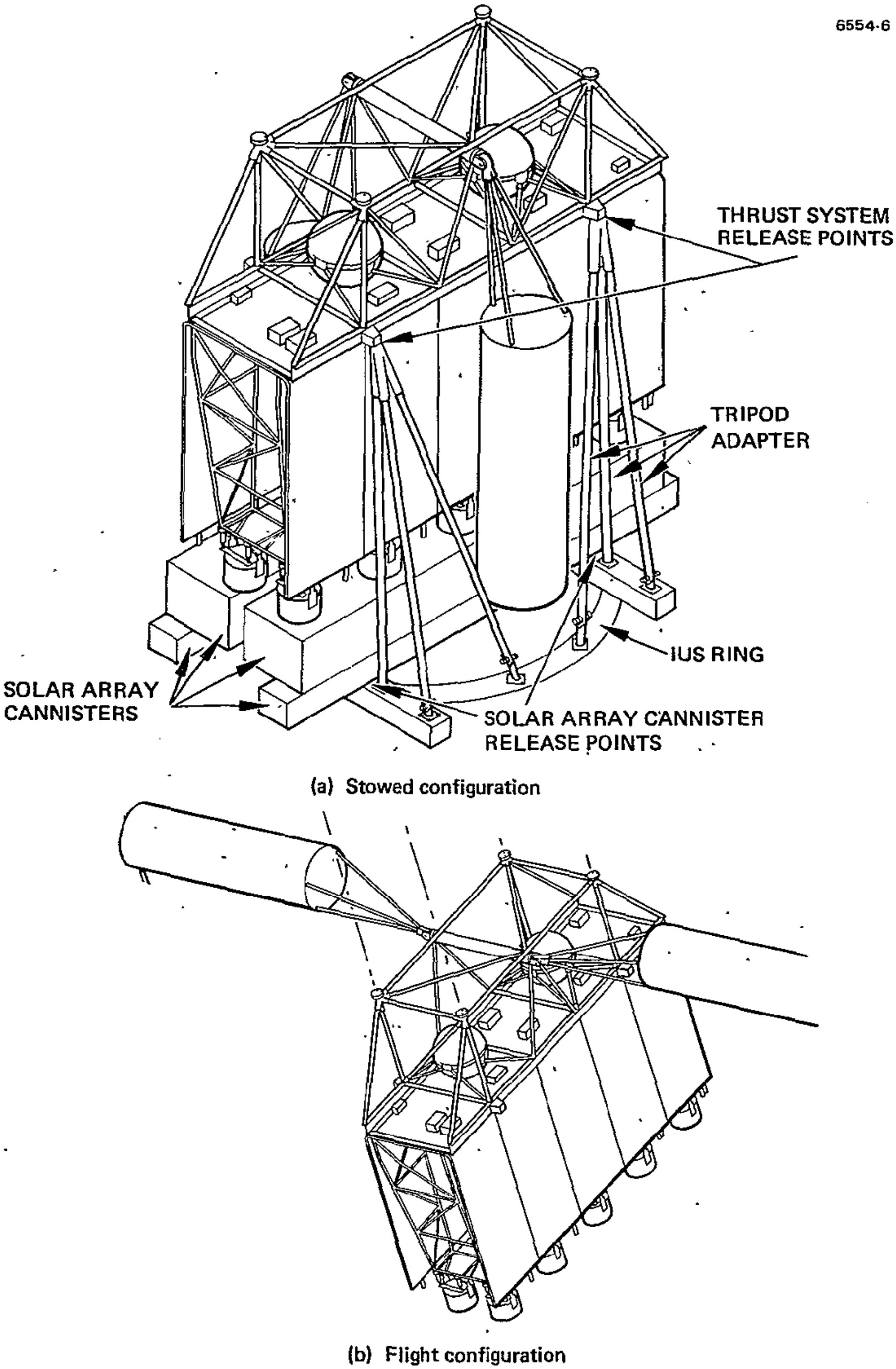


Figure 6. Thrust system isometric.

ORIGINAL PAGE IS
OF POOR QUALITY

will utilize a forward attach cradle between the interface module and the shuttle (not shown in Figure 6) to withstand the shuttle launch loads.

The thrusters are 30-cm, 900-series EMTs modified for operation at higher beam voltage: the associated gimbals provide for thruster gimbaling of up to 5° in the direction toward the astromast and 35° in the direction perpendicular to the astromast. The thrust module PMaC beam and discharge supplies are of conventional series resonant inverter design. The Hg propellant system utilizes a common, dual tank system.

3. Thrust System Operations and Performance

The thrust system operations profile was defined using the power profile in Figure 3 and the preliminary mission/trajectory data furnished by NASA LeRC. This operations profile is consistent with the June 1982 launch and December 1985 Halley's comet encounter dates and with the constant specific impulse assumed in the trajectory analysis. The thruster parameters and the management plan were selected (from among several alternatives) to achieve the highest possible thrust system efficiency and reliability consistent with a propellant mass at least as low as that allotted. However, this selection cannot be considered optimum, since it is implicitly related to mission flight dynamics and system design. Extensive iteration between the thrust system parameters and performance, total spacecraft system design characteristics, and mission/trajectory must be performed to arrive at an optimal set.

The key thruster parameters selected are shown in Table 5. The maximum power per thruster, P_{MAX}, was selected to be 6.4 kW, with an assumed constant beam voltage of 3,000 V. This is compatible with the postulated thruster capability, and it is consistent with the criterion for maximizing thrust-system efficiency and reliability:

- Operating with an integral number of thrusters at power levels close to P_{MAX} under high solar panel power conditions (48 kW at the beginning and the end of the mission).

- Operating with a minimum of two thrusters (rather than one) at power levels close to P_{MAX} to provide the thrust vector control function under conditions of low solar panel power (at large heliocentric distances).

With P_{MAX} = 6.4 kW, the above criteria led to the use of seven thrusters at low heliocentric distances and two thrusters at high heliocentric distances.

The resulting thruster operations plan is shown in Figure 7. This profile was generated by applying several additional criteria that optimize reliability: (1) equalization of total hours per thruster among operational thrusters; (2) keeping one thruster as spare (i.e., spreading the total thruster/mission hours among nine operational thrusters); and (3) turning thrusters on and off individually, rather than in pairs. This results in an average of approximately 13,600 hr of operating time per thruster, which provides a reasonable margin below the stipulated 15,000 hr of thruster life expectancy (prior to wearout). Although with ten operational thrusters the average hours per thruster would be correspondingly lower (by a factor of 9/10), standard reliability prediction algorithms indicate that the overall reliability without a spare would be significantly lower.

The thrust-system reliability predictions, shown in Table 5, were derived using the above thruster parameters, estimates of expected thruster failure rates prior to wearout, and reliability estimates of other thrust-system components. Lack of adequate data on expected thruster failure rates required that the results be given in terms of an estimated range, indicated in Table 5, corresponding to the range of failure rates believed to correctly bracket the expected thruster reliability. Reliability of other thrust-system components, which was estimated from a detailed analysis of components characteristics and parts counts, is believed to be reasonably accurate.

Table 5. Thrust System Performance Summary

System Characteristics		Comments
Thruster parameters		
9 operational thrusters and PMaC supplies		1 spare
Number of thrusters operating simultaneously		
7 maximum		Low AU (48 kW array power)
2 minimum		High AU
Maximum power per thruster: 6.4 kW		3 kV beam voltage (constant)
Average operating time per operational thruster: T = 13,600 hr		Same for each of the 9 thrusters
Thruster efficiency at average power: 76.2%		Average power = 5.9 kW
Thruster reliability = exp $(-\lambda T)$		Expected life = 15,000 hr
$10^{-6} < \lambda < 10^{-5}$		T < 15,000, λ = failure rate, failures/hr
Average thrust system efficiency: 70%		Excluding 400 W to mission module
Thrust system reliability range		
Estimated lower bound: 37%		$\lambda = 10^{-5}$ (pessimistic)
Estimated upper bound: 72%		$\lambda = 10^{-6}$ (closer to expected)
System mass, kg		
Thrust system, dry[a]	1010	Including 15% contingency and Hg residuals
Hg propellant	1810	
Solar array	700	Assumed (given by NASA LeRC)
Mission module	450	Assumed (given by NASA LeRC)
Total injected after IUS separation	3970	
Adapter[a]	130	Including 15% contingency
Total IUS payload	4100	

[a]See Section 3.B.3 for breakdown.

ORIGINAL PAGE IS OF POOR QUALITY

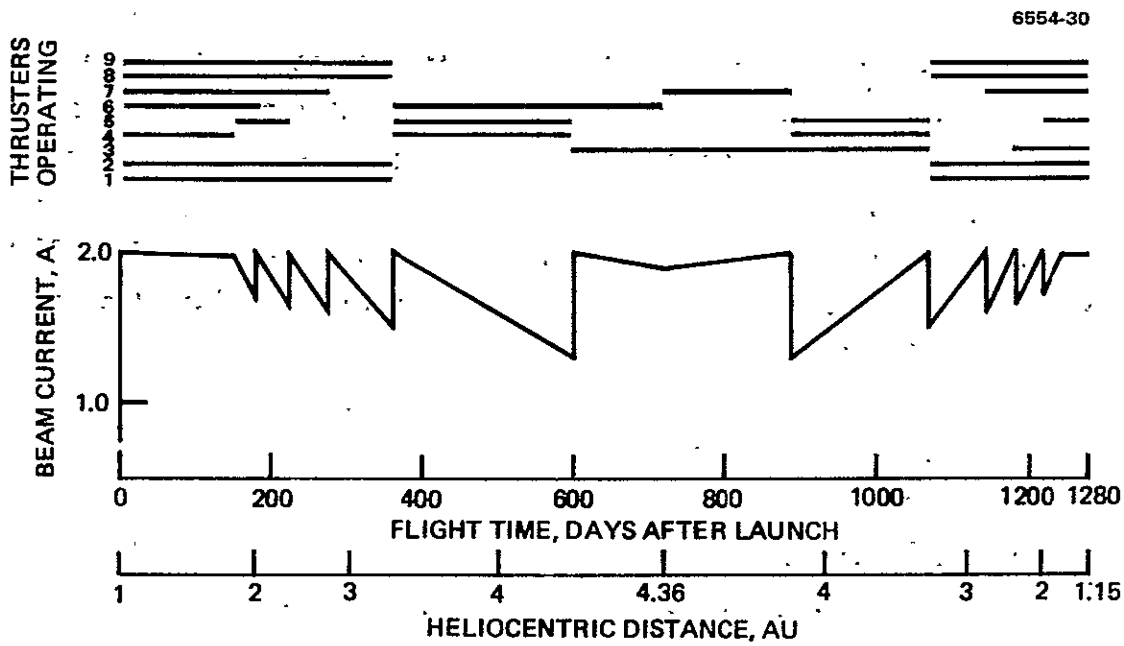


Figure 7. Thruster operation profile.

Average thrust-system efficiency, shown in Table 5, was calculated using the predicted thruster efficiency of 76.2% at an average thruster power level of 5.9 kW. Also considered in the calculation were thrust-system power requirements and dissipations, including housekeeping functions but excluding the 400 W of power supplied to the mission module. The indicated average thrust-system efficiency of 70% provides one input to the iterative system/trajectory analysis.

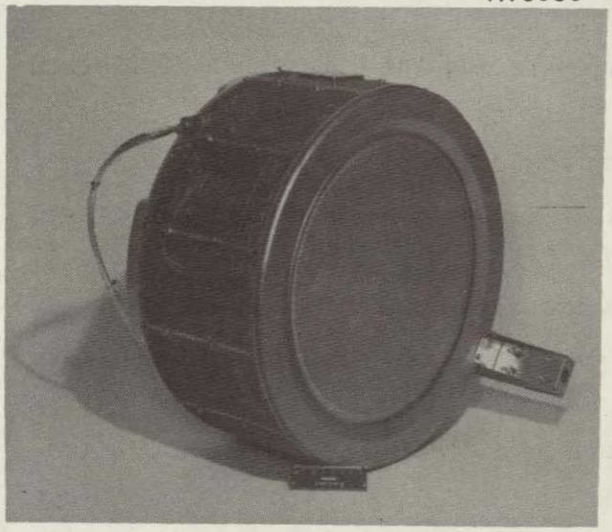
Table 5 shows estimates of system mass; these are based on thrust system estimates (presented in more detail in Section 3.B.3) and on mercury propellant requirements given by the thruster/power-profile analysis above. Thrust system mass estimates include a contingency of 15%. Using the mass estimates for the solar array and for the mission module provided by NASA LeRC, the resulting IUS payload mass of 4,100 kg is within the IUS capability for the launch energy determined from the trajectory analysis.

B. DESIGN DESCRIPTION

1. Thruster and Gimbals

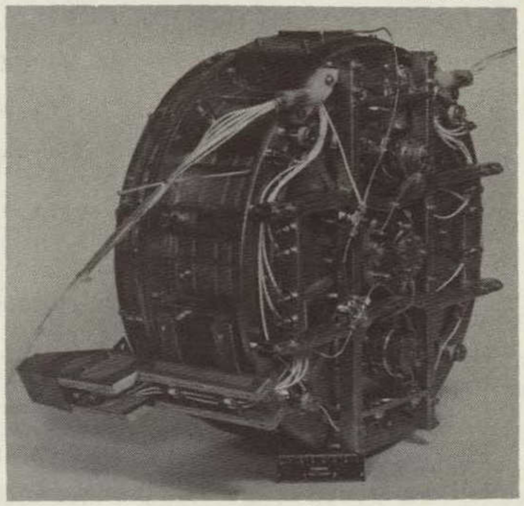
The ion thruster design adopted for this study is based on the 30-cm mercury ion thruster shown in Figure 8 (designated the 900-series EMT).[1,2] This thruster is basically a scaled-up version of the 15-cm thruster developed under NASA LeRC direction during the period from 1960 to 1969 and flight tested by NASA LeRC with SERT II. The 900-series EMT has evolved through an extensive development and testing program, which followed the SERT II program. The 30-cm thruster is designed to operate at a nominal power level of 2.5 kW, a thrust level of 128 mN, and a specific impulse of 3000 sec (1100 V beam voltage). The EMT technology base includes extensive documentation of thruster performance characteristics and critical component properties.[3-7]

The Halley's comet mission would require a somewhat higher specific impulse and thruster power level than provided by the baseline 900-series EMT design to reduce the mass of the propellant and thrust system. Although we did not attempt to optimize the mission or the trajectory, M10535



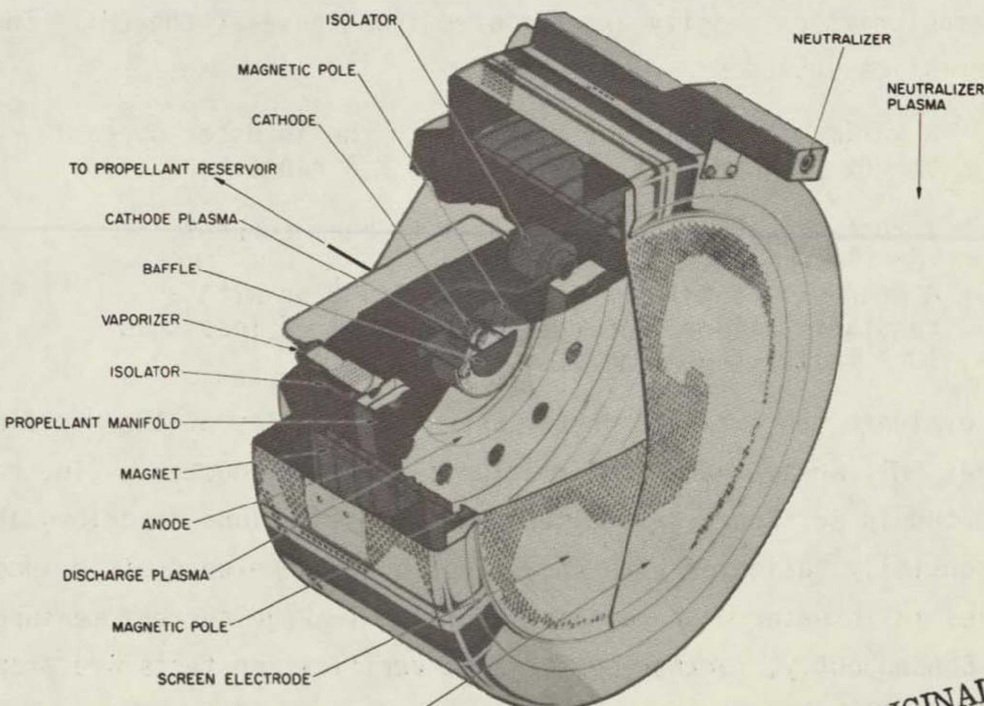
(a) FRONT VIEW

M10536



(b) BACK VIEW WITH COVER REMOVED

M10778



(c) CUT-AWAY ILLUSTRATION

Figure 8. 900-series 30-cm EMT.

a zero-order tradeoff study implied that a nominal thruster power of 6.4 kW at a specific impulse of 4,770 sec would be required. These specifications could be obtained by operating the thruster at 2 A beam current and 3,000 V beam voltage. Thus, the only performance extension required over that of the EMT design would be an increase in the beam voltage. As a first approximation, only two modifications to the 900-series EMT design would be required. First, the propellant electrical isolators, which are rated at approximately 1,500 V, would need to be replaced by equivalent components rated at an adequate margin above the 3,000 V beam voltage requirement. Second, the beam forming assembly, or ion optics, would need to be adjusted to extract and focus the 2 A ion beam at 3,000 V without interelectrode breakdown (arcing). This adjustment would be simply an increase in the spacing between the beam-forming electrodes.[8] For thrust system design and analysis, it was assumed that these adjustments to the 900-series EMT design could be successfully incorporated and that the resultant extended-performance thruster would have characteristics readily extrapolated from those of the EMT. These characteristics included:

- A maximum beam current of 2 A with the thruster operable at any beam current in the 1 A to 2 A range.
- Thruster wear rate proportional to beam current.
- A wearout lifetime greater than 15,000 hr with a constant failure rate (before wearout) of less than 10^{-5} failures per hour.

To evaluate the extended performance capabilities of the modified 900-series EMT, an approach confirmation task was conducted. (The results are reported in Section 6.) The performance assumptions described above were essentially validated with the exception of some unresolved inconsistencies in thruster wear rate and isolator leakage current measurements. Consequently, further performance verification tests are proposed in the initial phases of the Halley's comet program (discussed in Section 4).

The thruster operational parameters specified in the preceding section (6.4 kW maximum power and 3 kV constant beam voltage) were arrived at by a process of selection from among several alternative operating conditions; the results of this process are summarized in Table 6. This selection is tentative, pending further trajectory analysis and iteration with thrust system parameters and performance. For example, option D, which results in a lower propellant weight and a reduced thruster life requirement, may be preferred if the high specific impulse is acceptable, although an engineering trade analysis would still be required to determine the impact of the higher beam voltage required.

The gimbal mechanism required for this application (as specified by NASA[9]) is shown in Figure 9. Two linear actuators provide two axes of angular motion, with a range of angular adjustment of $\pm 5^{o}$ about the Z-axis and $\pm 35^{o}$ about the Y-axis indicated. The gimbal system can be readily integrated with the thrusters, although some additional development is required. The thruster/gimbal system must be subjected to a flight qualification program, which is included in the proposed program plan in Section 4.

2. Power Management and Control

The baseline PMaC subsystem, comprised of interface module units and five thrust module units (two thrusters per module), is described in block diagram form in Figure 10. The PMaC system is designed to operate within the specified solar array voltage range of 200 to 400 V from the main panel and 100 to 200 V from the auxiliary panel. It provides the required voltage and current inputs to operate the thrusters, supplies power for the thrust system housekeeping functions, and furnishes the required mission module power: 400 W during the thrust phase and 650 W after the thrust phase. The design also provides for the required isolation, filtering, interaction protection for EMI control, fault protection, and automatic recovery from thruster malfunctions. These recovery modes and the power management of normal thruster operations are directed by the PMaC controller in the interface module.

Table 6. Thruster Operational Parameters versus Design Options

Parameter	Option			
	A	B	C	D
Maximum number of thrusters operating simultaneously	8	7	7	7
Beam voltage (constant during mission), kV	2.7	2.9	3.0	3.3
Average beam current, A	1.80	1.83	1.83	1.78
Maximum thruster power, kW	6.0	6.3	6.4	7.1
Average thruster power, kW	5.3	5.7	5.9	6.3
Specific impulse, sec	4520	4690	4770	4980
Average thruster efficiency, %	75.4	76.0	76.2	76.3
Total Hg propellant required, kg	2025	1830	1810	1660
Operating time per thruster, hr				
10 operational, no spares	13,870	12,360	12,200	11,475
9 operational, 1 spare	15,410	13,733	13,600	12,750
8 operational, 2 spares	17,340	15,450	15,250	14,343

Selected Baseline

Option C with 9 operational thrusters and 1 spare

Selection Criteria

Reliability, Hg weight, power/voltage, I_{sp}, efficiency

Selection Rationale

- Option A rejected: poor reliability, large Hg weight
- Option D rejected: high voltage, I_{sp} probably too high
- Option C preferred to option B: higher reliability and efficiency
- 9 operational and 1 spare preferred for better system reliability

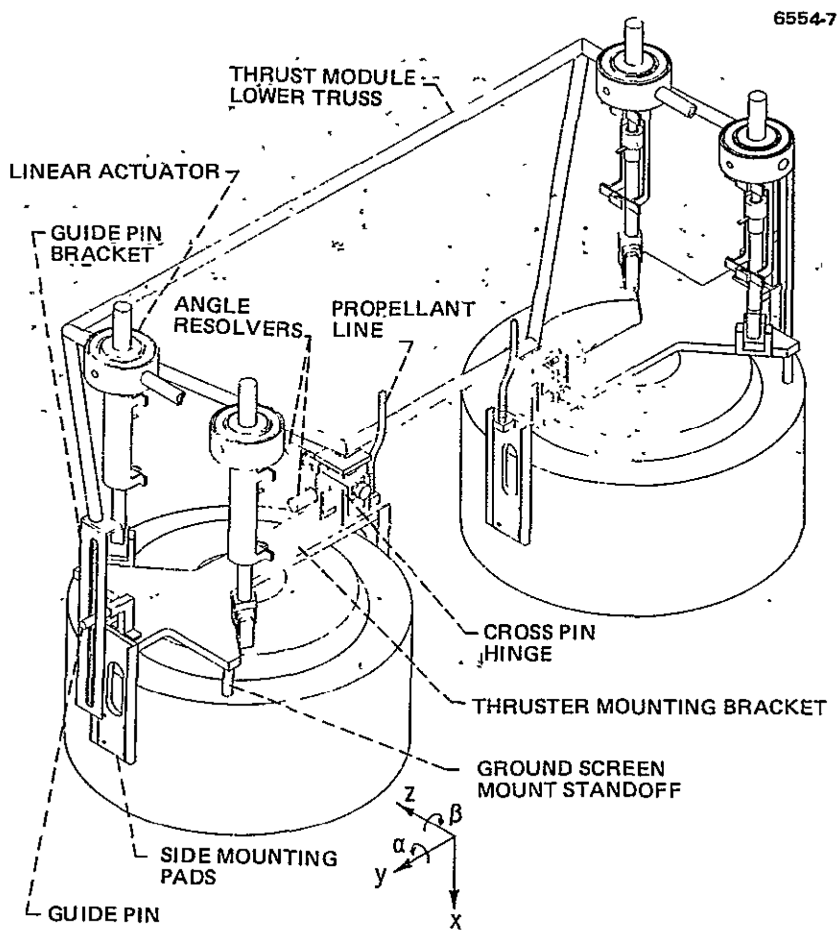
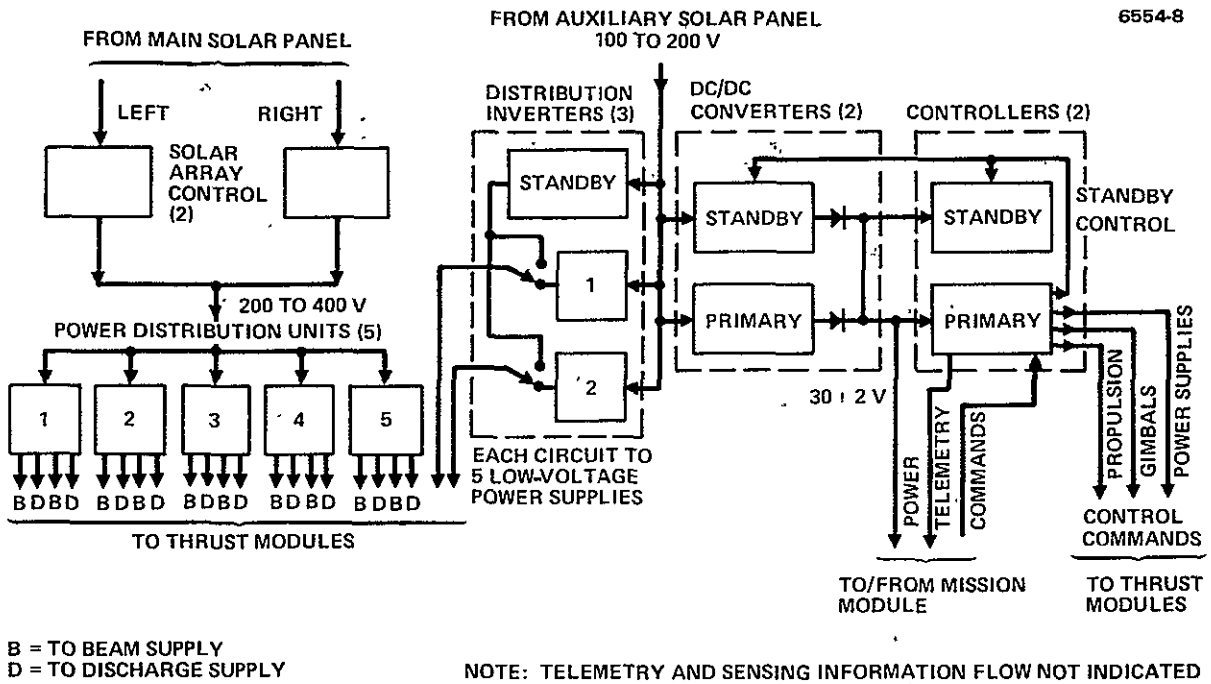
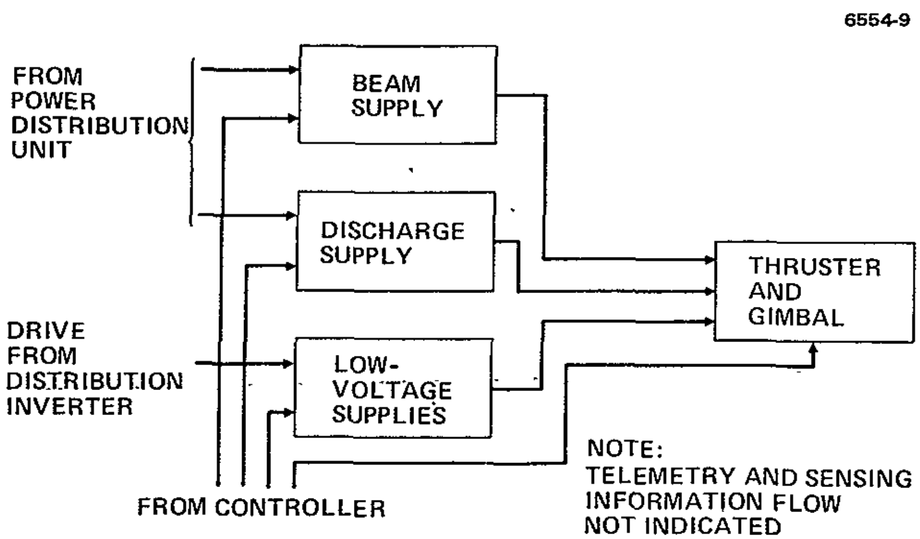


Figure 9. Gimbal system.



(a) Interface module.



(b) Half of thrust module (one thruster).

Figure 10. PMaC block diagram.

ORIGINAL PAGE IS OF POOR QUALITY

Table 7 lists the PMaC components, indicating their size, mass, power consumption, and redundancy. The redundancy was selected to ensure reasonable system reliability (consistent with system mass considerations). The resulting effective functional reliability is estimated to be 0.955 for the interface module PMaC subsystem, and 0.930 for the half-module PMaC equipment (per thruster).

The thrust module PMaC subsystem (one per thruster) consists of three modules — beam supply, discharge supply, and low-voltage power supply — in a common package. This package utilizes the NASA LeRC Z-frame packaging technique being pursued for the 3-kW power processor. The overall dimensions of this package are 1.02 m x 0.38 m x 0.15 m (40 in. x 15 in. x 6 in.). The beam supply module, which incorporates the accelerator supply, has an efficiency of 94%. Its efficiency is assumed to be constant over the operating range. The power supplies utilize the current-controlled series-resonant power-inversion circuit approach to the conventional power-processor circuitry currently under development for NASA LeRC for the 3-kW power level. All units feature fault protection at the input power bus, and output power bus protection from thruster arcing. The discharge supply provides thruster startup heater power. The principal current-voltage (I-V) characteristics of the individual supplies, as required for thruster operation, are summarized in Table 8.

Interface module PMaC units include solar array control and power-distribution units for channeling the power from the main solar panel to the 10 beam/discharge supplies, distribution inverters for driving the 10 sets of low-voltage supplies from the auxiliary solar panel, and dc/dc converters to transform the auxiliary panel power to meet the mission module, housekeeping, and controller power requirements. Transient current requirements are met by the batteries furnished in the mission module to support the 30 V bus.

Power-distribution units provide circuit isolation and contain individual filters for electromagnetic interference (EMI) control. Solar array control units — one for each array wing — ensure that voltage input

Table 7. PMaC Units

Unit	Unit Size, m (in.)	Unit Weight, kg	Unit Power Dissipation, W	Number of Units	
				Active	Standby
Interface module					
Power distribution	0.102 x 0.127 x 0.304 (4 x 5 x 12)	17.3	66	5	0
Solar array control	0.102 x 0.203 x 0.304 (4 x 8 x 12)	5.0	0	2	0
Distribution inverter	0.076 x 0.152 x 0.076 (3 x 6 x 3)	1.0	30	2	1
DC/DC converter	0.102 x 0.152 x 0.152 (4 x 6 x 6)	1.7	73	1	1
Controller	0.102 x 0.203 x 0.304 (4 x 8 x 12)	4.0	15	1	1
Thrust module				Per Module	
Beam supply	0.152 x 0.381 x 0.487 (6 x 15 x 19.2)	200	390	2	0
Discharge supply	0.152 x 0.381 x 0.274 (6 x 15 x 10.8)	5.0	52	2	0
Low-power supplies	0.152 x 0.381 x 0.127 (6 x 15 x 5)	6.3	26	2	0

Table 8. Thruster Power Supply Requirements[a]

Power Supplies	Maximum Voltage, V	Maximum Current, A
Beam supply		
Screen	3000	2.0
Accelerator	-500	0.02
Discharge supply	60	16.3
Low-voltage supplies		
Main and cathode vaporizer	9	1.5
Isolator heaters (startup only)	9	4.0
Neutralizer and cathode heater (startup only)	15	4.4
Neutralizer vaporizer	6	1.5
Neutralizer keeper	25	2.5
Cathode keeper	15	1.0
Magnetic baffle	2	5.0

[a]Power supply capacities, not operating points.

to the PMaC unit is maintained within the specified 2:1 ratio* over the mission; alternately, solar array tilting might be necessary.

The two operational distribution inverters — one for each set of five low-voltage power supplies — and the operational dc/dc converter have the transistor-bridge design with series dc regulators. Input filters are provided on the power bus, and fault protection is achieved with pulse-width modulated switches and series regulators driven by logic-sensitive to overcurrent conditions. Output transformers of the distribution inverter are located in the low-voltage power supplies. When a failure occurs, the redundant unit — one distribution inverter and one dc/dc converter — is automatically switched in either by the controller or by a self-sensing switching circuit.

The conceptual design of the controller is shown in Figure 11. To make the design definitive would require more exact knowledge of mission module design and interface requirements. In addition to providing for commands and management of normal thruster operations within pre-programmed parameter limits, the controller may also automatically respond to certain recoverable thruster malfunction modes. Some of these modes, and the corrective actions required, are shown in Table 9. By using its stored data base and pre-programmed logic, the controller can identify and categorize these modes by comparing measured power-supply parameters to a pre-stored pattern and analyzing the deviations; the controller then automatically initiates the required corrective command sequence.

Interface module units are packaged to facilitate equal power dissipation and weight among modules. The units are structurally mounted on the cold plates (as discussed below) to assure thermal and structural module similarity, thereby preserving thrust system modularity. Thermal control maintains mounting surface temperatures for all units below $50^{o}C$, and individual unit design ensures a thermal load less than 2 $W/in.^2$

*The expected voltage swing of the baseline array may be as large as 2.6:1.

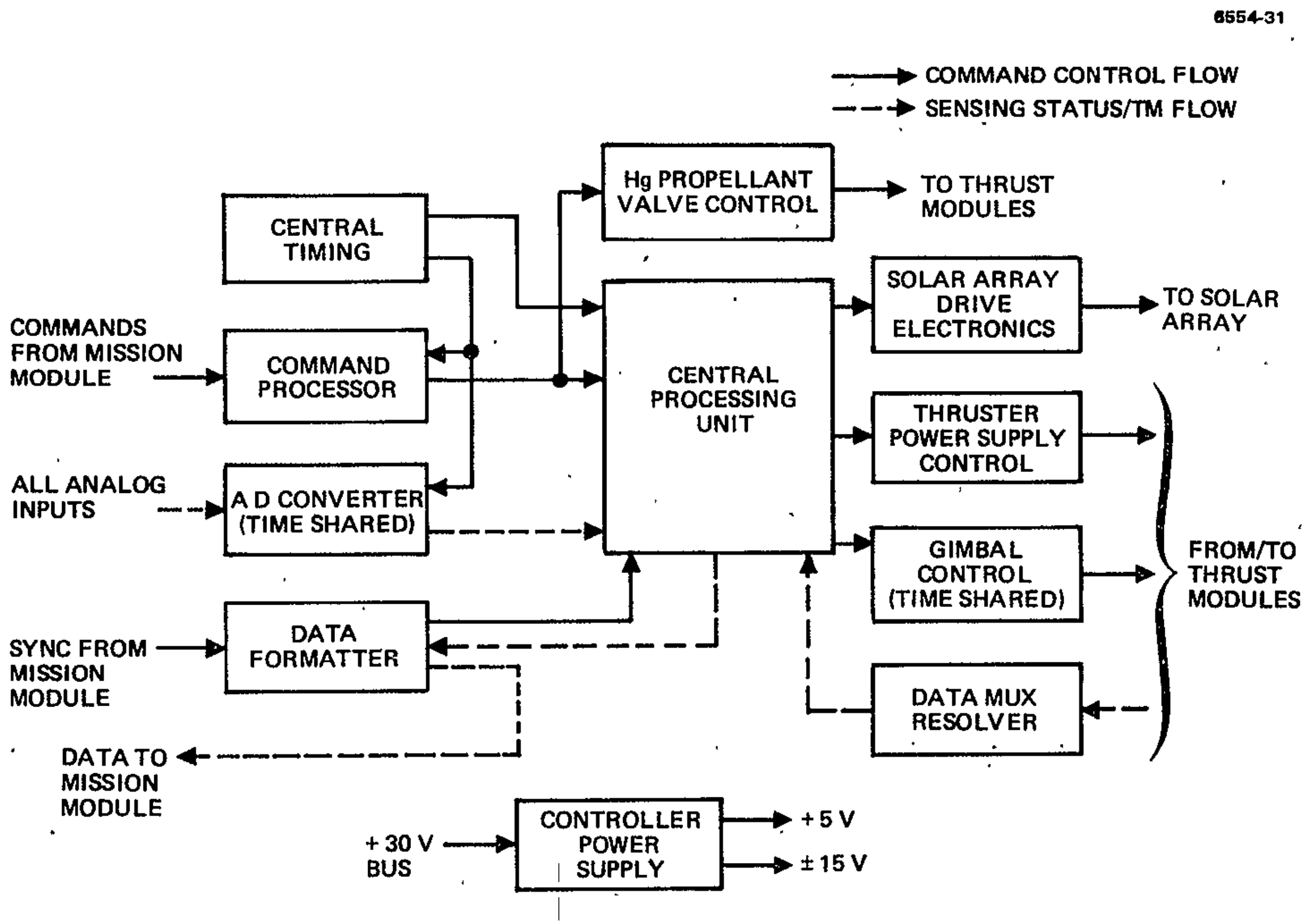


Figure 11. Controller block diagram.

Table 9. Recoverable Thruster Malfunction Modes

Malfunction	Manifestation (Parameter Deviation)	Cause/Source of Malfunction	Remedial Action Required
Screen (beam) overcurrent	$I_{Screen} > 3.5$ A for 0.5 sec[a] $I_{Acc} > 0.2$ A for 1 sec[a] $I_{Acc} > 0.4$ A for 0.1 sec[a]	Momentary high plasma density between extraction grids	Disconnect high voltage Reduce I_{Dischg} Restore high voltage
Discharge shifts to low mode operation	Low cathode Hg flow rate High main Hg flow rate High I_{Acc} I_{Screen} 10% below set point	Excess Hg in discharge chamber	Shut down main vaporizer until: Cathode vaporizer power reaches normal I_{Acc} reaches 0.3% of I_{Beam}
Screen accelerator breakdown	I_{Acc} repeatedly exceeds 0.4 A for 0.1 sec	Metallic flakes between grids (conductive path)	Activate grid clearing circuits (remove conductive path)
Isolator contamination	I_{Screen} repeatedly exceeds 3.5 A for 0.5 sec	Coated isolators[a] Foreign material between isolator shields[a] Liquid penetration[a]	Operate thruster with isolator heater and with discharge power

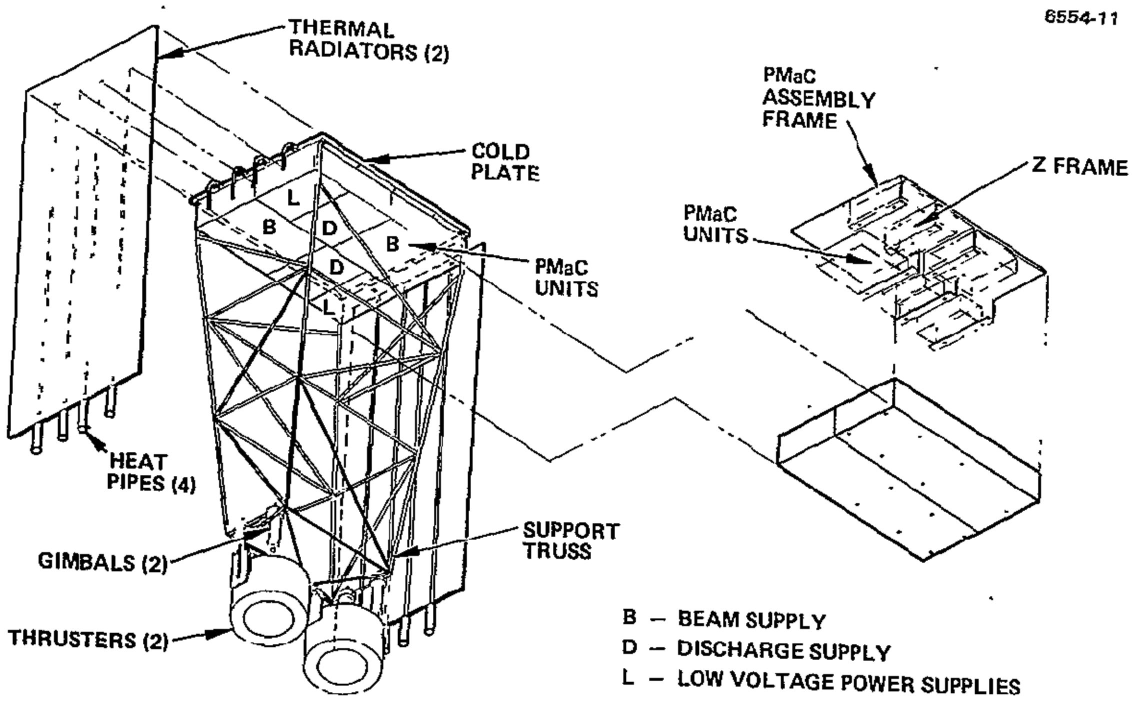
[a]Any one of these.

3. Structural Design

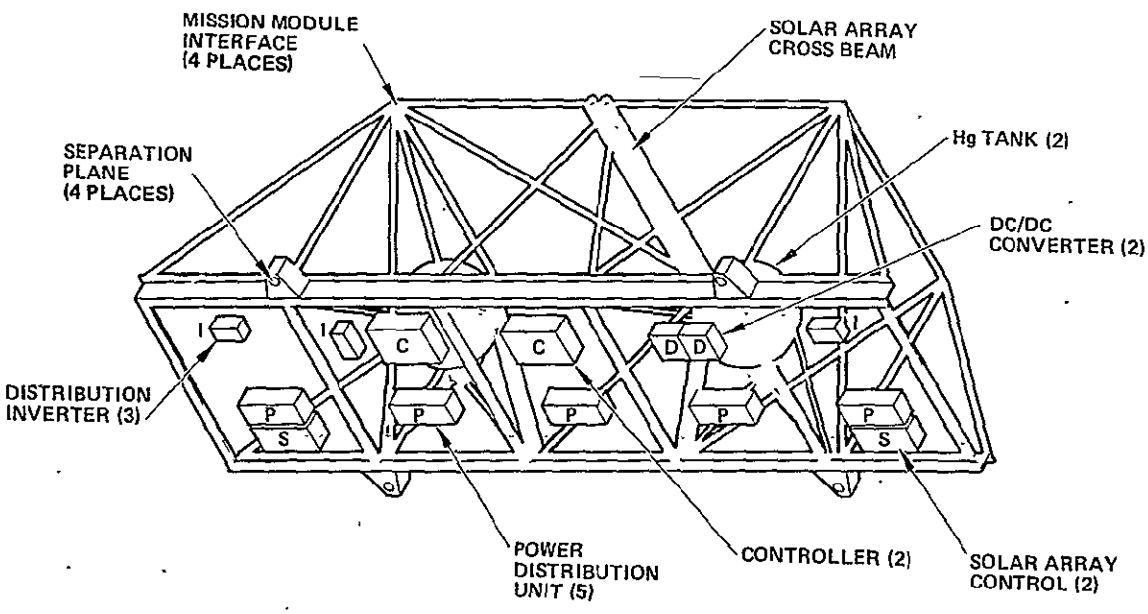
The structural configuration of the thrust system is shown in Figure 12; the key elements (and their assembly sequence) are given for both the thrust module and the interface module. Figure 13 shows design details in the three-view layout of the full configuration, which includes the stowed array and the adapter. The design was evolved from the following considerations:

- Accommodation of solar array stowage and deployment requirements
- Compliance with IUS and shuttle load requirements and constraints
- Minimization of in-orbit weight
- Provision for thrust system modularity
- Provision of a viable interface with the mission module and with the IUS
- Ease of assembly and accessibility.

Adapter design and structural sizing were governed by IUS loads, which dominate design requirements. The adapter is comprised of four beryllium tripods. This configuration, mandated by the large volume occupied by the stowed array, provides the requisite rigidity and strength without an excessive weight penalty. Minimum IUS interface impact was accomplished by utilizing the IUS ring, although it was necessary to add cross beams (as shown in Figure 13). The adapter tripods attach to the interface module at the four points shown. IUS separation is effected at these points using concentric, pyrotechnic separation bolts and push-off springs to produce linear separation with minimum tip-off rate. Simple rotation of tripod members occurs simultaneously. The array cannisters are supported directly by the adapter cross beams at four points (not explicitly shown). Separation is completed using push-off springs and pyrotechnic separation fasteners at these four points.



(a) Thrust module (one of five).



(b) Interface module.

Figure 12. Thrust system configuration.

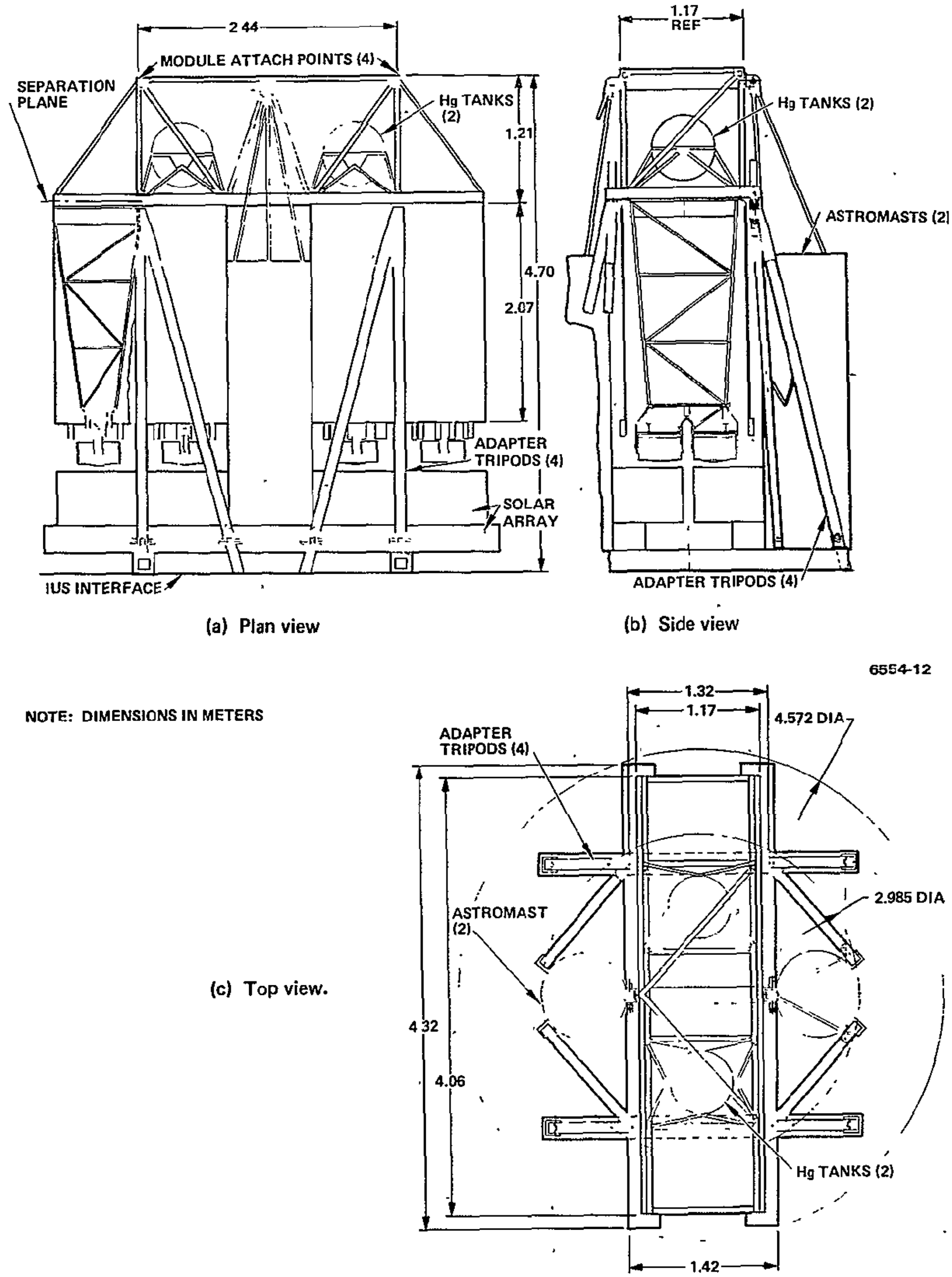


Figure 13. Thrust system layout.

Shuttle load conditions are satisfied with a supplemental forward support cradle (not shown), which attaches the interface module to the shuttle bay.

Thrust modules are virtually identical, thereby providing for design modularity. The only difference between the modules is in the composition of the interface module PMaC units mounted on top of the cold plates. These units, however, are sized and distributed to nearly equalize mass and power dissipation per module. Provision of additional mounting holes on top of the cold plates further permits full module interchangeability. As shown in Figure 12, the design features excellent accessibility to components on individual modules, a simple assembly sequence, and reasonable accessibility to components in the integrated configuration. A relatively simple mission module interface is assured by providing the largest footprint consistent with structural efficiency: four attach points along the circumference of the 1.3 m radius circle, with a 1 m distance between them.

The 4.7 m length of the thrust system was primarily determined by the length of the solar array. Thrust module radiators could be lengthened by 0.4 m without increasing the length of the thrust system. The longest lateral dimension — across the shuttle bay — was determined by the size of the thrust module PMaC package, specified by NASA LeRC to be 1.02 m x 0.38 m (40 in. x 15 in.) per thruster supply. The shape of this package, mounted in the Z-frame structure used for the conventional PMaC design, is shown in Figure 12. The 0.76 m (30 in.) width per thrust module (two PMaC packages) resulted in the 4.3 m width of the five-module configuration; this will just fit into the shuttle bay. The other lateral dimension is determined by the stowed array and the IUS cross-beam supports.

The materials selected for the thrust system structural elements are shown in Table 10. These materials were selected to minimize weight and to satisfy the criteria indicated in the table. The thrust system mass breakdown and dynamic mass properties are summarized in Tables 11 and 12, which show the mass breakdown rounded off to the nearest 5 kg, as well as the location of the center of gravity and the moments of

Table 10. Structural Materials

Structural Element	Material	Selection Consideration
Interface truss		
Lower frame	Aluminum	Strength and manufacturability (forming)
Fittings	Aluminum	Strength and machinability
Tubes	Beryllium	Low mass and high stiffness
Tanks	Stainless steel	Per NASA LeRC specification
Thrust modules		
Cold plate honeycomb core and face sheets	Aluminum	High thermal conductivity
Radiators	Aluminum	High thermal conductivity
Heat pipes	Stainless steel	Proven CTS design
Truss tubes	Titanium	Strength and low thermal conductivity
Fittings	Aluminum	Strength and machinability
Adapter		
Tubes	Beryllium	Low mass and high stiffness
Beams	Beryllium	Low mass and high stiffness
Fittings	Aluminum	Strength and machinability

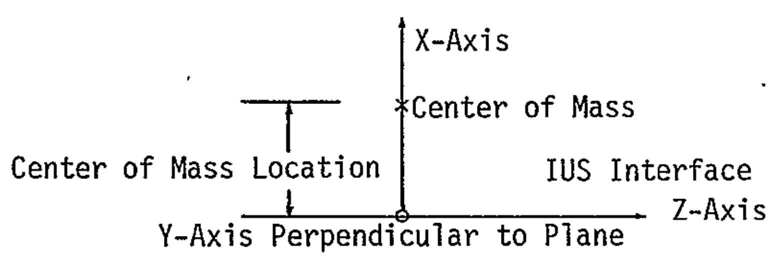
Table 11. Thrust System Mass Summary

Subsystem	Mass, kg	
Thrust modules (5)		620
Thrusters/gimbals	120	
PMaC unit and harness	340	
Thermal control	115	
Propellant lines/valves	10	
Structure and miscellaneous	35	
Interface module		260
PMaC unit and harness	140	
Propellant storage and distribution and residuals	60	
Solar array drive	10	
Structure and miscellaneous	50	
Subtotal		880
15% contingency		130
Thrust system, dry		1010
Hg propellant		1810
Thrust system, wet		2820
Adapter, including contingency		130

Table 12. Thrust System Dynamic Mass Properties

Property	Mission Phase	
	At IUS Separation (Tanks Full)	At End of Thrust Phase (Residuals Only)
Center of mass: location above IUS interface	3.6 m	3.1 m
Moment of inertia about the center of mass		
I_X	2800 kg-m^2	1500 kg-m^2
I_Y	4000 kg-m^2	2200 kg-m^2
I_Z	1100 kg-m^2	500 kg-m^2
Products of inertia about the center of mass	negligible	negligible

Coordinate Reference:



inertia for the two extreme propellant loadings during the mission. Configuration symmetry yields very small cross products of inertia.

4. Mercury Propellant Storage and Distribution

The propellant storage and distribution system, shown schematically in Figure 14, consists of two stainless-steel mercury-storage tanks, stainless-steel feed lines, nitrogen and mercury feed valves, a distribution manifold, solenoid latching valves, field joints, flexible gimbal lines, and tank temperature and pressure transducers. The propellant tank and distribution system design was specified by NASA LeRC.[9] A two-tank configuration was selected, although the baseline design accommodates the single-tank alternative.

The principal advantages of the two-tank configuration are better dynamic load response; more flexibility in selecting the most favorable center-of-gravity (CG) locations; simpler assembly and better accessibility; and approximately a 1 kg lower net mass, primarily because of the lighter interface module truss structure. The single-tank configuration merits further consideration, however, because it would eliminate one problem present with the two-tank configuration. With two tanks, the possibility exists that the tanks may not empty at the same rate, which would impair vehicle balance during flight. Furthermore, the single-tank design is somewhat more reliable because it has fewer parts, although this difference may not significantly affect overall system reliability.

The propellant tank uses a nitrogen gas expulsion technique to supply the propellant to the thruster. The system operates at 276 kPa (40 psi) with the tank full and at 104 kPa (15 psi) at depletion. The design is based on the approach employed for the SERT II spacecraft, with the shape of the bladder support liner modified so that only the volume of the required mercury is supported by the liner.

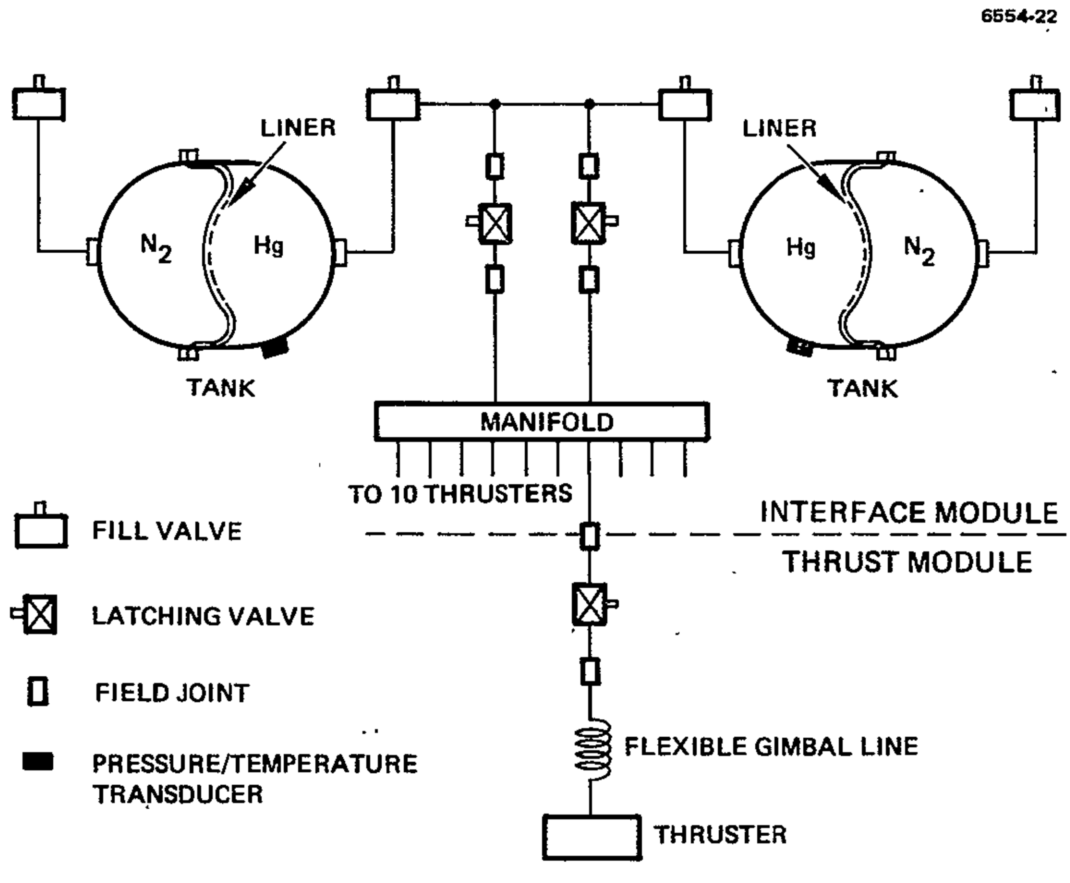


Figure 14. Schematic of the propellant storage and distribution system.

ORIGINAL PAGE IS
OF POOR QUALITY

5. Solar Array Drive

The solar array drive employs the design developed for NASA LeRC;[10] it is shown in Figure 15. The drive system consists of two drive mechanisms and corresponding electronics, incorporated in the PMaC controller. The function of the drives is to rotate the solar array as required during the mission.

6. Thermal Control

The design of the thrust system thermal control subsystem is described in Figures 16 and 17. The design evolved from an extensive tradeoff analysis, based on the mission module and solar array interface data base in Section 3.A.1, to ensure compliance with the thermal requirements of the thrust system (presented in Tables 13 and 14).

Thermal control is provided by the cold-plate/radiator assembly on each module, with two radiators per module; these use the type of variable-conductance heat pipes (VCHPs) used in the Communications Technology Satellite (CTS) type. The VCHPs are embedded in the structure and thermal blankets. Thrust module and interface module PMaC units are mounted on the two sides of the cold plate; they are arranged to approximately equalize heat dissipation per module. The heat pipes are coplanar to permit ground testing. The design is facilitated by the stipulated attitude constraint that the radiators never be illuminated by sunlight.

The principal design parameters are indicated in Figure 17 and in Table 15. Cost and weight considerations led to the nonredundant heat pipe design of four heat pipes per radiator, spaced as indicated. Each heat pipe extends the full length of the cold plate. The design could, however, be readily modified to incorporate additional (redundant) heat pipes, if it is deemed necessary (e.g., as a contingency in the event of heat pipe failure).

The resultant thermal design was subjected to a computer analysis, which also properly accounted for the thrust system "view factors" to the mission module and solar array. The resultant temperature predictions (shown in Table 12 in direct comparison with design requirements) fully demonstrate thermal control design adequacy.

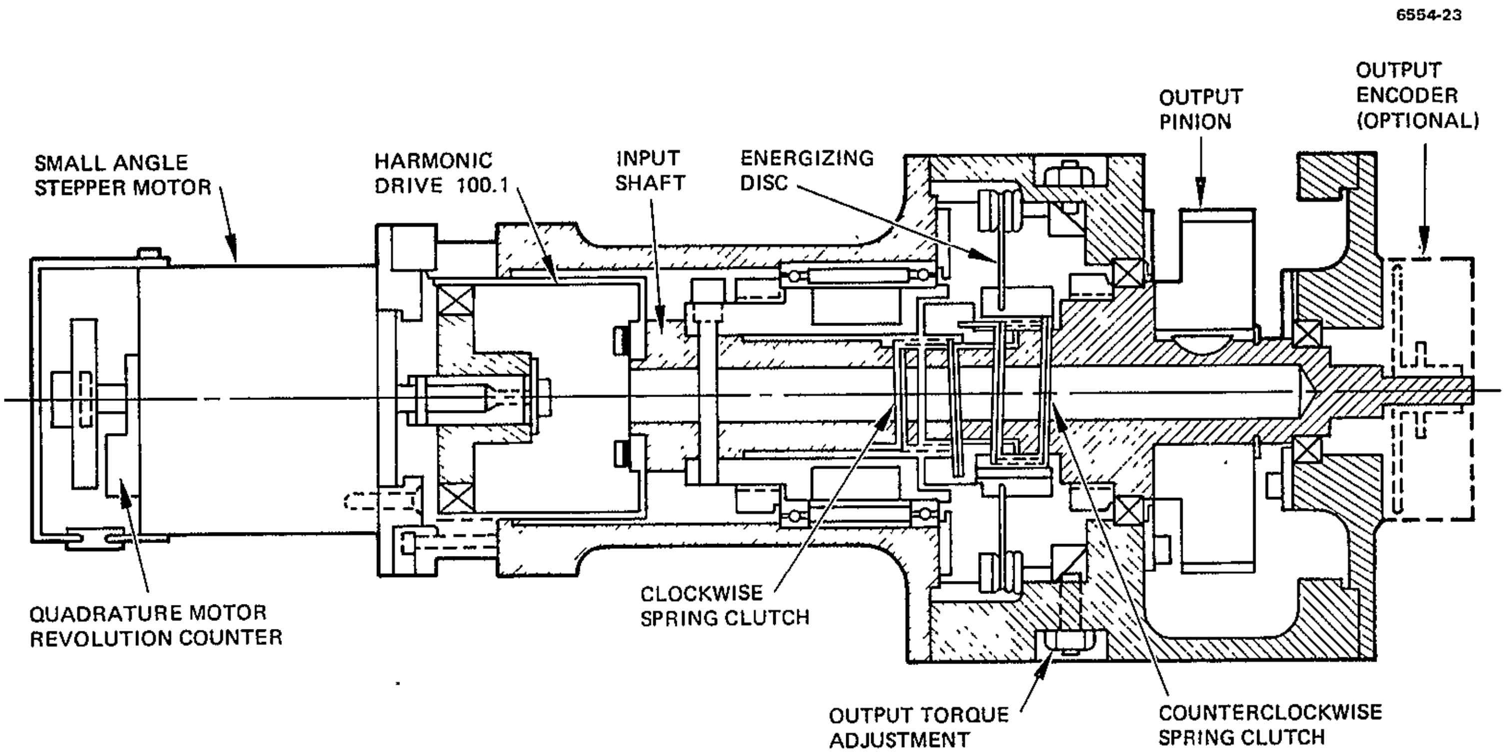


Figure 15. Solar array drive mechanism.

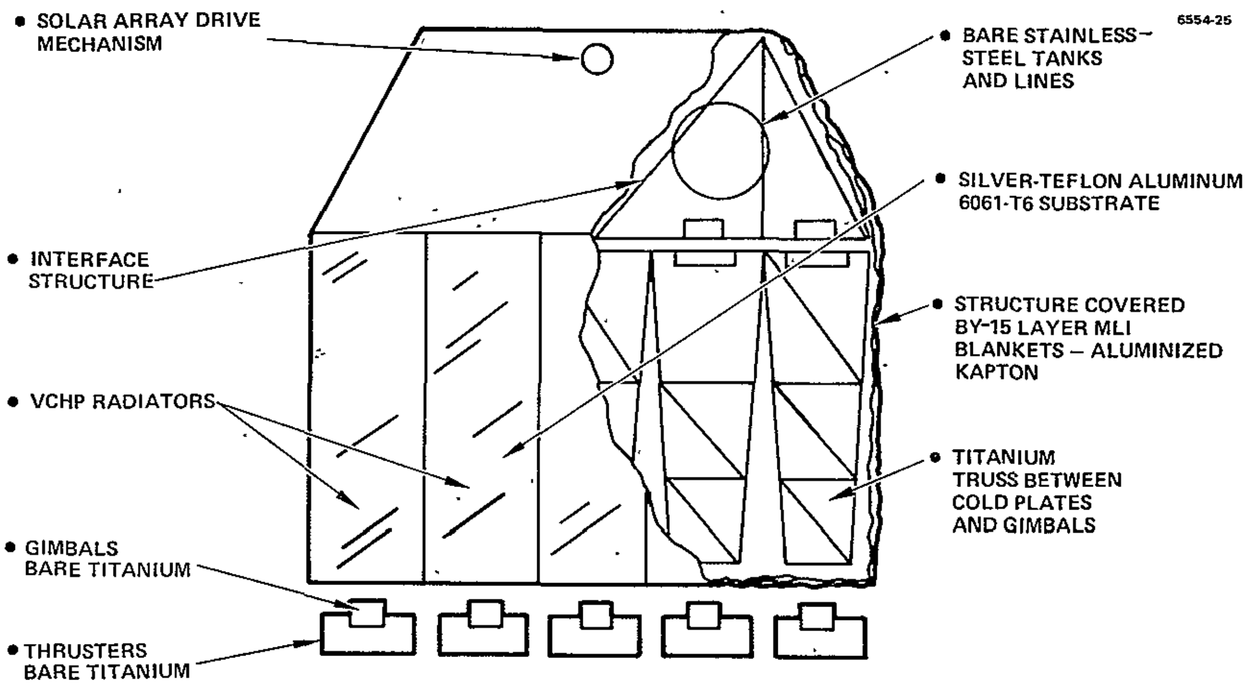


Figure 16. General description of thermal control design.

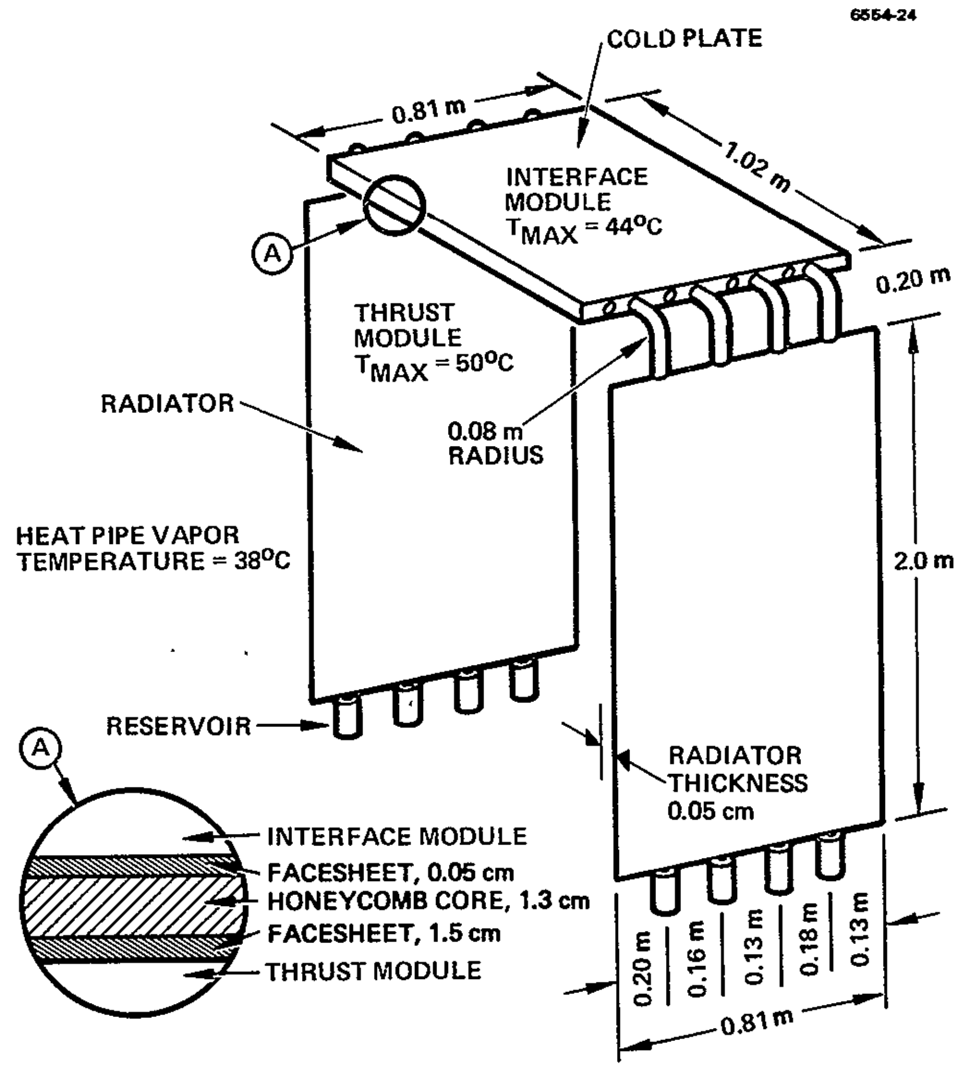


Figure 17. Radiator/ VCHP design description.

Table 13. Thermal Design Criteria and Predicted Performance

Unit or Subsystem	Temperature, °C			
	Minimum Condition (All Thrusters Off at 4.5 AU)		Maximum Condition (All Thrusters On at 1 AU)	
	Allowable Limit	Predicted Value	Allowable Limit	Predicted Value
PMaC mounting surface	- 30	-21	50	49
Propellant tanks	- 40	-31	150	36
Propellant lines	- 40	-15	150	45
Thrusters	-100	-68	300	254
Gimbals	- 65	-57	125	112
Solar array drive	- 30	-20	60	47
Structure	-185	-43	200	65

Table 14. Power Dissipation Breakdown

Units or Subsystem	Power Dissipation, W	
	Maximum	Minimum
Thrusters, each module	750	0
PMaC		
Each module	936	0
Interface module	113	65
Solar array drive	4.5	0

Table 15. Design Characteristics of the Radiator and the Heat Pipes

Radiators
Size, each radiator: 2.0 m x 0.81 m x 0.005 m
Mass, each radiator: 6.2 kg
Total mass, 10 radiators: 62 kg
CTS-Type VCHP
Dynamic range (from full on to full off): 28^{o}C
Leakage per VCHP (full off): 1 W
Mass: Heat pipes 0.258 kg/m
Reservoirs 0.155 kg each
Heat transport (at 50^{o}C):
Capability: 305 W-m (12,000 W-in.) (max)
Design requirement: 269 W-m (10,600 W-in.) (max)
216 W-m (8,500 W-in.) (average)

ORIGINAL PAGE IS
OF POOR QUALITY

SECTION 4

PROGRAM PLAN

A. GENERAL CONSIDERATIONS

The program plan was developed with the objective of delivering the spacecraft thrust system soon enough before the stipulated 1 June 1982 launch date to provide adequate time for spacecraft integration and testing. The intent is to minimize FY 1978 funding requirements. Also considered in developing the program plan were the stipulated contract award dates and phases: Phase I, design definition, starting 1 April 1978, and Phase II, system acquisition, starting 1 October 1978.

In structuring the program plan, the fundamental assumption was made that the thrust system (including the adapter) should be developed, designed, fabricated, and delivered as a complete major subsystem. Because of the intrinsic electrical, structural, and thermal interfaces inherent in the development and design of this subsystem, it is not considered technically viable to parcel out the components of this major subsystem for development and delivery by separate organizations for subsequent integration at the spacecraft level. There are several examples of such intrinsic design interfaces that require a single technical focal point if they are to be resolved during the development phase; these include (1) interactions among the thruster, the thrust module PMaC components, and the interface module PMaC components; (2) thermal design that requires full cognizance of all elements of the thrust modules and of the interface module; (3) structural design that cannot be assured or properly tested except at the thrust system level (including adapter); (4) propulsion subsystem design tanks and distribution system that involves both the interface modules and the thrust modules. On the other hand, the interface between the thrust system and the other major elements of the complete spacecraft — solar array and mission module — is comparatively simple, and can be readily implemented by providing the required simulators and mass models. In any event, the management of system interfaces poses a major program challenge (including interfaces with the shuttle and with the IUS).

B. MASTER SCHEDULE — PROGRAM PLAN OVERVIEW

The program plan calls for the delivery of the fully tested flight thrust system on 1 June 1981. Figure 18 presents an overview of the thrust system program plan; key milestones and the development/procurement time spans are shown in Figure 19, the master schedule.

The proposed plan features three sequential (but partially overlapping) activities: development, qualification, and flight hardware procurement. These activities are shown in the simplified flow chart overview in Figure 20. Each activity culminates in major module-level tests followed by system-level tests during the time periods shown in Figure 19. Each activity then results in delivery to the spacecraft of the

- Thrust system electrical model on 1 March 1980 for early spacecraft-level electrical compatibility tests, as required
- Thrust system qualification model on 1 December 1980 as a potential "pathfinder"
- Flight thrust system delivery on 1 May 1981, 13 months before launch.

C. REQUIREMENTS FOR ADVANCED DEVELOPMENT AND PROCUREMENT

Although the key elements of the proposed program plan generally correspond to the stipulated two-phase definition/acquisition program, it will be necessary to begin development and procurement substantially before the scheduled initiation dates for the two phases of the program (1 April 1978 and 1 October 1978). One way these advanced development and procurement activities might be implemented is suggested in Section 4.H. The reason these advanced activities are needed is evident from the development and procurement time spans indicated in Figure 19; the need stems primarily (but not entirely) from the lead time required for the development of PMaC hardware. Specific requirements for advanced development and procurement are shown in more detail in Figure 21. In particular, considering the lead times required, it is deemed mandatory to initiate PMaC design definition no later than September 1977, and to

6554-14

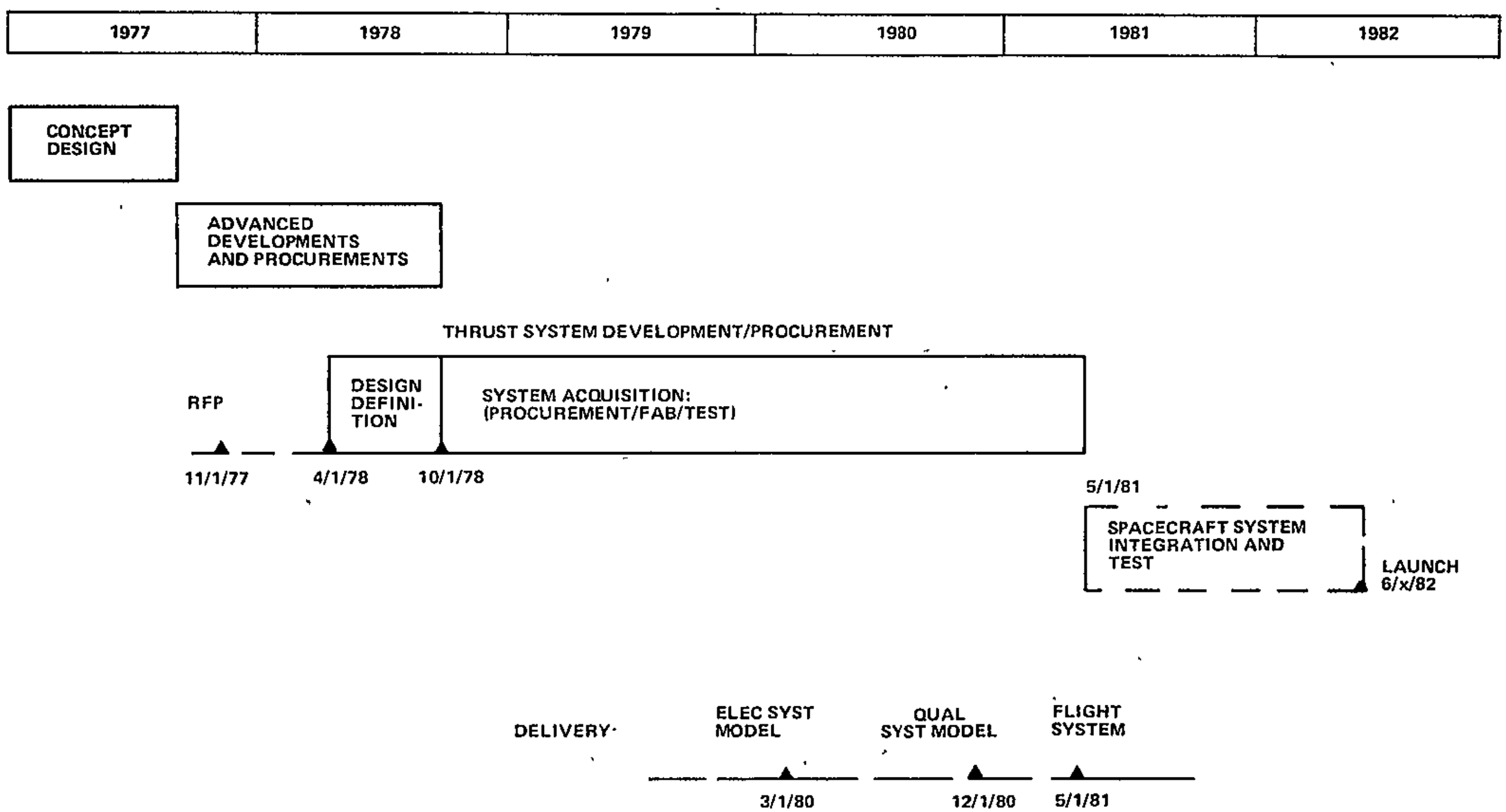


Figure 18. Overview of the thrust system program plan.

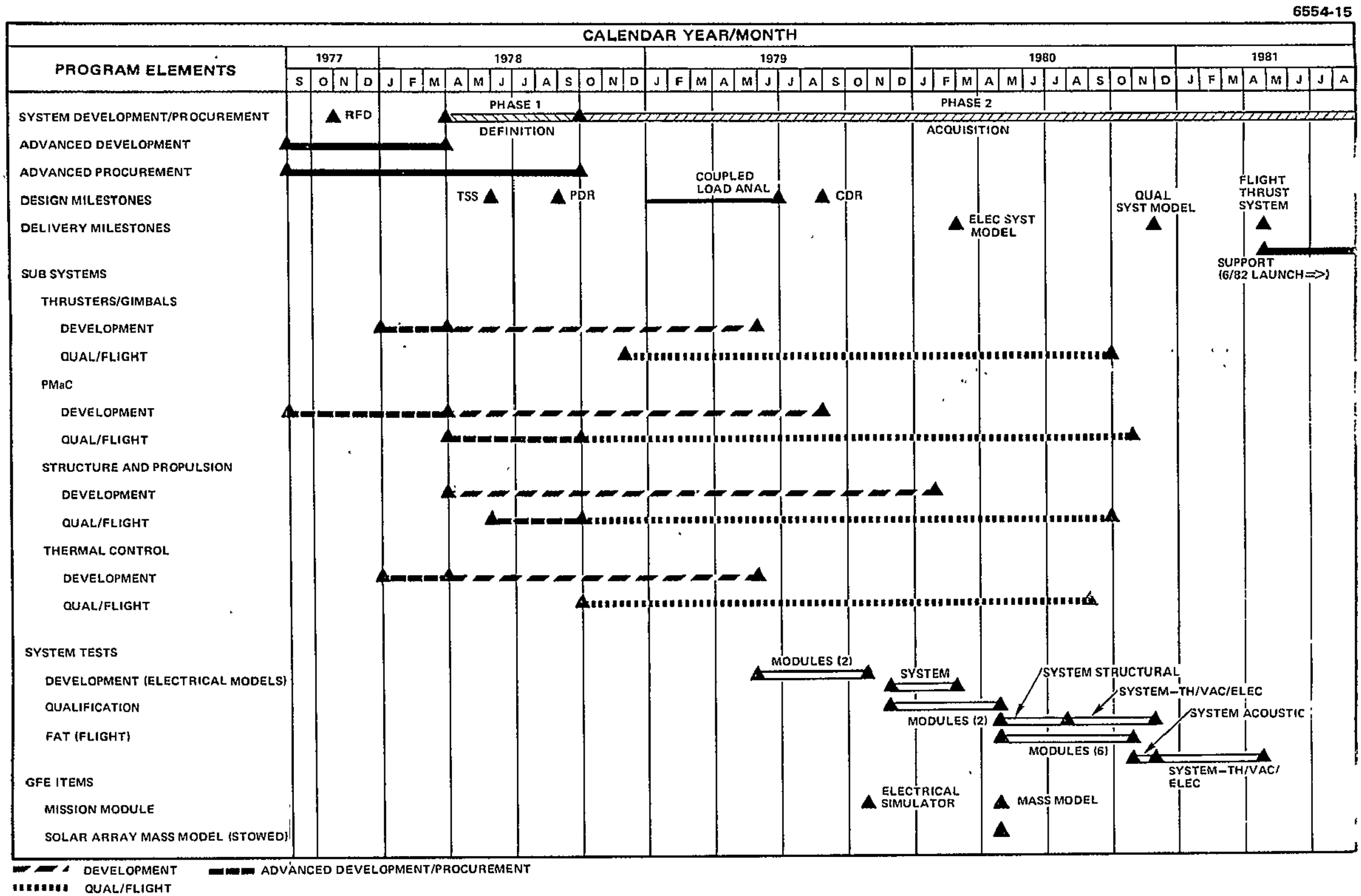
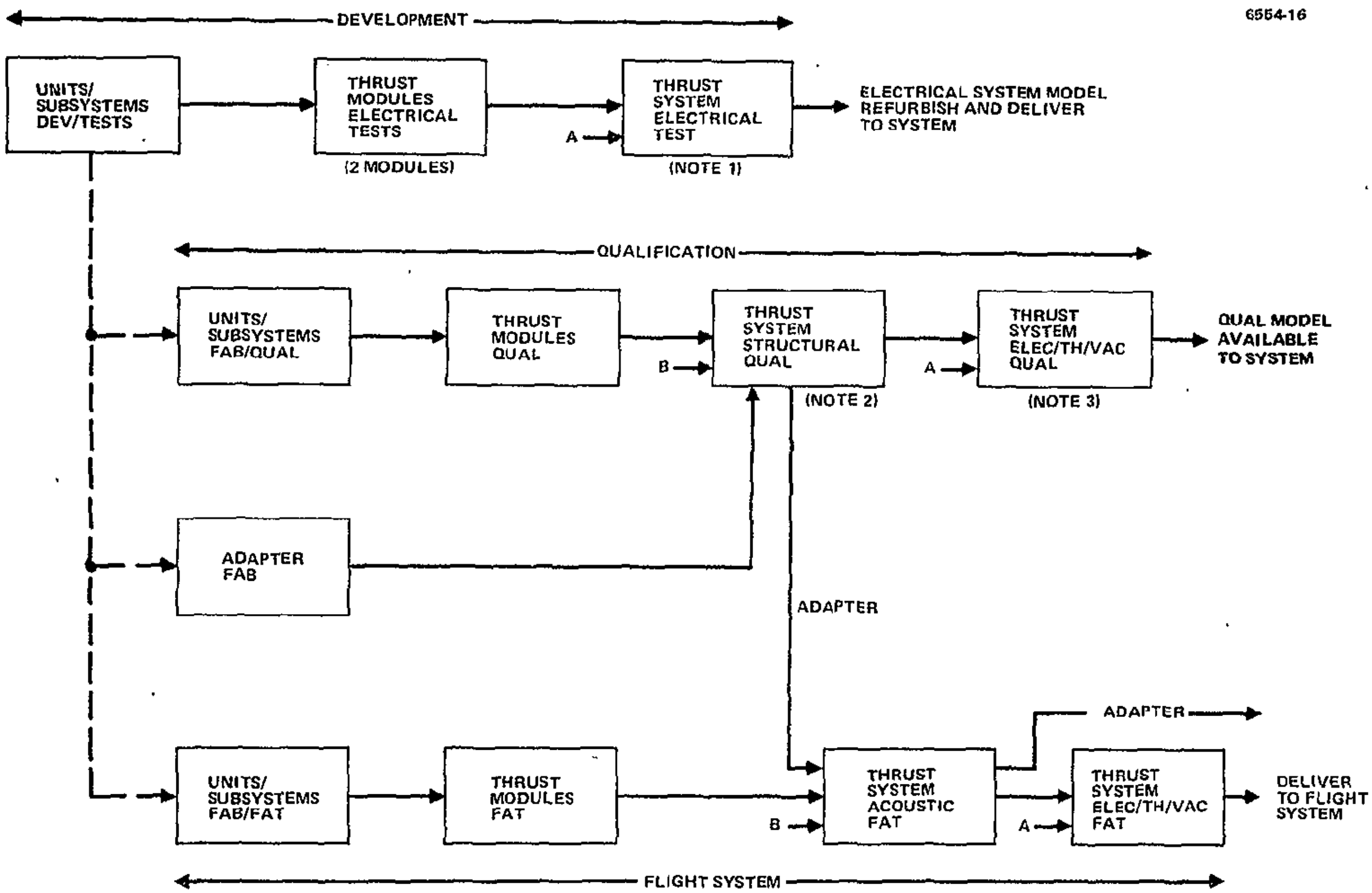


Figure 19. Master schedule for the thrust-system program.



NOTES:
(1) 3 DUMMY MODULES, SINGLE STRING INTERFACE PMaC DEVELOPMENT MODEL UNITS
(2) 3 DUMMY MODULES; DUMMY INTERFACE PMaC FLIGHT QUALITY UNITS (FROM QUAL)
(3) 3 DUMMY MODULES; SINGLE STRING INTERFACE PMaC FLIGHT QUALITY UNITS (FROM QUAL)

A = MISSION MODULE ELECTRICAL SIMULATOR

B = MISSION MODULE MASS MODEL, AND SOLAR ARRAY (STOWED) MASS MODEL

Figure 20. Simplified flow chart of program development, procurement, and testing.

start PMaC hardware procurements without delay. In addition, to meet the delivery date for the heat pipes, it is necessary to begin development of the final specifications by 1 January 1978 and to begin procurement by 1 March 1978. Beryllium delivery lead times require advanced procurement starting 1 January 1978. Figure 21 also shows the proposed immediate initiation of thruster performance verification tests using the modified 900-series thrusters.

D. DEVELOPMENT PROGRAM

The development activity shown in Figure 20 comprises PMaC system and thruster development, and parallel developments of the other major subsystems: thermal control, propellant storage, solar array drive, structure, and adapter. The PMaC-electronics/thruster development program is shown in more detail in the flow chart in Figure 22. A schedule for all the development activities is shown in Figure 23.

The PMaC-electronics/thruster development program features sequential breadboard- and development-model module-level tests, followed by tests at the thrust-system level using a single-string interface module PMaC unit and the mission module electrical simulator. To ensure that major intermodule interactions are explored, two full modules will be fabricated and tested. All developmental model electronics will be flight configured, but use commercial parts. The system is therefore considered not to be flight quality; no module environmental testing is included in this development. Correspondingly, structural thermal, and propulsion subsystems for these configurations are either non-flight or simulated, as required. Thermal control in vacuum chambers is provided by separate means. After the thrust system electrical tests are completed, the thrusters will be replaced by equivalent electrical load simulators for subsequent spacecraft-level electrical compatibility tests (in air), as desired.

Thermal control development is a separate parallel activity that entails the designing, developing, and life testing of heat pipes and the designing and testing of a separate thermal model. Corresponding parallel propellant-subsystem, solar-array-drive, and structure/adapter

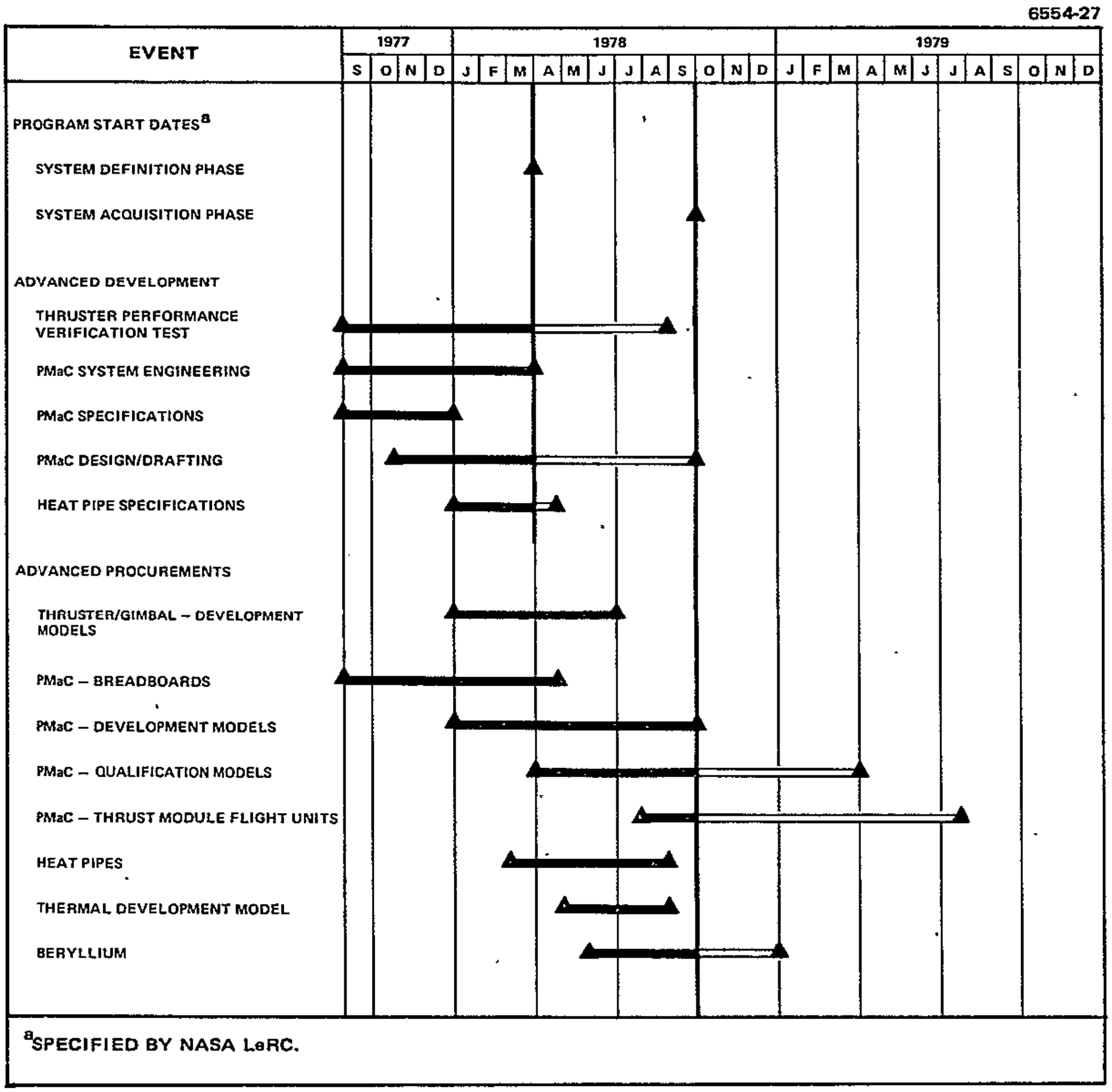


Figure 21. Requirements for advanced development procurement.

ORIGINAL PAGE IS
OF POOR QUALITY

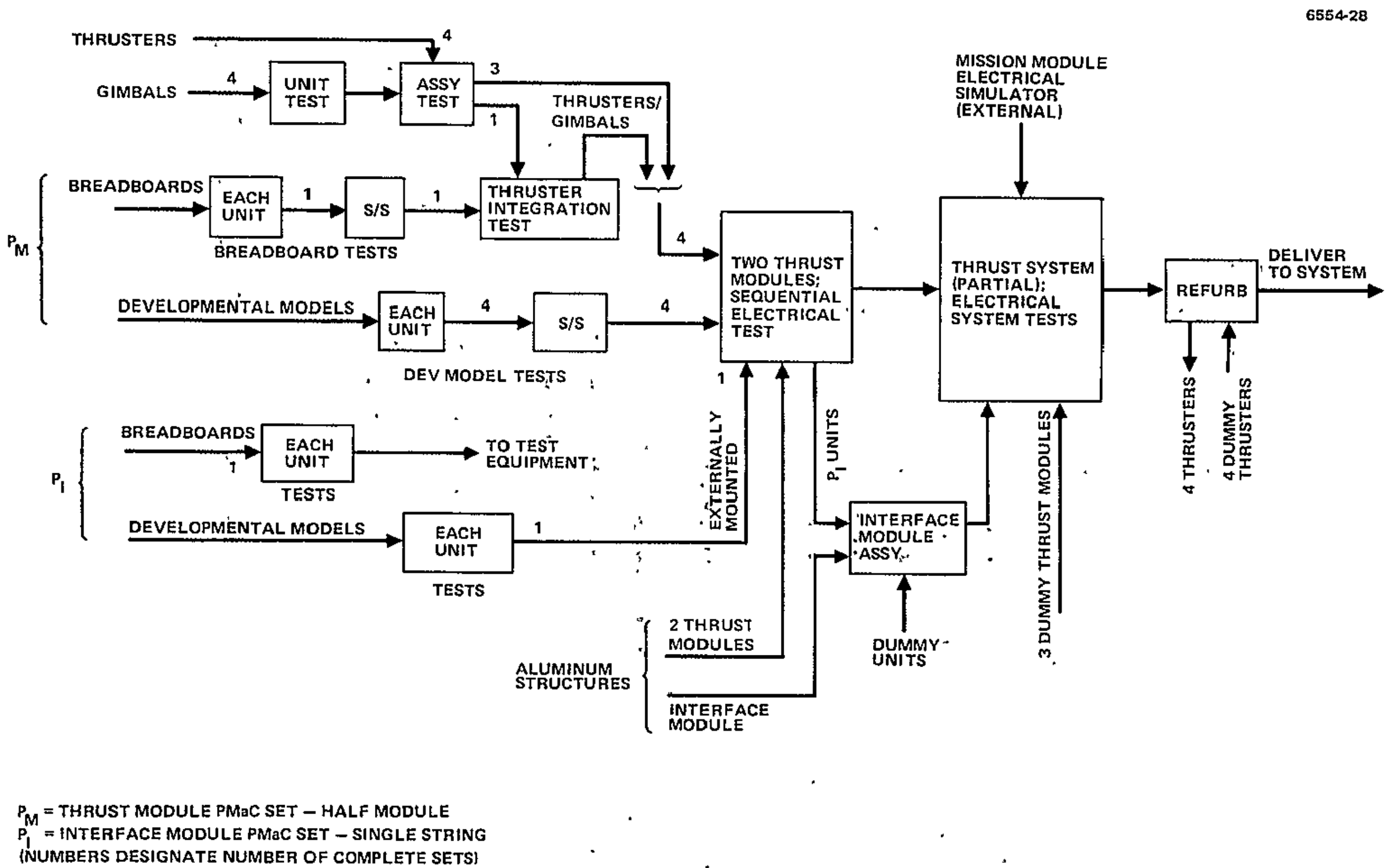
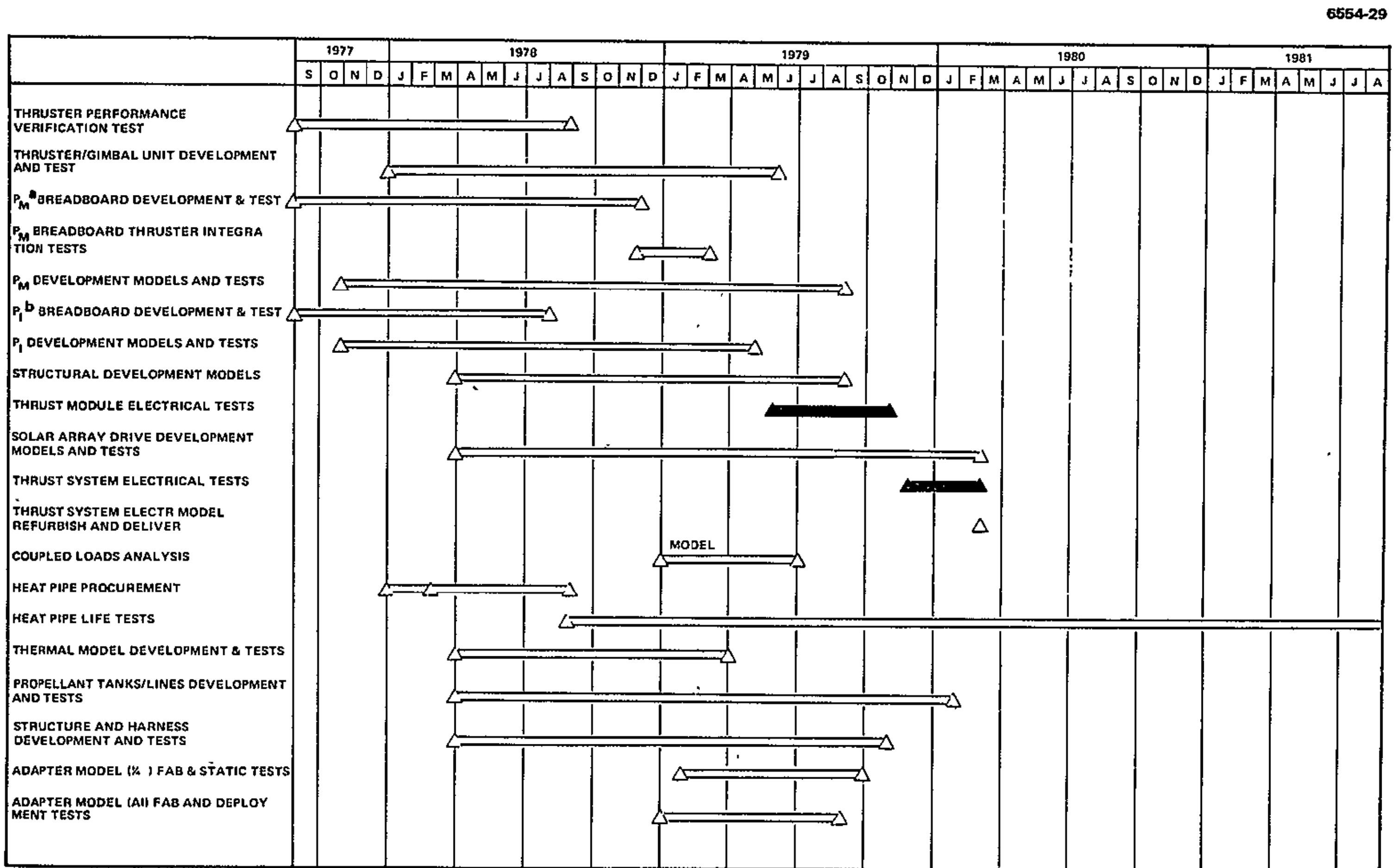


Figure 22. Development flow chart for thrusters/PMaC-electronics subsystem.



[a] P_M = THRUST MODULE PM&C SET HALF MODULE
[b] P_I = INTERFACE MODULE PM&C SET SINGLE STRING

Figure 23. Development program schedule.

development is also indicated. Structural development includes static tests of one adapter tripod, the development of structural math models and coupled-load analysis, and deployment tests using an aluminum adapter model.

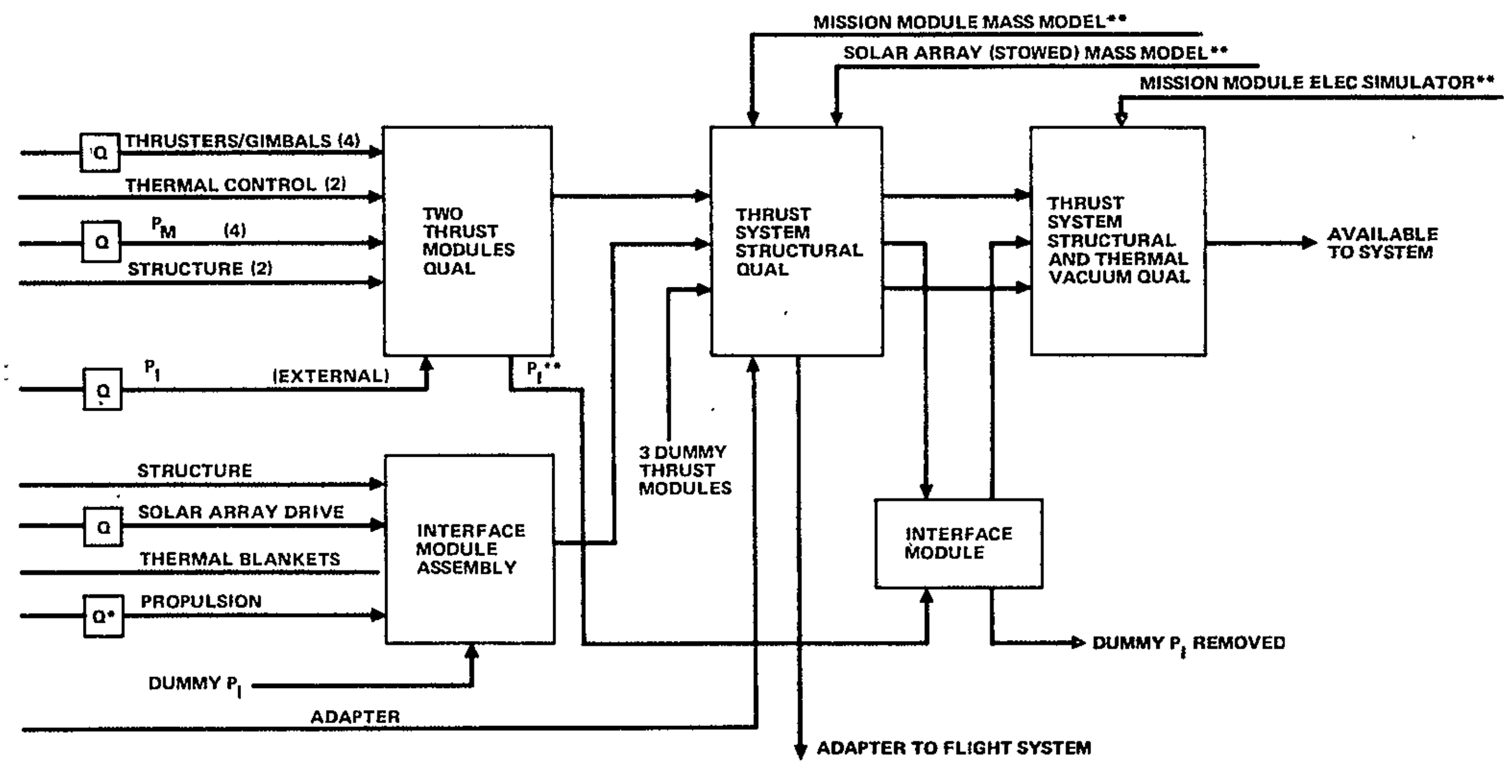
E. QUALIFICATION PROGRAM

A flow chart for the qualification program is shown in Figure 24, with the corresponding schedule shown in Figure 25. The proposed qualification plan features a comprehensive, albeit minimum-cost, program to assure maximum confidence in thrust system electrical and environmental integrity prior to delivery. This plan would greatly reduce the possibility of discovering problems at the spacecraft level; such a late discovery would probably cause a nonrecoverable schedule slippage.

After unit-level qualification of the thrusters and gimbals, electronics, solar array drive, and propellant tanks, two complete thrust modules will be assembled and subjected to complete electrical testing and environmental testing (in vacuum), using externally mounted interface module PMaC electronics. Module-level tests will be used to qualify the thermal subsystem. The subsequent qualification program at the thrust-system level will consist of two distinct tests: a structural qualification test in a vibration facility, and an electrical and thermal vacuum qualification test in a thermal vacuum facility.

The structural qualification test, which serves to validate system structural integrity (including the integrity of the adapter and of the propellant storage and distribution subsystem), will be performed on a simulated full structural assembly that will include the mass models of the mission module and of the stowed solar array. Dummy interface PMaC units and three dummy thrust modules with simulated thermal control will be used to minimize cost; their use will not significantly jeopardize technical integrity. Then, after the mass models and the adapter are removed, and the qualification PMaC interface units are installed, the electrical and the thermal-vacuum tests will be conducted using the mission module electrical simulator.

6554-17



P_M = THRUST MODULE PMaC – HALF MODULE
P_I = INTERFACE MODULE PMaC
Q = UNIT – LEVEL QUAL; Q* · ONE QUAL TANK (SECOND = DEVELOPMENT MODEL)
** = REMOVE AFTER TEST FOR USE IN ACCEPTANCE TEST OF FLIGHT MODULES

Figure 24. Qualification program flow chart.

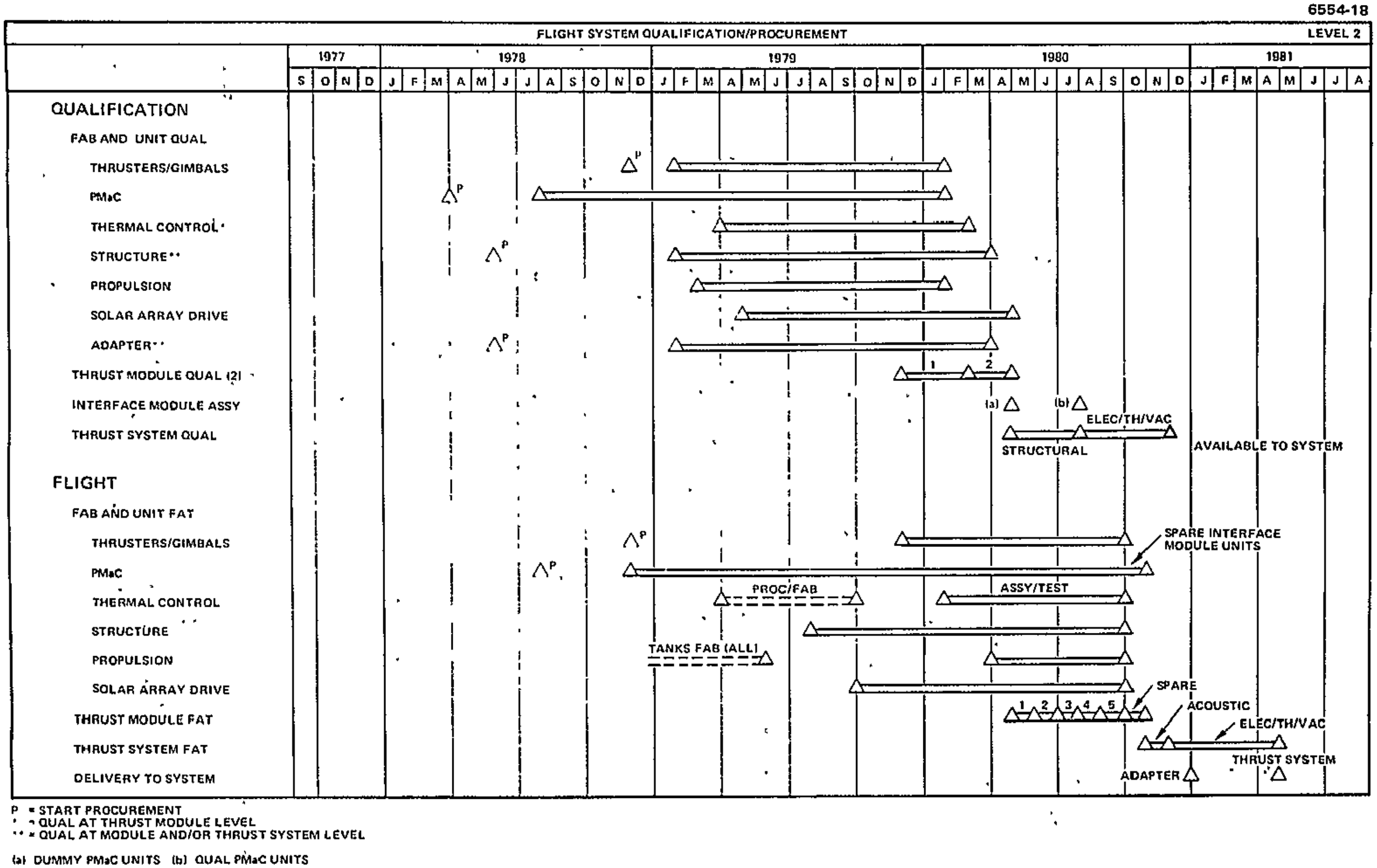


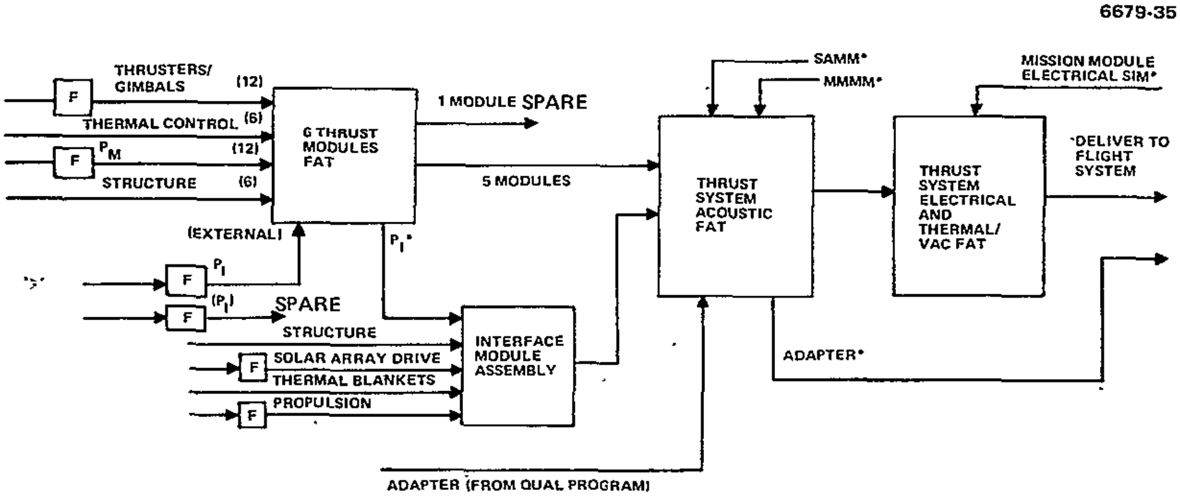
Figure 25. Qualification and flight system schedule.

Using the electronic units in the qualification tests would preclude their being used for flight without first being reconditioned, and schedule considerations do not allow time for such reconditioning. Furthermore, this plan calls for the qualification thrust system to be delivered intact to the spacecraft immediately after the qualification program. Therefore, we propose that a separate set of flight units and flight spares be procured for the flight system. The significant exception to this proposal is the beryllium adapter, which is to be delivered and used in the flight system after the structural qualification program is completed.

F. FLIGHT SYSTEM PROCUREMENT AND TESTING

The procurement, fabrication, assembly, and testing steps for the flight system and flight spares are indicated in the flow diagram in Figure 26 (the corresponding schedule is shown in Figure 25). This procurement and testing program will begin shortly after the qualification program because of schedule pressure, but with a sufficient lag to allow modest changes resulting from the qualification program to be incorporated; major design changes could not be made in the time allotted, however.

The flight acceptance test (FAT) program sequence is similar to the qualification program sequence, except that the test configurations and levels of testing are significantly different. Units and modules will undergo the FAT program at lower levels of environmental exposure. All five modules will be tested; one additional complete module, which will serve as a flight spare, will also be tested. The qualification model interface PMaC electronics will be used to acceptance test these flight modules before the flight model interface PMaC electronics become available. A single-string set of spare interface module PMaC units will also be fabricated and tested. At the thrust system level, the structural FAT program will be conducted on the completely assembled flight configuration (including the adapter and the mission-module and solar-array mass models), but an acoustic environmental exposure is deemed adequate. The subsequent electrical and thermal vacuum testing of the thrust system will essentially be identical to that performed earlier on the qualification thrust system.



F = UNIT – LEVEL FLIGHT ACCEPTANCE TEST
P_M = THRUST MODULE PM&C – HALF MODULE; P_I = INTERFACE MODULE PM&C
SAMM = SOLAR ARRAY MASS MODEL, MMMM = MISSION MODULE MASS MODEL
(P_I) = SINGLE STRING P_I UNITS
* = REMOVE AFTER TEST

Figure 26. Flight system test and integration flow chart.

After the FAT program is completed, the thrust system will be delivered to the spacecraft for integration, testing, and launch. The adapter will be available earlier — after the system acoustic FAT program is completed.

The required units and subsystems are summarized in Table 16. The required types and quantities of the principal units of the thrust system are indicated; these reflect the specific requirements of the program plan. The proposed plan for spare parts is also indicated in Table 16. It includes the assembled unit; module flight spares, and spares planned to be procured at the piece part and subassembly level. Table 16 also shows the dummy models of thrust system components required for the various test configurations, and the postulated GFE simulators and mass models.

G. FACILITIES PLAN

To implement the proposed program plan will require highly specialized vacuum test facilities for the development testing, qualification testing, and flight acceptance testing of the thrust-system components (thruster/PMaC electronics), the thrust-system modules, and the full thrust-system assemblies. The problem is compounded by the schedule-dictated requirement for parallel testing, by the physical size of the thrust system, and by the fact that not all of the potential facilities would be made available for use with mercury. In addition, vibration and acoustic facilities are required for the thrust-system structural-qualification and FATs, respectively.

Facility requirements are further deterrents to performing thrust-system qualification testing at the spacecraft level because it would be difficult to provide the much larger chamber required. There is a readily available chamber for the electrical/thermal vacuum tests of the thrust system alone — the "Tank 6" facility at NASA LeRC.

Many suitable vibration and acoustic facilities are available for thrust system structural tests. The proposed facility plan for electrical tests at the unit, module, and thrust-system levels is shown in Figure 27. Two existing Hughes facilities should readily be able to accommodate the

Table 16. Required Units and Subsystems

Units/Assemblies	Quantities[a] Developmental B	D	Flight Quality Q	F	Spare	Piece Parts/ Subassemblies (Spare)
Thrust system subsystems						
thruster/gimbal	-	4	4	10	2	1 full, plus 4 ea: CIV, MIV, NIV, grid set
PMaC thrust module set (one beam/discharge/LV supply)	1	4	4	10	2	30% extra parts for all units
PMaC interface module	1[b]	1[b]	1	1	1[b]	30% extra parts for all units
Structure thrust module	-	2[c]	2	5	1	Tubes (50% of module)
Structure interface module	-	1[d]	1	1	-	Tubes (50% of module)
Thermal control	0.5[e]	1	2	5	1	30% extra pipes; one extra set all else
Tanks	-	1	2[f]	2	1	
Solar array drive	-	1	2	2	1	
Propulsion lines	-	0.5	1	1	-	One set
Adapter	0.25[g]	1[h]	1→	(1)	-	50' tubes
Dummy						
Thruster (electrical simulation[i]	-	4	-	-	-	
Thrust module (mass model)	-	3[j]	3	-	-	
PMaC interface module (mass model)[k]	-	-	1	-	-	
GFE						
Mission module electrical simulation	-	1 →	(1)	→(1)	-	
Mission module mass model	-	-	1	→(1)	-	
Stowed array mass model	-	-	1	→(1)	-	

[a]B - Breadboards or equivalent development assemblies
D - Development models (nonflight) — e.g., electrical PMaC models
Q - Qualification models (flight quality) — "engineering models"
F - Flight units/assemblies

[b]Denotes single string

[c]Aluminum

[d]Aluminum

[e]Life test (half module)

[f]One to unit qualification burst test (D-tank installed on system qualification)

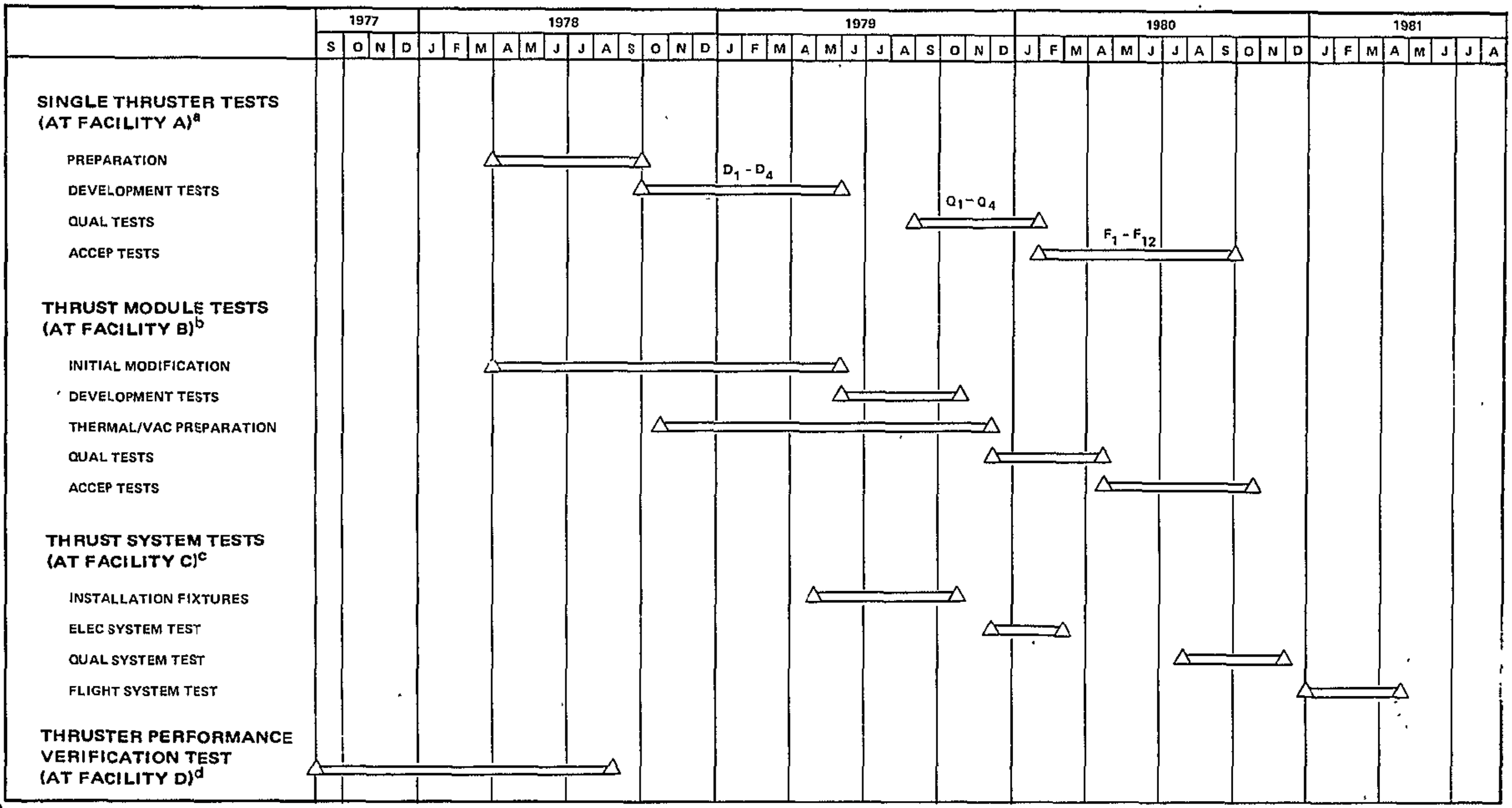
[g]Static (one tripod)

[h]Aluminum (articulation tests)

[i]For electrical system model

[j]Aluminum

[k]Flight simulation



[a] VACUUM FACILITY A – SUITABLE FOR SINGLE THRUSTER TESTS (HRL 5' CHAMBER)
[b] VACUUM FACILITY B – CHAMBER MODIFIED TO ACCOMMODATE MODULE TESTING (HRL 9' CHAMBER)
[c] VACUUM FACILITY C – CHAMBER SUITABLE FOR TESTING COMPLETE SYSTEM (LeRC TANK 6)
[d] VACUUM FACILITY D – VERIFICATION FACILITY WITH FROZEN MERCURY COLLECTOR (HAC C3)

Figure 27. Test facilities plan.

parallel unit-level and module-level tests with only minor modifications. The proposed schedule overlap is sufficient to allow these two facilities to be used efficiently and sequentially. To use the NASA LeRC "Tank 6" facility proposed for tests at the thrust-system level would require only that a suitable mounting adapter be provided. Scheduled phasing would permit the efficient, sequential use of this facility for the thrust system development, qualification, and FAT programs. The fourth facility shown in Figure 27 is currently available at Hughes and is used for laboratory tests of ion thrusters; this facility could be used to conduct the proposed thruster performance verification tests early in the program.

The proposed facilities plan, admittedly predicated on the assumption that the Hughes Aircraft Company will be responsible for thrust system development, is not a unique solution. But it does indicate that at least one solution is available for implementing the proposed program plan.

H. RECOMMENDED THRUST-SYSTEM PROCUREMENT AND MANAGEMENT PLAN

The recommended thrust system procurement and management plan, presented in Figure 28, is consistent with the ground rules in Section 4.A, with the requirements for advanced development and procurement in Section 4.C, and with the other features of the program plan. This figure illustrates that a viable procurement structure is available and makes recommendations regarding the assignment of responsibilities. Admittedly, alternate procurement plans are possible.

The recommended plan for a complete thrust system was developed under the supervision of NASA LeRC within the program schedule, starting with the contract award 1 April 1978. Advanced development and procurement requirements will be met by early, direct funding and management by NASA LeRC; these programs can then be phased at suitable times, as indicated in the program plan, to the responsible thrust system contractor.

We recommend that the prime contractor for the thrust system be directly responsible for the specific areas indicated. This

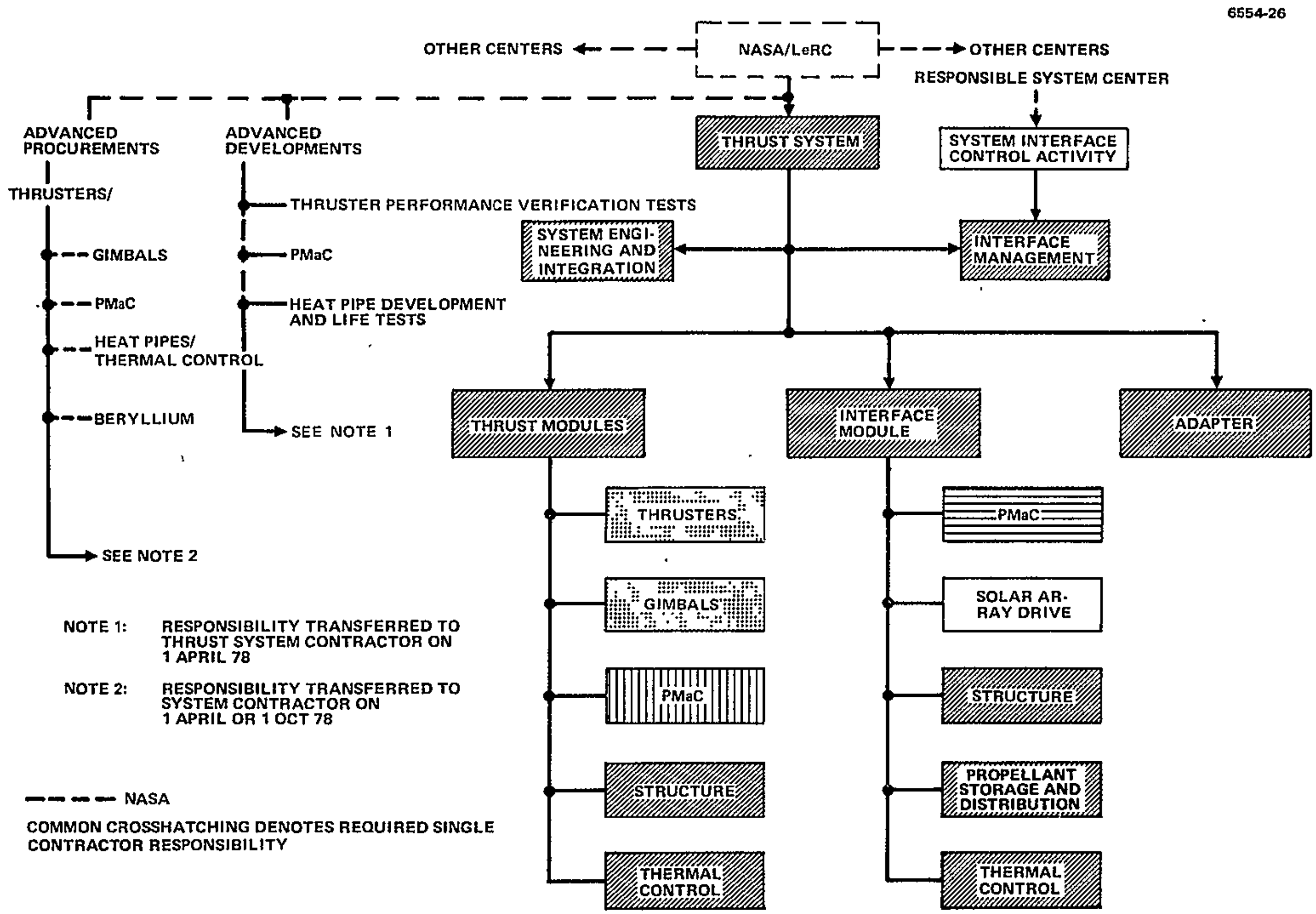


Figure 28. Recommended thrust system procurement and management plan.

recommendation reflects the general ground rules discussed in Section 4.A. We also recommend that special attention be paid to system interfaces. This is reflected in the proposed central system interface control activity and in the centralized thrust system interface management group; this group must coordinate the communication of system interface specifications.

SECTION 5

ESTIMATED PROGRAM COST

Cost estimates, prepared on the basis of available data, are summarized in Table 17. These estimates are preliminary and approximate. The total estimated cost of $53.7 million (in FY 1977 dollars and excluding fee) for the development, procurement, and testing of the thrust system is disaggregated by work breakdown structure (WBS) categories and by fiscal year requirements.

The program plan's requirement for advanced development and procurement is reflected in the estimated $13.4 million cost for FY 1978 (which includes the funds required for September of FY 1977). Included in these estimates are the costs for (1) the development of all units and subsystems through the testing and delivery of unit and subsystem models; (2) the qualification program, which includes all unit/subsystem procurements, fabrication, assembly, and testing through the delivery of the qualification models; (3) the flight system, including all unit/subsystem procurement, fabrication, assembly, and testing through delivery of the flight thrust system; (4) support during spacecraft system testing and for launch mission operations; (5) requisite auxiliary ground equipment (AGE) (including shuttle cradle), ground support equipment, and modifications to and operation of facilities; and (6) all the required interface technical and management activities. The major items presumed to be government furnished equipment (GFE) were

- Mission module electrical simulator and mass models
- Stowed solar array mass model
- 900-series EMTs for early thruster performance verification tests
- NASA LeRC test facility.

The cost of personnel to conduct tests at the NASA LeRC facility is, however, included in the estimates.

Table 17. Preliminary Estimate of Thrust System Costs[a] by Category and by FY

Work Breakdown Structure Category	Cost,[b] $\$10^6$	Fiscal Year	Cost,[b] $\$10^6$
Thrusters and gimbals	5.8	1978[c]	13.4
PMaC — thrust modules	15.0	1979	19.8
PMaC — interface module	8.1	1980	15.2
Thermal control	2.5	1981	3.8
Propellant storage and distribution	0.9	1982	1.2
Solar array drive	0.6	1983	0.2
Structural mechanics	1.2	1984	0.05
Structure and harness	2.5	1985	0.05
		Total	53.7
Design integration	1.4		
System engineering	4.2		
System tests	2.1		
AGE	1.7		
Facilities	0.3		
Spacecraft test and integration	0.4		
Pre-launch operations	0.2		
Mission operations	0.3		
Program management	6.5		
Total	53.7		

[a]Fee excluded.

[b]Expressed in FY 1977 dollars.

[c]Includes September 1977.

The estimates of system engineering and program management costs correspond to the manpower loading curves for these two activities as shown in Figure 29; these manpower estimates correspond to the level of effort versus time as reflected in the fiscal year costs in Table 17. The cost of mission operation support corresponds to the proposed five men for the first six months after launch and an average of one man for the remaining three years of the mission.

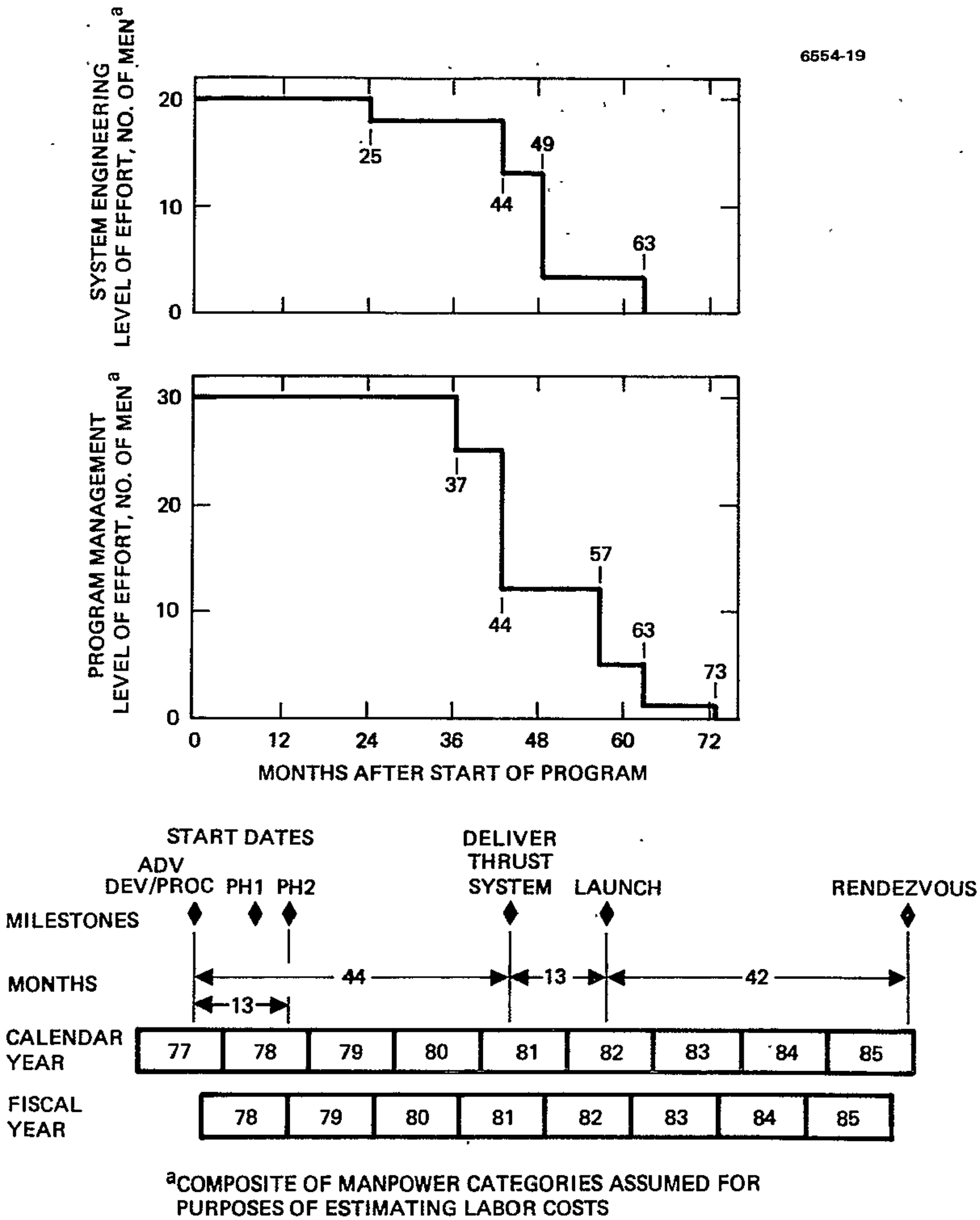


Figure 29. Proposed system engineering and program management manloading.

SECTION 6

APPROACH CONFIRMATION AND ANALYSIS

The approach confirmation and analysis task was conducted in parallel with the definition and evaluation of the thrust system conceptual design (described in the preceding sections). Three technology areas were investigated. One was to evaluate thruster performance and lifetime characteristics for the modifications in thruster operating and design parameters needed to satisfy the extended performance application. A second area was the design and evaluation of the high-voltage isolators needed for thruster operation at higher specific impulse (beam voltages to 5 kV). The third area was to further explore the potentially attractive CDVM concept as an alternative to the conventional beam supply. Results of this investigation are summarized in this section, and a detailed discussion is presented in Volume IV of this report.

A. THRUSTER PERFORMANCE AND LIFETIME EVALUATION

The objectives of this task were to demonstrate the operation of the 30-cm EMT, modified as required, in the specific impulse range of 4,000 to 5,000 sec and at power levels in the 6 to 8 kW range required for this extended performance mission application. Using the previously demonstrated performance at 3,000 sec and 2.5 kW (see Section 3.B.1) is a starting point, this investigation sought to achieve this extended capability with minimal thruster modifications.

The task was successfully accomplished: the required operation was obtained with the 900-series EMT design by simply increasing the interelectrode spacing of the ion acceleration electrodes. It was necessary to operate without the 900-series EMT propellant electrical isolators because the voltage rating of these components is below the extended performance requirement. All other elements of the thruster design functioned well at the higher specific impulse and power levels. A high voltage propellant electrical isolator is needed; recommendations for these are discussed in Section 6.B.

The performance characteristics of the EMT have been explored empirically to the extent that performance parameters for 3,000 sec operation can be determined analytically for a specified beam current. We used this analytic model extensively during this study to explore parameter variations for specified beam voltage and beam current. Measured performance and analytically predicted performance were compared (see Figure 30). The relatively good agreement between them provides a high level of confidence that:

- The model describes thruster operation reasonably well
- The basic thruster processes are not significantly different in the extended performance range
- The parameters documented for EMT operation are not expected to be significantly different under extended performance operating conditions.

The one exception under the last one is some degree of concern relative to thruster lifetime. The discharge chamber wear rates (resulting from ion sputtering) were measured using multilayer thin-film erosion monitors; these rates were greater than the wear rates computed using ion densities and sputtering rates and than those measured in other EMT programs. Since a satisfactory explanation for these differences was not obtained in the time allotted, the question of wear rate remains to be answered. In fact, confirmation of wear rates appears to be required for the 900-series EMT operating in either the normal or extended-performance range. In any case, several relatively minor design modification options exist[11] that could yield a wearout lifetime in excess of the required 15,000 hr. Such design modifications must be investigated and specified before October 1978 if they are to be incorporated in the program described in Section 5, but this should not present any serious problems.

B. THRUSTER ISOLATOR DESIGN AND EVALUATION

The current 30-cm EMT propellant isolator is designed for a maximum operating voltage of 1.5 kV. The objective of this task was to define and evaluate design modifications required for extending the operational

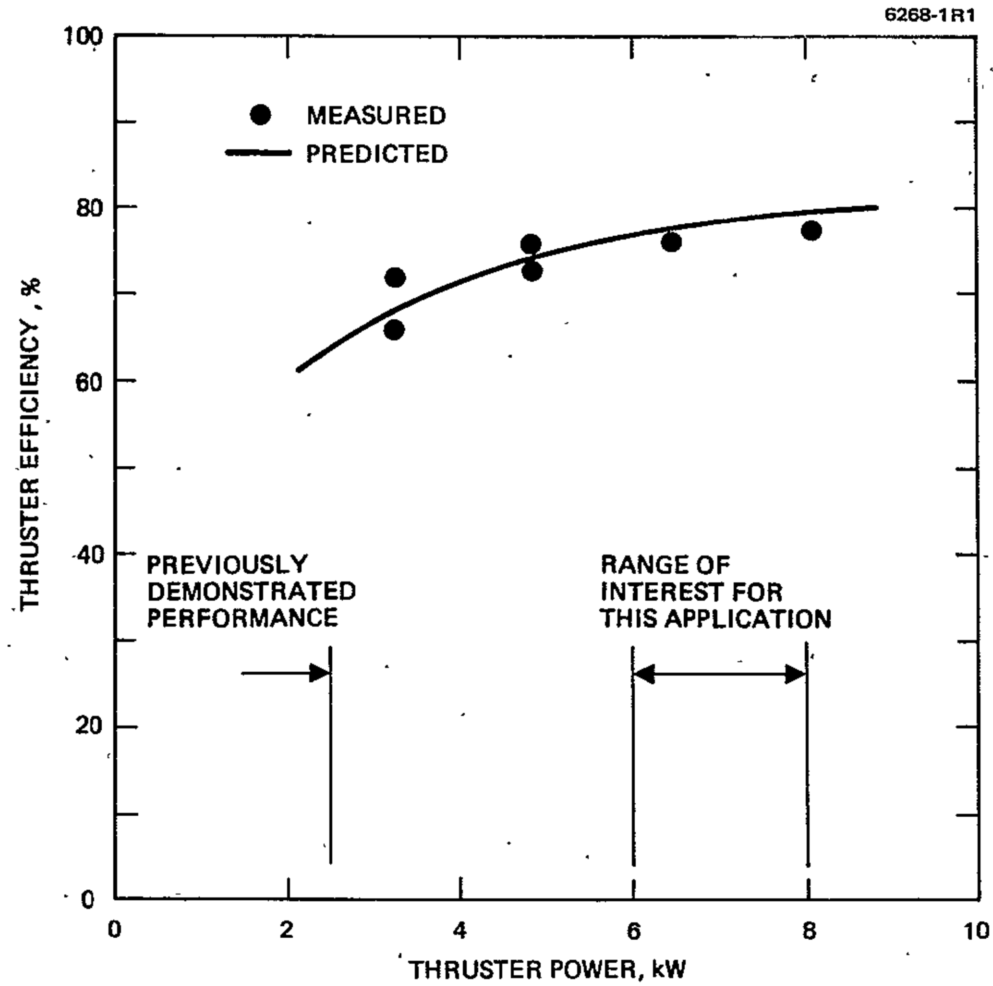
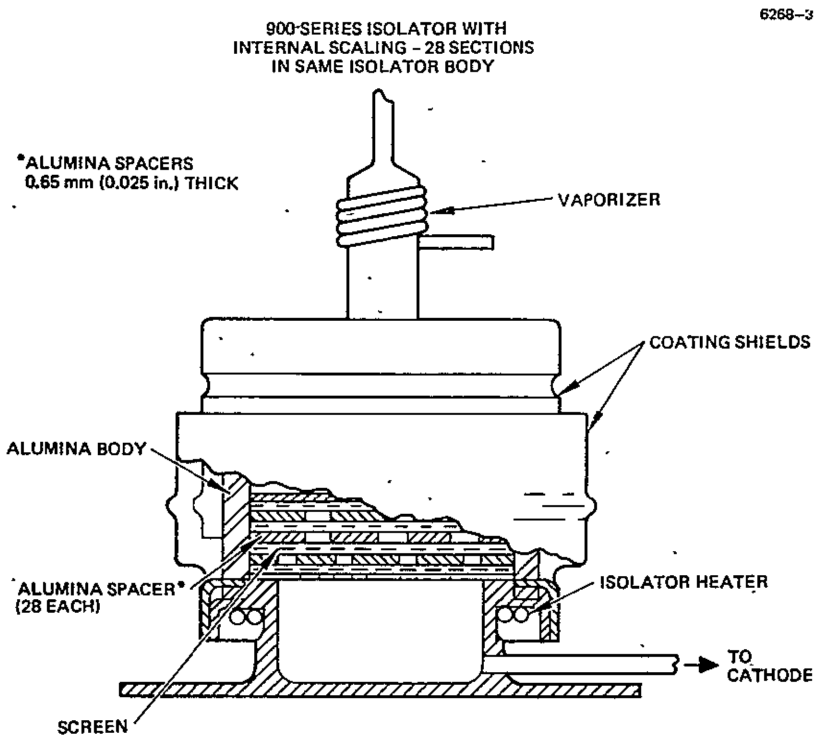


Figure 30. Comparison of measured and predicted thruster performance.

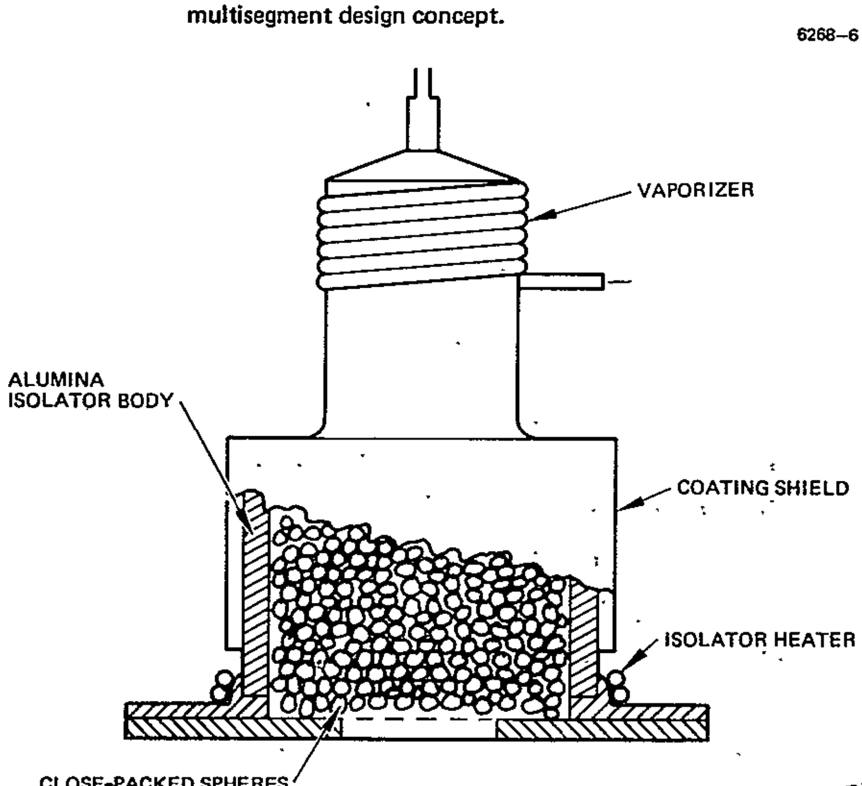
range up to 5 kV to provide a margin over the 4 kV estimated to be required for the Halley's comet mission. Two design concepts were explored.

The first concept is based on the EMT design; in it, the insulating region of the vapor flow channel is divided into several short segments, as shown in Figure 31(a). This concept is based on the principle that the voltage applied across the isolator divides equally such that the voltage across any segment is less than the Paschen minimum. (The breakdown voltage between parallel electrodes is a function of electrode spacing, gas pressure in the interelectrode space, type of gas, and electrode materials. These functional relationships have been determined empirically and are called Paschen curves.) The number of segments in the new isolator was increased by a factor of four over the 900-series isolator (from 7 to 28) to obtain the required isolation. Components were built and tested for two separate design approaches. The first maintained the same segment length as in the EMT design; the second maintained the same overall isolator length. Both designs were determined to be capable of withstanding voltages of up to 6 kV without breakdown under full operating temperature and mercury vapor flow.

The second isolator concept uses an insulating labyrinth to inhibit breakdown. An isolator that utilizes this concept was fabricated and tested. The design, shown in Figure 31(b), has ceramic spheres tightly packed in the insulating chamber. Based on the results reported by other investigators, the diameter of the ceramic spheres should have been ≈0.1 to 0.2 mm. But the only ceramic spheres obtainable were a mixture ranging from 0.2 to 2.0 mm in diameter, with the greatest number measuring about 1 mm. Although the larger spheres were screened out of the mixture, the isolator tested exhibited breakdown at slightly over 2 kV. Although these findings do not conclusively eliminate this concept, it is evident from the test results that the breakdown of the insulating labyrinth isolator is significantly more sensitive to temperature and vapor flow than is the multisegment isolator. Consequently, the multisegment design was selected for further investigation and was subjected to extended testing.



(a) High-voltage propellant electrical isolator using multisegment design concept.



(b) High-voltage propellant electrical isolator using insulating labyrinth design concept.

ORIGINAL PAGE IS
OF POOR QUALITY

Figure 31. Schematic of the two isolator concepts considered.

A cathode-isolator-vaporizer (CIV) assembly using the multi-segment approach was tested for 300 hr with both the isolator temperature and vapor flow rate set to values higher than normal. A voltage of 4 kV was applied across the isolator and the leakage current was continuously monitored and recorded. Figure 32 shows the increase in leakage current that was observed. After completion of the 300 hr test, it was experimentally determined that the magnitude of the leakage current varies with both isolator temperature and vapor flow rate and is essentially linear with the applied voltage. Similar leakage current behavior was observed in the development of the 900-series EMT isolator[12] and was eventually correlated with surface contamination of the ceramic insulation. This leakage was eliminated from the 900-series EMT isolator by taking appropriate assembly precautions and by shadow shielding the insulator. Although determining the reasons for the leakage current behavior shown in Figure 32 was beyond the scope of this study, they are probably the same as for the similar behavior of the 900-series EMT isolator. Although the EMT fabrication and handling procedures were used in preparing the isolator tested here, these procedures may not be adequate for the higher voltage, temperature, and vapor flow rates used in this test. Consequently, a systematic re-examination of procedures and operating parameters is required to establish the procedural specifications necessary to eliminate isolator leakage under the extended performance conditions.

If isolator leakage cannot be eliminated, but can be limited to a linear increase with time that is no greater than shown in Figure 32 (about 0.4 μA/hr), then the isolator design could be considered to be adequate because the total leakage at 15,000 hr would only be 6 mA per isolator, which is only a fraction of one percent of the total beam current. A verification that leakage remains linear is still required, however, because past experience indicates[12] that leakage current behavior such as this becomes exponential with time when a given value of leakage current is reached. The results of this analysis and test program have shown that the multi-segment isolator design is acceptable for the extended performance 5 kV operation of the 30-cm EMT, with the

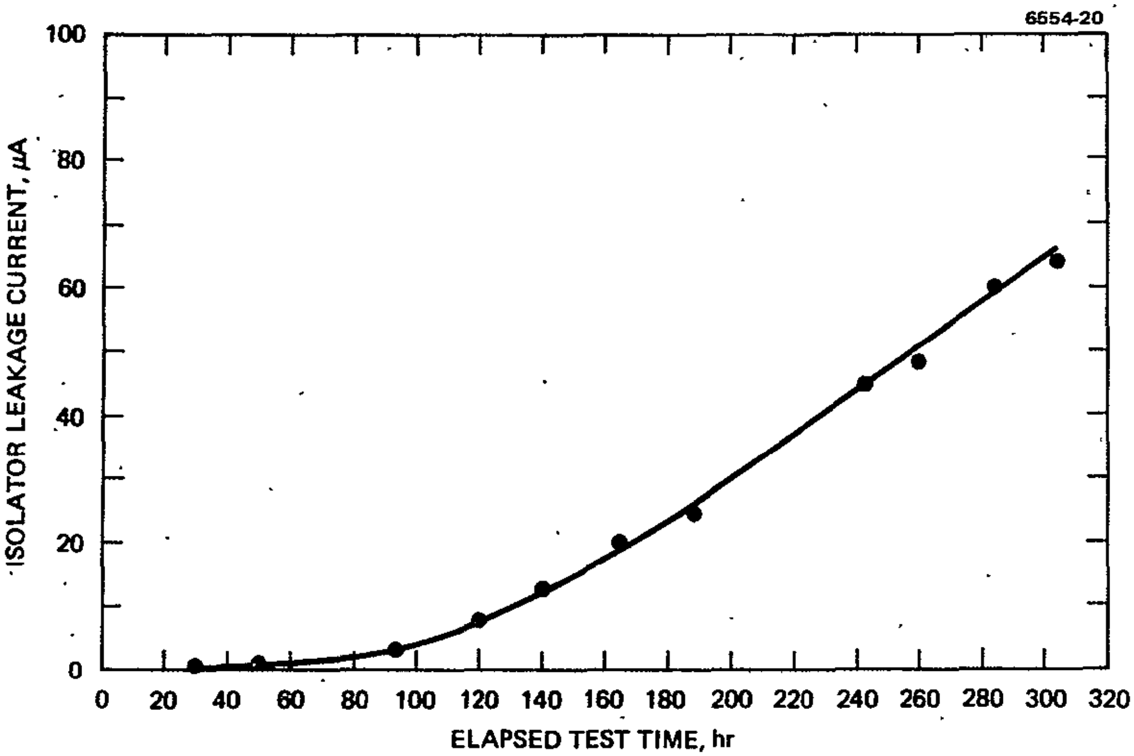


Figure 32. Isolator leakage current versus elapsed test time for design in Figure 31(a).

qualification that a further confirmation is required of the time/temperature/leakage behavior of this design.

C. DESIGN, TESTING, AND EVALUATION OF A 1 kW VOLTAGE MULTIPLIER MODEL

The CDVM concept is believed to offer potentially significant mass and reliability benefits to solar electric propulsion as a replacement for the conventional beam supply. A development program was therefore begun to demonstrate design feasibility and the performance of this concept at higher power levels (the eventual goal is operation in the 6 kW range for the Halley's comet or other missions). Specifically, the objective of the CDVM task in this program was to design, fabricate, and test a 1 kW CDVM model, thereby confirming analytic performance predictions both at 1 kW and for potential design extension to 6 kW.

Design tradeoffs resulted in the selection of a five-phase system. Compared to single-phase designs, this multiphase approach minimizes peak currents in semiconductor devices, dramatically reduces the total capacitance requirement, and reduces the weight of the input and output filters because the ripple frequency is higher. A design with more than five phases was considered to be unnecessarily complex at this stage of development.

An integral part of the task was to design and fabricate low-loss, light-weight capacitors. To minimize the power losses in the CDVM capacitors, special termination techniques were developed; these yielded consistently low termination resistance. Polysulfone dielectric film was chosen because of its low dissipation factor and wide wervice temperature. The units fabricated were 0.6 μF, 0.45 μF, and 0.30 μF capacitors, rated at 600 Vdc and used at 300 Vdc maximum. Other components used for the model were commercially available devices. The power transistors chosen were Motorola MJ7261 (rated at I_C = 15 A continuous and at V_{ceo} = 400 Vdc maximum). Semtech 3FF50 rectifiers were chosen because of their fast reverse recovery time (30 nsec); they are rated at 500 Vdc blocking voltage, and 1 A dc continuous forward current.

Magnetic components were designed using commercially available cores and bobbins.

The CDVM model was successfully operated at a power output level in excess of 1 kW over a range of input voltages and load currents, and under various load fault conditions. Full recovery from a short-circuit at the output has been demonstrated. Table 18 summarizes some of the significant test results. Of special interest are the relatively low component weight-to-power ratio (0.5 kG/kW), low output ripple voltage (less than 1% peak to peak), and high efficiency (in excess of 96%).

Results of this investigation confirm previous predictions of efficiency and weight, and strengthen confidence that a 6 kW CDVM can be fabricated. It is anticipated that the 6 kW design could be accomplished using available components and the same basic design, except that the number of phases and stages would have to be increased. Fabricating and testing such a 6 kW model logically constitutes the next phase of CDVM development.

Table 18. CDVM Test Results Summary

Parameter	Value
Test conditions	
Input voltage, Vdc	300
Load (resistive), Ω	∿1830
Operating frequency, kHz	65
Test results	
Output voltage, Vdc	1474.5
Output current, Adc	0.805
Output power, W	1187
Voltage transfer ratio	4.915
Total input power, W	1233
Overall efficiency, % (including logic and drive losses)	96.2
Output ripple voltage, V (peak-to-peak)	12
Weight of all components, kg	0.59
Component weight/output power ratio, kg/kW	0.5

SECTION 7

RISK ASSESSMENT

A. TECHNICAL ASSESSMENT

The most significant technical risks that can be identified for the proposed thrust system design are listed in Table 19. Each is believed to be resolvable through application of sound engineering effort on a time scale compatible with the Halley's comet mission. Principal concerns are thruster design and performance in the extended performance range, PMaC electronics complexity and reliability for prolonged operation in space, and potential system EMI effects. No significant technical risks are believed to arise in structural/thermal design, and no novel technology (with associated risks) is required in the conventional PMaC design adopted. The risks listed in Table 19 do not include those associated with the solar array or with other components of the spacecraft for the Halley's comet mission.

The areas of concern in Table 19 are those that could be identified at this preliminary, design concept stage of development. The only new component is the high-voltage isolator, and the results from the concurrent design and testing effort reported in Section 6.B provide a high level of confidence for this development. Problems associated with PMaC electronics complexity, controller EMI susceptibility, and thruster reliability in the high-power operation mode are believed to be manageable with the application of available engineering skills during the design development phase.

Fulfillment of the 13,600 hr thruster-life requirement for this application is, perhaps, the only problem with a still unconfirmed solution. Intensive effort is being applied, and the proposed program plan recommends that these efforts be continued during the initial development phase. Further evaluation of wear rates by tests has a reasonable probability of proving the adequacy of the current design. Alternately, readily implemented design modifications are available to increase life expectancy, such as use of an ion optics design that incorporates a small-hole accelerator grid.

C-2

Table 19. Significant Technical Risks Associated with Thrust System Design

Risk	Reason for Concern
Isolator life and performance at high voltage	New component
Thruster failure rate at high-power of high-voltage operation	Greater energy into accel grid during arcing and increased stress on insulators
Thruster life	15,000 hr not yet demonstrated for 900-series or uprated design
Complexity and reliability of PMaC design for high-power operation	High thermal loading and stress level of components (qualifiability)
	High parts count (added redundancy costly)
Controller EMI susceptibility during high-power thrusting	Nature/effect of severe EMI environments not addressed or provided for

B. SYSTEM INTERFACES

The basic interfaces between the thrust system and the other major spacecraft elements — solar array and mission module — are simple. There is, however, an intrinsic interrelationship between (1) the design and performance of the thrust system and (2) the design, requirements, and constraints of the other major elements of the spacecraft (the solar array, mission module, IUS, and shuttle).

The challenge is to affect the early specification of the major system interfaces and to manage the interfaces during the program. There is no technical deterrent to the specification of the interfaces and subsequent design of the major systems by the individual responsible parties. Under the plan recommended here, and presented in Section 4, a single contractor (under NASA LeRC sponsorship) is responsible for thrust system design, procurement, and delivery. It is anticipated that the thrust system contractor would participate with NASA LeRC in a total system interface working group. By establishing management of the program at this level, the challenge of the design of each major system — thrust system, solar array, and mission module — and integration of the systems into the Halley's comet mission spacecraft can be met.

There are several areas of design interdependence:

(1) The design of the thrust system will be significantly affected by design characteristics and by the requirements of the other components of the Halley's comet system. The assumptions that it was necessary to make during this study must be verified and/or changed to further improve the overall design.

(2) The design of the thrust system affects the design characteristics of the other system components.

(3) Overall design integrity and performance also depends on factors that involve all subsystems. To resolve potential problems and assure system integrity require a coordinated analysis and test effort by all participants.

Specific examples of each category are presented below to illustrate the nature and scope of the interface effort involved.

In category (1), the following factors play a major role in defining thrust system design:

- The size and shape of the solar array stowed envelope
- Solar array power profile
- Mission module physical and thermal characteristics and requirements
- Mission module control system constraints
- Mission module data processing design characteristics and requirements
- Mission module operations doctrine (definition of PMaC controller)
- Mission module EMI susceptibility
- Mission profile/trajectory (thruster power levels, utilization plan, life requirements)

- IUS loads
- IUS clearance requirements and tipoff rates
- Shuttle loads
- Shuttle safety and other operation constraints.

Examples of subsystem designs (other than in the thrust system) that are affected by thrust system characteristics (category (2)) are

- Solar array profile management plan (reconfiguration requirements)
- Solar array deployment requirements (prevention of Hg ion impingement)
- Maximum power tracking design
- Mission module control system design (including requirements for spacecraft tilting)
- Mission module data processing design
- Mission module electromagnetic compatibility (EMC) design features
- Shuttle cradle design.

System-level technical activities (category 3) comprise iterative analysis and design tasks implicit in the above listing, as well as additional activities; these additional activities

- Coupled load analyses (IUS and shuttle)
- Coupled thermal analyses
- Combined trajectory/mission analyses
- System level EMI — analysis and testing
- Mission management and mission operations plan
- Integrated system tests.

Effective management is mandatory for the successful resolution of these difficult system interfaces. The interface management plan should include the following organizational and technical control features:

- Clearly defined central authority and responsibility
- Responsible and responsive channels of direct involvement and reporting by all participants to this central authority
- Early definition of subsystem designs
- Effective control of design changes
- Design definition and timely provision of simulators and models.

The first two of the above are reflected in the recommended procurement plan in Section 4 (Figure 28). The third item, early definition of designs, is probably the most crucial requirement from a schedule standpoint and the most difficult one to implement. It is reflected in the master phasing schedule, Figure 19, and in the proposed program plan in Section 4.

C. PROGRAM IMPLEMENTATION

An overall appraisal of the probability of successfully accomplishing the development, procurement, and testing of the thrust system for the Halley's comet mission must consider, in addition to the technical risks associated with the achievement of design goals and with the resolution of interfaces (discussed in the preceding sections), the schedule risks in meeting the required milestones and the economic risks of cost estimates.

Adequate time is believed to be available to accomplish the Halley's comet mission, provided that the initial phases of the program are implemented without delay. The key requirements are

- Immediate initiation of PMaC system design and of advanced development and procurement of breadboard units

- Immediate initiation of thruster performance verification tests

- Initiation of procurements for thruster components on or about 1 January 1978

- Initiation of heat pipe development and procurement in the spring of 1978.

Postponing these advanced activities would probably result in nonrecoverable schedule slippage. The time spans for the other phases of the program, including system integration after thrust system delivery and before launch, are believed to be tight but adequate, even allowing for a reasonable number of the development problems expected for this type of program.

Confidence in the overall success does not, however, preclude the need to identify and acknowledge the existence of schedule risks. These are summarized in Table 20 in order of concern, with the most serious risks listed first.

Table 20 necessarily includes some of the technical and interface concerns of the preceding subsections to the extent that they affect schedule concerns. An important schedule concern involves PMaC electronics development: even with advanced development and procurement, and with the overlap provided in the program plan among the development, qualification, and flight procurement phases, the time available for these activities will require an intensive engineering effort. The overlap between these activities is itself a further concern (as indicated in Table 20) because of the possibility that significant design changes may be required. An equal concern is the potentially serious schedule slippage that could occur if the requirements for interface definition and management discussed in the preceding subsection are not met.

Potential tradeoffs exist among some of the technical and schedule concerns, system design parameters (notably mass allowance), and available funds. For example, development risks (and associated schedule concerns) regarding structural design for loads or regarding heat pipe reliability could, in principle, be alleviated by providing greater mass contingency

Table 20. Principal Schedule Risks

Tight PMaC development schedule (even with advance procurements)
Timely interface definition
Design characteristics of mission module and solar array
Interface requirements
Interface specifications
Prompt definition and early freeze of thrust subsystem design
Timely delivery of advance procurement critical parts
PMaC parts (hybrids)
Beryllium
Availability of heat pipes: development/delivery
Efficient management and control of interfaces
Spacecraft system interfaces
Shuttle interfaces
IUS interfaces
Overlap between development/qual/flight design and test
Impact of technical/design changes
Availability of personnel for parallel test operations
Unavailability of backup facilities for thrust module/thrust system tests
Special procurement risks (risk/cost trades)
Single shuttle cradle
Single adapter
Single beryllium vendor

and/or by including additional, parallel design and testing activities. Schedule risks could be further reduced by fabricating additional spares during the development phase and for flight units.

It is much more difficult to reliably assess economic risks. The cost estimates presented in Section 5, which are based on extensive experience in the design and procurement of space systems, are believed to be fairly accurate. However, they are dependent on the assumptions made regarding program scope and system interfaces, and, more importantly, on program contingencies arising from the technical, interface, and schedule risks. Furthermore, cost estimates depend on the procurement plan. The estimates provided in Section 5 should therefore be treated as, at best, a funding baseline, and plans for total program cost must take these additional factors into account.

SECTION 8

STUDY CONCLUSIONS

The study successfully met its objectives; the principal accomplishments are discussed below.

A. CONCEPT SELECTION

An attractive baseline configuration for the thrust system was selected for the 30-cm extended-performance mercury ion thruster from among a spectrum of options considered. The selected baseline uses a concentrator solar array and a conventional PMaC design.

B. CONCEPTUAL DESIGN OF BASELINE CONFIGURATION THRUST SYSTEM

A highly integrated and versatile thrust system configuration was generated for the HCRM. This design while being near optional in terms of performance, mass, and reliability, maintained a significant amount of modularity. The modular feature of the thrust system allows for technology growth and configuration flexibility for other missions and provides for a standard thrust module which can be manufactured and tested with relative ease.

C. EXTENDED PERFORMANCE MERCURY ION THRUSTER

Adaptability of the 900-series, 30-cm thruster design to the 6 to 7 kW extended performance operation range (which is required for the Halley's comet mission) was demonstrated with only minor design modifications required, and an acceptable high-voltage isolator design was validated by laboratory tests.

D. DESIGN SENSITIVITY

The sensitivity of the baseline design to the key design parameters and to the level of solar array power was established, and areas of potential improvement through iterative mission/trajectory analysis were identified.

E. DEVELOPMENT OF THE CDVM

Design and performance of the alternative PMaC design concept utilizing the CDVM, which has potential mass, efficiency, and reliability advantages over the conventional beam supply, has been successfully demonstrated by laboratory model tests at power levels in excess of 1 kW, and shows promise of extension to the 6 kW level.

F. GROWTH POTENTIAL

A significant level of potential growth capability (e.g., for other mission applications) has been provided in the baseline design by using a modular design approach and by design features that minimize the modifications required for an increase in thrust levels, for augmentation of thermal control, and for the substitution of the CDVM for the conventional beam supply.

G. TECHNICAL RISKS

The technical risks associated with the thrust system design have been identified. Since the problems posing the risks are considered resolvable through nominal engineering development, the risks are judged to be acceptable for mission application.

H. INTERFACES

Interfaces with the solar array, mission module, IUS, and shuttle for the Halley's comet mission have been identified, and significant technical effort and management attention will be required for their successful resolution, including the conduct of iterative mission/trajectory analysis and design optimization, and an early definition of key design parameters.

I. PROGRAM PLAN

A viable program plan and an associated procurement plan have been generated for the baseline configuration, with schedule requirements and priorities identified, that can lead to the successful accomplishment of the Halley's comet mission.

J. COSTS

Cost estimates and fiscal year funding requirements for the thrust system development and procurement for the Halley's comet mission have been generated, indicating a total cost of about $54M in FY 77 dollars, excluding contractor fee, of which approximately $13.5M is required for advanced development and procurement in FY 78 to meet schedule requirements.

REFERENCES

1. T.D. Masek, R.L. Poeschel, C.R. Collett, and D.E. Schnelker, "Evolution and Status of the 30 CM Engineering Model Ion Thruster," AIAA Paper No. 76-1006, Key Biscayne, Fla., November 1976.

2. R.T. Bechtel and V.K. Rawlin, "Performance Documentation of the Engineering Model 30 CM Diameter Thruster," AIAA paper No. 76-1033, Key Biscayne, Fla., November 1976.

3. C. Collett, et al., "Thruster Endurance Test," NASA CR-135011, May 1976.

4. C. Collett, "Fabrication and Verification Testing of ETM 30 CM Diameter Ion Thrusters," NASA CR-135193, April 1977.

5. M.J. Mirtich and W.R. Kerslake, "Long Lifetime Hollow Cathodes for 30 CM Mercury Ion Thrusters," AIAA Paper No. 76-985, Key Biscayne, Fla., November 1976.

6. M.J. Mirtich, "The Effects of Exposure to LN_2 Temperatures and 2.5 Suns Solar Radiation on 30 CM Ion Thruster Performance," NASA TMX-71652 (1975).

7. J.C. Oglebay, "Thermal Analytic Model of a 30 CM Engineering Model Mercury Ion Thruster," NASA TMX-71680 (1975).

8. R.L. Poeschel and R.P. Vahrenkamp, "High Power Operation of a 30 CM Mercury Bombardment Ion Thruster," AIAA Paper No. 76-1007, Key Biscayne, Fla., November 1976.

9. J.E. Cake, G.R. Sharp, J.C. Oglebay, F.J. Shaber, and R.J. Zavesky, "Modular Thruster Subsystem Approaches to Solar Electric Propulsion Module Design," NASA TMX-73502, November 1976.

10. F.D. Berkopec, J.C. Sturman, and R.W. Stanhouse, "Solar Array Drive System," Proc. 10th Aerospace Mechanisms Symposium, JPL-TM-33-777, Jet Propulsion Laboratories (1976).

11. R.P. Vahrenkamp, "Characteristics of a 30 CM Thruster Operated with Small Hole Accelerator Grid Ion Optics," AIAA Paper No. 76-1030, Key Biscayne, Fla., November 1976.

12. M.A. Mantenieks, "Status of 30 CM Diameter Mercury Ion Thruster Isolator Development," AIAA Paper No. 76-1027, Key Biscayne, Fla., November 1976.

100
PAGE INTENTIONALLY BLANK

PRECEDING PAGE BLANK NOT FILMED

DISTRIBUTION LIST

Recipient		Copies
NASA Lewis Research Center 21000 Brookpark Road Cleveland, OH 44135		
Dr. Bernard Lubarsky	MS 3-3	1
Dr. Seymour C. Himmel	MS 3-7	1
Daniel J. Shramo	MS 303	1
Elmer H. Davison	MS 3-3	1
Howard W. Douglass	MS 501-5	1
H. Warren Plohr	MS 501-5	1
Robert R. Lovell	MS 501-4	1
James F. DePauw	MS 501-4	1
James E. Cake	MS 501-4	25
Allan Jones	MS 500-313	1
Library	MS 60-3	1
Report Control Office	MS 5-5	2
Public Information Office	MS 3-11	1
NASA Scientific and Technical Information Facility P. O. Box 8757 Baltimore - Washington Internation Airport Baltimore, MD 21240		10
NASA Headquarters Washington, DC 20546		
Dr. John E. Naugle	Code AA	1
Dr. James J. Kramer	Code R	1
James Lazar	Code RP	1
Wayne R. Hudson	Code RPE	1
Jerome P. Mullin	Code RPP	1
Fred J. DeMeritte	Code RC	1
William C. Hayes, Jr.	Code RS	1
Edward A. Gabris	Code RS	1
Frank T. Rosenburg	Code BR	1
John F. Yardley	Code M	1
John H. Disher	Code MT	1
Lester K. Fero	Code MTE	1
Joseph B. Mahon	Code MV	1
B. C. Lam	Code MVE	1
Dr. Noel W. Hinners	Code S	1
A. Thomas Young	Code SL	1
Daniel H. Herman	Code SL	1
Paul F. Tarver	Code SL	1
Joseph F. Cork	Code SL	1

NASA George C. Marshall Space Flight Center
Marshall Space Flight Center, AL 35812

W. R. Lucas	DA01	1
R. G. Smith	DD01	1
R. E. Austin	PS04	3
L. E. Young	EC12	1
J. L. Miller	EC12	1

Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91103

Dr. Bruce C. Murray	M.C. 180-905	1
J. N. James	M.C. 180-901	1
Norman R. Haynes	M.C. 180-704	1
Dr. K. L. Atkins	M.C. 180-700	3
Dr. Peter T. Lyman	M.C. 157-205	1
J. R. Stuart	M.C. 233-307	1
R. Ploszaj	M.C. 233-307	1
J. E. Graf	M.C. 122-123	1

Jet Propulsion Laboratory 1
Attn: E. N. Costogue FH-B201
2500 Foothill Blvd.
Pasadena, CA 91103

TRW Defense and Space Systems Group 1
One Space Park
Redondo Beach, CA 90278

J. J. Biess	M.C. 82/2367	1
A. D. Schoenfeld	M.C. 82/27200	1

Rockwell International 1
Attn: Dr. James B. Weddell SK71
12214 Lakewood Blvd.
Downey, CA 90241

Boeing Aerospace Co. 1
Attn: Charles Terwilliger M.C. 84/95
Seattle, WA

Fairchild Space and Electronics Co. 1
Attn: Steven S. Myers
Germantown, MD 20767

Science Applications, Inc. Attn: John Niehoff One Woodfield Place Bldg., Suite 109 1701 East Woodfield Road Schaumburg, IL 60196	1
Battelle Columbus Laboratories Attn: R. R. Teeter 5058 King Avenue Columbus, OH 43201	1
General Dynamics Convair Division Attn: J. W. Streetman P. O. Box 80847 San Diego, CA 92138	1
Wright-Patterson Air Fórce Base AFAPL Wright-Patterson Air Force Base, OH	
Hans VonOhain	1
J. Reams	1
L. Obery	1
SAMSO/YAT P. O. Box 92960 Worldway Postal Center Los Angeles, CA 90009	
Col. Woods	1
Capt. Robert Ford	1
The Aerospace Corporation P. O. Box 92957 Los Angeles, CA 90009	
John Mosich	1