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FINAL REPORT
SYSTEM DESIGN OF THE PIONEER VENUS SPACECRAFT

VOLUME 10
PROPULSION/ORBIT INSERTION SUBSYSTEM STUDIES

By

B. J. ROSENSTEIN

ET AL.

July 1973

Prepared Under

Contract No. [redacted] NAS2-7250

By

HUGHES AIRCRAFT COMPANY

EL SEGUNDO, CALIFORNIA

For

AMES RESEARCH CENTER

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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PREFACE

The Hughes Aircraft Company Pioneer Venus final report is based on study task reports prepared during performance of the "System Design Study of the Pioneer Spacecraft." These task reports were forwarded to Ames Research Center as they were completed during the nine months study phase. The significant results from these task reports, along with study results developed after task report publication dates, are reviewed in this final report to provide complete study documentation. Wherever appropriate, the task reports are cited by referencing a task number and Hughes report reference number. The task reports can be made available to the reader specifically interested in the details omitted in the final report for the sake of brevity.

This Pioneer Venus Study final report describes the following baseline configurations:

- "Thor/Delta Spacecraft Baseline" is the baseline presented at the midterm review on 26 February 1973.
- "Atlas/Centaur Spacecraft Baseline" is the baseline resulting from studies conducted since the midterm, but prior to receipt of the NASA execution phase RFP, and subsequent to decisions to launch both the multiprobe and orbiter missions in 1978 and use the Atlas/Centaur launch vehicle.
- "Atlas/Centaur Spacecraft Midterm Baseline" is the baseline presented at the 26 February 1973 review and is only used in the launch vehicle utilization trade study.

The use of the International System of Units (SI) followed by other units in parentheses implies that the principal measurements or calculations were made in units other than SI. The use of SI units alone implies that the principal measurements or calculations were made in SI units. All conversion factors were obtained or derived from NASA SP-7012 (1969).

The Hughes Aircraft Company final report consists of the following documents:

Volume 1 - Executive Summary - provides a summary of the major issues and decisions reached during the course of the study. A brief description of the Pioneer Venus Atlas/Centaur baseline spacecraft and probes is also presented.

Volume 2 - Science - reviews science requirements, documents the science-peculiar trade studies and describes the Hughes approach for science implementation.

Volume 3 - Systems Analysis - documents the mission, systems, operations, ground systems, and reliability analysis conducted on the Thor/Delta baseline design.

Volume 4 - Probe Bus and Orbiter Spacecraft Vehicle Studies - presents the configuration, structure, thermal control and cabling studies for the probe bus and orbiter. Thor/Delta and Atlas/Centaur baseline descriptions are also presented.

Volume 5 - Probe Vehicle Studies - presents configuration, aerodynamic and structure studies for the large and small probes pressure vessel modules and deceleration modules. Pressure vessel module thermal control and science integration are discussed. Deceleration module heat shield, parachute and separation/despin are presented. Thor/Delta and Atlas/Centaur baseline descriptions are provided.

Volume 6 - Power Subsystem Studies

Volume 7 - Communication Subsystem Studies

Volume 8 - Command/Data Handling Subsystems Studies

Volume 9 - Altitude Control/Mechanisms Subsystem Studies

Volume 10 - Propulsion/Orbit Insertion Subsystem Studies

Volumes 6 through 10 - discuss the respective subsystems for the probe bus, probes, and orbiter. Each volume presents the subsystem requirements, trade and design studies, Thor/Delta baseline descriptions, and Atlas/Centaur baseline descriptions.

Volume 11 - Launch Vehicle Utilization - provides the comparison between the Pioneer Venus spacecraft system for the two launch vehicles, Thor/Delta and Atlas/Centaur. Cost analysis data is presented also.

Volume 12 - International Cooperation - documents Hughes suggested alternatives to implement a cooperative effort with ESRO for the orbiter mission. Recommendations were formulated prior to the deletion of international cooperation.

Volume 13 - Preliminary Development Plans - provides the development and program management plans.

Volume 14 - Test Planning Trades - documents studies conducted to determine the desirable testing approach for the Thor/Delta spacecraft system. Final Atlas/Centaur test plans are presented in Volume 13.

Volume 15 - Hughes IR&D Documentation - provides Hughes internal documents generated on independent research and development money which relates to some aspects of the Pioneer Venus program. These documents are referenced within the final report and are provided for ready access by the reader.

Data Book - presents the latest Atlas/Centaur Baseline design in an informal tabular and sketch format. The informal approach is used to provide the customer with the most current design with the final report.

CONTENTS

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1. SUMMARY

1.1 PROPULSION SUBSYSTEM MAJOR ISSUES

The Pioneer Venus orbiter and multiprobe missions require spacecraft maneuvers for successful accomplishment. This report presents the results of studies performed to define the propulsion subsystems required to perform those maneuvers. Primary goals were to define low mass subsystems capable of performing the required missions with a high degree of reliability for low cost. Consistent with these goals was the objective of using existing, flight-proven components and subsystem design concepts wherever possible.

Major issues studied consisted of:

1) Regulated pressure versus pressure blowdown operating mode.

2) Helium versus nitrogen pressurizing gas.

3) Propellant selection (spacecraft control, orbit insertion).

4) Propellant management technique selection.

5) Thruster selection and arrangement optimization.

6) Latch valve arrangement optimization.

7) Orbit insertion motor selection.

A review was performed of all applicable propellants and thruster types, as well as propellant management techniques. Based on this review, a liquid monopropellant hydrazine propulsion subsystem was selected for all multiprobe mission maneuvers, and for all orbiter mission maneuvers except orbit insertion. A pressure blowdown operating mode was selected using helium as the pressurizing gas. The forces associated with spacecraft rotations were used to control the liquid-gas interface and resulting propellant orientation within the tank.

Thrusters were arranged in two groups on each spacecraft to provide redundant capability for performing all maneuvers with a minimum of

TABLE 1-1. SUMMARY OF STUDY ISSUES

Issue	Baseline Configuration	Alternatives	Rationale for Selection
Pressure Supply	Blowdown	Regulated	Blowndown represents lower mass and cost, and greater design simplicity
Pressurizing Gas	Helium	Nitrogen	Lower Mass
Propellant Management Technique	Centrifugal Force	Bladder, Surface tension device	Lowest mass and cost Greatest design simplicity
Propellant Tradeoff	1) Monopropellant Hydrazine for spacecraft control 2) Solid propellant for orbit insertion	Cold gas, hot gas, liquid bipropellant, monopropellant H_2O_2, electric	1) Low mass, simple design available hardware 2) Lower cost, flight proven performance
Thruster arrangement	22.2 N thruster Six thrusters - probe bus Seven thrusters - orbiter	4.15 N thruster in various arrangements	Minimize maneuver duration, provide reasonable maneuver granularity, provide redundancy, minimize number of components
Propellant utilization effect on center of mass	-	-	Define center of mass excursion for all mission phases.
Operational flexibility	-	-	Develop operational and handling plans
Power and thermal insulation requirements	1) Thruster valve heaters 2) Tank heaters 3) Passive thermal design elsewhere 4) Orbit insertion motor nozzle heater	1) All passive thermal design 2) Thruster catalyst bed heaters	Avoid heaters where possible, yet maintain temperatures within allowable limits.
Latch valve arrangement	Two latch valves provide two thruster groups	1) Zero latch valves 2) Four latch valves	Latch valves required. Minimize weight and cost. Provide high reliability.
Thruster operating requirements	-	-	Define operating modes, duty cycles, life requirements.
Orbit insertion motor selection	TE-M-521 motor (stretched) selected for Type II trajectory (Atlas/Centaur)	SVM-2 for Type II	High performance, existing design technology, minimum weight

components and without complicating the subsystems plumbing design. A dual latch valve configuration was selected with one latch valve for each thruster group using existing, flight-qualified components from the highly successful Intelsat IV and Telesat programs, the propulsion subsystem represents a highly reliable design.

The maneuver inserting the spacecraft into orbit about Venus presents unique requirements resulting in a separate subsystem designed specifically for that purpose. A tradeoff was performed of liquid bipropellant, monopellant hydrazine, and solid propellant motors, with the solid propellant selected based onthe lower cost, and flight-demonstrated performance.

A summary of all study issues is presented in Table 1-1.

1.2 BASELINE DESCRIPTION

The baseline propulsion subsystem for the Atlas/Centaur launched spacecraft control uses monopropellant liquid hydrazine. The subsystem operates in a pressure blowdown mode, and stores propellant and pressurant together in a common tank to provide a low mass, reliable design. Two tanks are located 180 deg apart about the spacecraft spin axis. The tank size selected accomodates the propellant required for either the orbiter or probe spacecraft, this providing common hardware for the two configurations. The propellant/pressurant interface and propellant orientation within the tanks is controlled by the centrifugal force associated with spacecraft rotation. This represents a low cost, low mass, high reliability design.

Interconnecting manifolds equalize the pressure between tanks and distribute the propellant uniformly from each tank through appropriate filters to the thrusters. Thrusters are arranged in two groups, for redundancy, with one axial and a pair of radial thrusters in each group for the probe spacecraft configuration, and an additional aft-mounted axial thruster for the orbiter configuration. A bistable latch valve in the manifold feeding each thruster group permits isolation of that group in the event of propellant valve leakage. A pressure transducer in the liquid manifold provides a telemetered reading of subsystem internal pressure. Propellant flow is controlled by electrical signal to the torque motor-operated, dual-seat valve supplied with each thruster. The propellant decomposes in the thruster and the hot gases exhaust through the nozzle to produce the required thrust.

Propellant and pressurant are loaded through a single fill and drain valve, a concept proved successful on the Intelsat IV, Telesat, and Basic Bus Programs.

TABLE 1-2. SUBSYSTEM HARDWARE

Unit	Characteristics	Selection Criteria	Hardware Source
Tank	Conispherical design 6 AL 4V titanium Volume = 37690 cm^3 ($2300\ in^3$)	Propellant mass requirement Qualified for flight-on Marisat	Marisat
Thruster	22.2 N thrust (nominal) Shell 405 ABSG catalyst Predictable performance Hot restart capability	Thrust level provides low maneuver duration Small pulse I_{bit} provides maneuver granularity	Intelsat IV A
Latch valve	Bistable 11.9 cm x 3.18 cm dia (4.68 in x 1.25 in dia) Current drain 2.5 A @27 Vdc at 21°(70°F) Signal duration 0.050 sec	Existing, flight proven	Intelsat IV
Propellant valve	Torque motor operated Dual seat series redudnant	Valve and thruster qualified as an assembly; high reliability; flight-proven	Intelsat IV, Telesat, Basic Bus
Pressure transducer	Potentiometer type continuous power drain 0.25 W at 28 Vdc	Existing, flight-proven component	Intelsat IV, Telesat
Fill and drain valve	Titanium body Carbide ball seat Redundant seat	Existing, flight-proven component	Intelsat IV, Telesat, Basic Bus
Filter	Etched disc element titanium body 10μ filtration rating	Existing, flight-proven component	Intelsat IV, Telesat, Basic Bus
Tubing	6 AL 4V titanium 0.64 cm (0.25 in) dia 0.05 cm (0.020 in) wall thickness	Existing, flight-proven design	Intelsat IV, Telesat, Basic Bus
Fittings	6 AL 4V titanium Forgings	Existing, flight-proven component	Intelsat IV, Telesat, Basic Bus
Plenum	6 AL 4V titanium Volume = 116.3 cm^3 ($7.1\ in^3$) 32.56 cm x 2.69 cm dia (12.82 in x 1.06 in dia)	Existing, flight-proven component	Intelsat IV
Temperature sensors	0.09W @ 28 Vdc	Existing, flight-proven component	Intelsat IV, Telesat, Basic Bus
Propellant valve heater	Selectable power (0.5 or 1.25 W) Molded to fit valve	Existing, flight-proven design modified for power requirement	Intelsat IV, Telesat, Basic Bus
Tank heater	5 W Molded to fit conical portion of tank	Existing, flight-proven design modified for power requirement	Basic Bus

The propulsion subsystem is basically an all-welded system fabricated from 6Al - 4V titanium (a technique developed on Intelsat IV and used successfully on all subsequent satellites.) Weld interfaces with some stainless steel components are accomplished with titanium to stainless steel diffusion bonded, coextruded transition joints. Thus, the only mechanical joints in the system are the redundantly sealed fill and drain valve and propellant control valves for each thruster.

All components have been flight proven on the Intelsat IV or Telesat satellite or qualified for flight on Intelsat IV or Marisat. A detailed summary of component derivation is presented in Table 1-2, along with component masses.

The subsystem uses passive thermal control by placing components and interconnecting manifolds under insulation blankets. Heaters are required on each thruster valve as well as on each tank to maintain the respective temperatures above the allowable minimums.

The baseline orbit insertion motor for the Atlas/Centaur, Type II trajectory mission consists of a Thiokol Model TE-M-521 motor modified for the required propellant load. The propellant and expended insert mass required to place the spacecraft into Venus orbit is 143.3 Kg (316.0 lb), requiring a lengthening of the cylindrical portion of the titanium motor case by 13.0 cm (5.1 in), with corresponding modification to the internal motor insulation and nozzle design to accommodate the increased mass flow. Additionally, the high density TP-H-3135 propellant replaces the lower energy TP-H-3135 propellant to provide a performance advantage.

A summary of propulsion and orbit insertion motor subsystem hardware is presented in Table 1-2.

2. INTRODUCTION

The orbiter and probe bus spacecraft require the capability for maneuvering during cruise, planetary encounter, orbit insertion, and on-orbit mission phases.

Two separate subsystems are provided to perform these maneuvers. The propulsion subsystem provides the impulse required for all maneuvers involving attitude control, cruise trajectory corrections, spin speed changes, and adjustments to the spacecraft orbit about Venus. A separate subsystem provides the impulse for the orbit insertion maneuver.

Analyses were performed in accordance with the NASA/ARC statement of work, to define viable subsystem designs capable of meeting mission requirements. These analyses were originally intended to cover the spacecraft configurations and mission requirements associated with the Thor/Delta launch vehicle. They were expanded to include the effects of utilizing the Atlas/Centaur launch vehicle.

The format of this report is designed to present the discussions of the propulsion and orbit insertion subsystems separately. Each discussion begins with the presentation of applicable subsystem maneuver requirements, and concludes with descriptions of the baseline designs applicable to missions using Thor/Delta and Atlas/Centaur launch vehicles, respectively.

3. PROPULSION SUBSYSTEM

3.1 SUBSYSTEM REQUIREMENTS

The propulsion subsystem must provide the necessary impulse to perform spacecraft maneuvers.

For the multiprobe mission, the spacecraft maneuvers consist of velocity increments to correct trajectory and to target the probes; precession maneuvers to control the spacecraft attitude; and spin-up/down maneuvers to control the spacecraft spin-rate (including, in the case of the Atlas/Centaur launched mission, initial spacecraft spinup after separation from the launch vehicle). The maneuver requirements are summarized in Tables 3-1 and 3-2. The baseline missions considered for this study are:

1977 launch, Type I trajectory (Thor/Delta)

1978 launch, Type I trajectory (Atlas/Centaur)

For the orbiter mission, cruise-phase maneuvers similar to the multiprobe mission are required. Maneuvers unique to the orbiter mission consist of orbit insertion (see Section 4) precession maneuvers to properly orient the spacecraft for orbit insertion and post-insertion attitude control; spin speed change to slow the spin rate for science measurement, and velocity increments to correct for orbit insertion dispersion errors, modify periapsis altitude, and counteract in-orbit perturbations caused by solar, aerodynamic and other forces. The maneuver requirements are summarized in Tables 3-3 and 3-4. The baseline missions considered for this study are:

1978 launch, Type II trajectory, north periapsis (Thor/Delta)

1978 launch, Type II trajectory, south periapsis (Atlas/Centaur)

A thrust range of 4.4 to 35.6 N (1 to 8 lb) is required to perform maneuvers in a reasonable time and still provide the necessary granularity. In order to support the required maneuvers, thrusters must be capable of continuous and pulsing operation.

The vibration environments for shelf-mounted components and propellant tanks are shown in Table 3-5.

TABLE 3-1. THOR/DELTA LAUNCH VEHICLE MULTIPROBE MISSION

1977 - Type I

Maneuver	Velocity Change m/sec	Spin Speed Change rpm	Precession deg
Initial despin		40	
Initial reorientation			70
Precession			94
First TCM	71.9		
Precession			94
Precession			45
Second TCM	4.2		
Precession			45
Third TCM	0.4		
Fourth TCM	0.1		
Despin		30	
Precession			127
Despin		15	
(Large probe release)			
Spinup		56.2	
Precession			24
Target small probes	5.7		
(Small probe release)			
Despin		11.2	
Precession			53
Precession			83
Target probe bus	16.9		
Precession			19
Velocity change	1.0		
Totals	100.2	152.4	654

TABLE 3-2. ATLAS/CENTAUR LAUNCH VEHICLE MULTIPROBE MISSION

Maneuver	Velocity Change m/sec	Spin Speed Change rpm	Precession deg
Initial spinup		25	
Precession			95.0
First TCM	11.4		
Precession			95.0
Second TCM	0.6		
Third TCM	0.1		
Despin to 15 rpm		10	
Precession			127.0
(LARGE PROBE RELEASE)			
Precession			24.0
Spin up to 57 rpm		42	
Target small probes	5.7		
(SMALL PROBE RELEASE)			
Despin to 20 rpm		37	
Precession			53.0
Precession			83.0
Target spacecraft	16.9		
Precession			12.0
Velocity change	1.0		
Precession			7.5
Totals	35.7	114	496.5

TABLE 3-3. THOR/DELTA LAUNCH VEHICLE ORBITER MISSION

Maneuver	Velocity Change m/sec	Spin Speed Change rpm	Precession deg
Initial despin		40	
Initial Reorientation			80
Precession			94
First TCM	71.7		
Precession			94
Despin		30	
Precession			45
Second TCM	6.3		
Precession			45
Third TCM	0.3		
Fourth TCM	0.1		
Reorientation for OIM			207
(Orbit insertion)	1069.8)*		
Precess to orbit attitude			207
Correct insertion errors	26.0		
Despin to 5 rpm		25	
Adjust periapsis	13.0		
Maintain orbit	88.0		
Correct attitude drift			285
Totals	205.4	95	1057

*Not included in Totals

TABLE 3-4. ATLAS/CENTAUR LAUNCH VEHICLE ORBITER MISSION

Maneuver	Velocity Change m/sec	Spin Speed Change rpm	Precession deg
Initial spinup		25	
Precession			95
First TCM	11.4		
Precession			95
Second TCM	0.6		
Third TCM	0.1		
Despin to 15 rpm		10	
Reorientation for OIM			56
(Orbit insertion	1072.0)*		
Precess to orbit attitude			124
Correct insertion errors	28.0		
Despin to 5 rpm		10	
Adjust periapsis	13.0		
Maintain orbit	145.0		
Correct attitude drift			285
Totals	198.1	45	655

*Not included in totals

TABLE 3-5. PROPULSION SUBSYSTEM QUALIFICATION VIBRATION LEVELS (ATLAS/CENTAUR)

Sinusoidal (Frequency sweep at 2 oct/min)			
		Acceleration, g, 0-peak	
Axis	Frequency Hz	Thrusters and Shelf-Mounted Components	Propellant Tanks
Thrust	5 to 15	2.9	2.3
	15 to 20	10.0	8.0
	20 to 30	20.0	10.0
	30 to 60	55.0	20.0
	60 to 80	20.0	10.0
	80 to 150	10.0	8.0
	150 to 400	3.75	3.75
	400 to 2000	7.5	7.5
Lateral	5 to 13	2.5	2.5
	13 to 20	7.0	6.0
	20 to 40	4.0	4.0
	40 to 150	12.0	25.0
	150 to 400	3.75	3.75
	400 to 2000	7.5	7.5

Random (Test duration = 4 min/axis)			
Thrusters, Tanks, and Shelf-Mounted Components			
Axis	Frequency Hz	PSD Level g^2/Hz	g_{rms}
Thrust	20 to 60	+13 dB/oct	17.8
	60 to 150	1.0	
	150 to 260	-13 dB/oct	
	260 to 2000	0.1	
Lateral	20 to 150	0.20	14.7
	150 to 300	-3 dB/oct	
	300 to 2000	0.10	

TABLE 3-5. PROPULSION SUBSYSTEM QUALIFICATION VIBRATION LEVELS (THOR/DELTA) (Continued)

Sinusoidal (Frequency sweep at 2 oct/min)			
		Acceleration, g, 0-peak	
Axis	Frequency Hz	Thrusters and Shelf-Mounted Components	Propellant Tanks
Thrust	5 to 15	2.9	2.3
	15 to 21	10.0	8.0
	21 to 55	5.0	4.0
	55 to 85	40.0	15.0
	85 to 150	10.0	5.5
	150 to 400	4.5	4.5
	400 to 2000	7.5	7.5
Lateral	5 to 13	2.5	2.5
	13 to 25	5.0	4.0
	25 to 55	1.5	1.7
	55 to 150	8.0	20.0
	150 to 400	4.5	4.5
	400 to 2000	7.5	7.5

Random (Test duration = 4 min/axis)			
Thrusters, Tanks, and Shelf-Mounted Components			
Axis	Frequency Hz	PSD Level g^2/Hz	g_{rms}
Thrust	20 to 60	+13 dB/oct	17.8
	60 to 150	1.0	
	150 to 260	-13 dB/oct	
	260 to 2000	0.1	
Lateral	20 to 150	0.2	14.7
	150 to 300	-3 dB/oct	
	300 to 2000	0.1	

TABLE 3-6. SUMMARY OF TRADE STUDY RESULTS

Task No.	Trade Study	Options Considered	Conclusions
PP 1	Pressure supply system and Pressurant	Blowdown subsystem Regulated pressure subsystem Helium versus nitrogen gas	Blowdown design selected — low cost, low mass, high reliability Helium selected — low mass
PP 2	Propellant management techniques	Centrifugal force Positive expulsion Surface tension device	Centrifugal force selected — low mass, low cost, simplicity
PP 3	Propellant tradeoff	Cold gas Hot gas Liquid bipropellant Liquid monopropellant Electric	Monopropellant hydrazine selected for propulsion subsystem — low mass, simplicity, available hardware Solid motor selected for orbit insertion — low cost, available technology
PP 4	Thruster arrangement	Thruster size 22.2 N (5 lb_f) 4.45 N (1 lb_f) Number of thrusters Thruster location	22.2 N (5 lb_f) thruster selected — minimize maneuver time, adequate maneuver granularity Probe bus — 2 axial, 4 radial Orbiter — 3 axial, 4 radial Minimize nutation Maneuver redundancy
PP 5	Propellant utilization effect on center of mass		Center of mass excursions defined. Constrained to movement along spin axis (no radial excursion)
PP 6	Operational flexibility		Operational plans developed
PP 7	Power and thermal insulation requirements	Passive versus active thermal control Heater power requirements	Thermal blankets cover all components and interconnecting tubing Thruster valve heaters required 1.25 watts - axial 0.50 watt - radial Orbit insertion motor Decouple from thrust tube with Kapton superinsulation Cover exposed nozzle with 2 mil aluminized Teflon 5 W throat heater (Thor/Delta)
PP 8	Latch valve arrangement	Number of latch valves Valve arrangement	Two latch valves required Thrusters divided into two groups (one group per latch valve for reliability)
PP 9	Thruster operating requirements		Pulsing and continuous thruster operation defined as a function of applicable mission Number of starts Continuous duration Number of pulses Pulse duty cycle Nominal and off nominal missions, redundant mode operation evaluated Comparison of requirements with thruster qualification status
PP 10	Orbit insertion motor selection	Thor/Delta Type I trajectory Type II trajectory Atlas/Centaur Type I trajectory Type II trajectory	Thor/Delta (Type II): TE-M-521 motor, (off-loaded) Atlas/Centaur (Type I): TE-M-616 motor (Type IIS): TE-M-521 motor (stretched)

General design goals are to minimize subsystem mass in addition to being simple and reliable. An important goal is to use as many existing, flight-qualified components as possible to minimize program development and qualification requirements.

3.2 SUBSYSTEM TRADES

This study considers the designs available to meet the requirements of the multiprobe and orbiter missions for Thor/Delta and Atlas/Centaur launched spacecraft.

A series of trade studies was conducted to determine the most suitable propellant for these missions; to select a simple, low-mass propellant management technique and subsystem design; and define the effect of propellant utilization on the spacecraft center of mass throughout the mission. The thruster arrangement on the spacecraft was studied to provide redundant capability for the required maneuvers with a minimum number of components without compromising operational simplicity. Upon selecting the thruster arrangement, heater power and thermal insulation, maneuver cross coupling effects, and thruster operating requirements were defined.

The criteria used to rank tradeoff alternatives included low subsystem mass; use of existing, flight-qualified hardware and design concepts; and subsystem design simplicity (low cost and high reliability). The results of the studies performed are discussed in the following sections. A summary of pertinent conclusions is presented in Table 3-6.

Pressure Supply System and Pressurant (Task PP-1)

A tradeoff of a regulated versus a blowdown pressure supply was conducted for the monopropellant hydrazine propulsion subsystem selected in the PP-3 Propellant Trade Study (described in a following section). Subsystem designs were defined for Thor/Delta and Atlas/Centaur launched missions utilizing existing, available components which are or will be flight-qualified prior to use on the Pioneer Venus program. The blowdown design was selected because it represented the lowest mass, highest reliability, and lowest cost approach for all missions.

Thruster performance is represented by I_{sp} and varies as a function of subsystem pressure. As shown in Figure 3-1, I_{sp} decreases only slightly with pressure, thus the mission average I_{sp} for the blowdown subsystem is very nearly equal to the maximum available performance.

A comparison of subsystem mass for the two design approaches is shown in Figure 3-2. The mission maneuver requirements, governing propellant loads, have been refined since the completion of this trade study; however, the advantage, based on the lower subsystem mass, of the blowdown

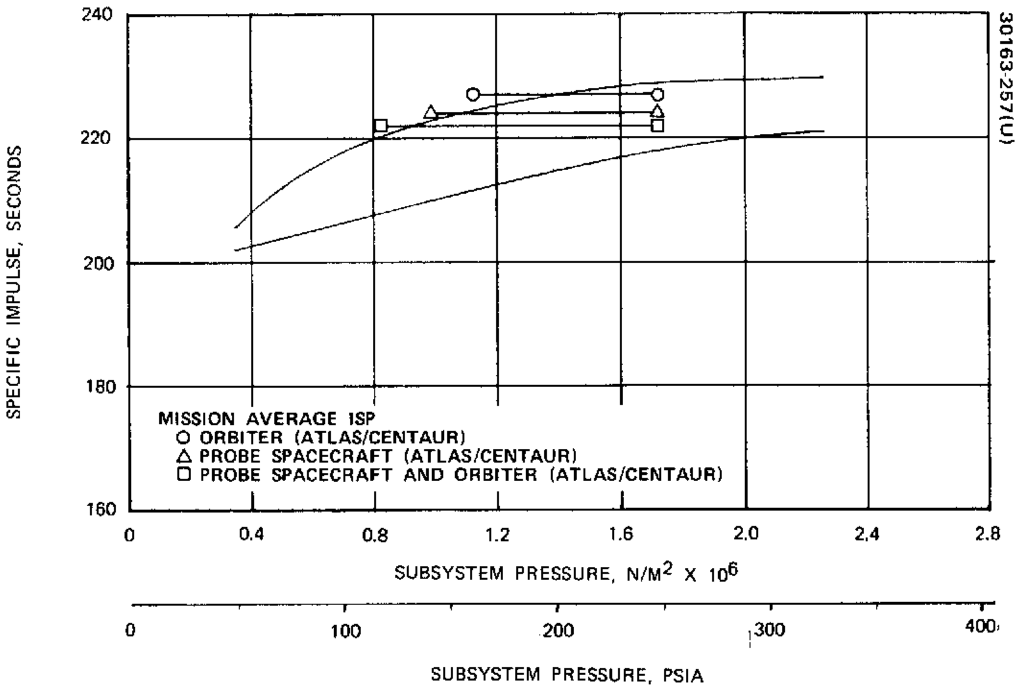


FIGURE 3-1. THRUSTER PERFORMANCE SPECIFIC IMPULSE VERSUS SUBSYSTEM PRESSURE

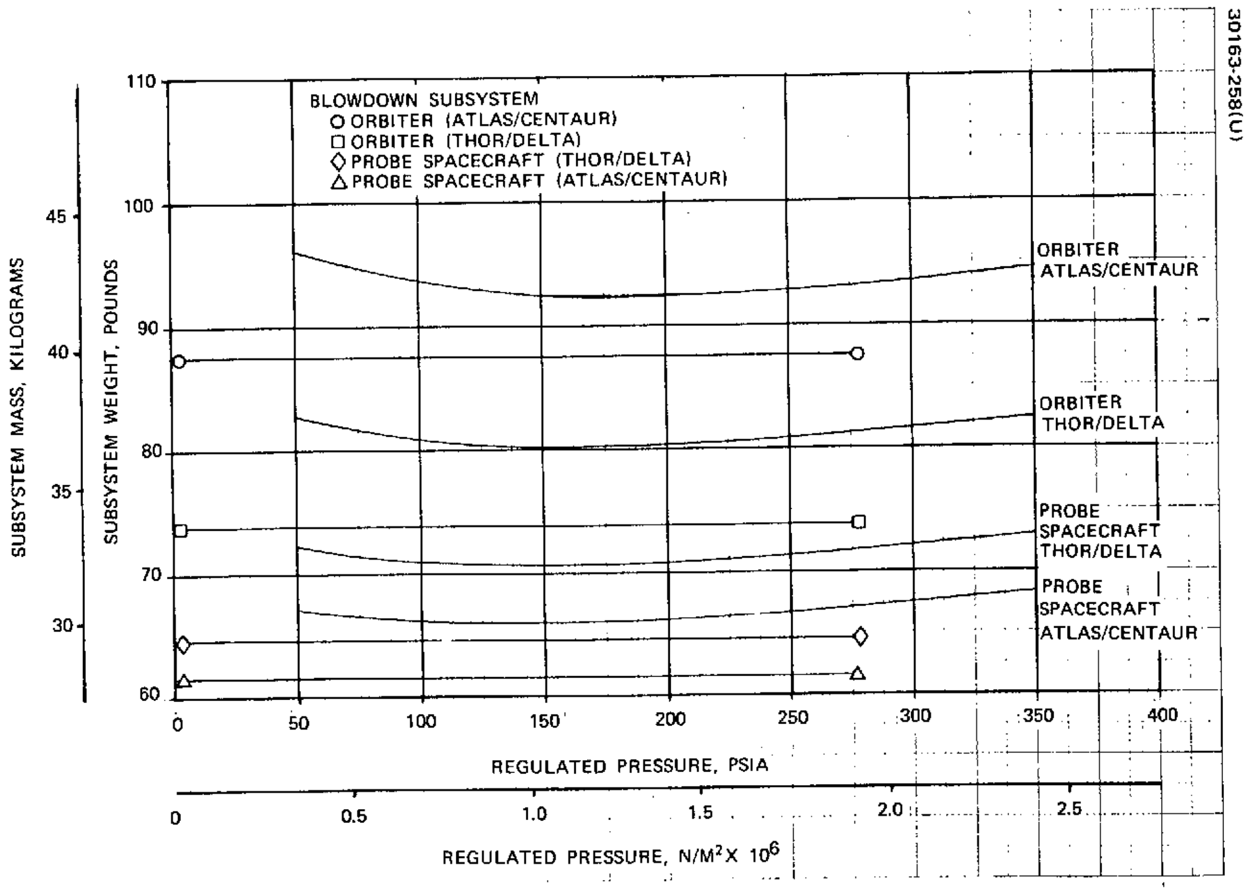


FIGURE 3-2. SUBSYSTEM MASS VERSUS REGULATED PRESSURE

TABLE 3-7. REGULATED PRESSURE VERSUS PRESSURE BLOWDOWN PROPULSION SUBSYSTEM RELIABILITY COMPARISON

	Thor/Delta		Atlas/Centaur	
	Regulated	Blowdown	Regulated	Blowdown
Probe	0.991	0.998	0.989	0.996
Orbiter	0.968	0.989	0.973	0.991

TABLE 3-8. REGULATED PRESSURE VERSUS PRESSURE BLOWDOWN PROPULSION SUBSYSTEM PROGRAM COST COMPARISON

	Cost Increment of Regulated Subsystem Relative to Corresponding Blowdown Subsystem	
	Thor/Delta	Atlas/Centaur
Probe[1]	+ $45.7K	+ $43.7K
Orbiter	+ $11.4K	+ $ 8.6K
Total program	+ $57.1K	+ $52.3K

(1) Includes spares and non-flight subsystems

design remains unchanged. A comparison of the subsystem reliability and costs is shown in Tables 3-7 and 3-8, respectively, and further supports selection of the blowdown design for these missions. It should be noted that the reliability models used to obtain Table 3-7 were preliminary at this point in the study. The models were developed more fully during Study Task PP-8.

An additional consideration of this study was the comparison of nitrogen and helium gases as the pressurizing agent.

Helium was selected due, primarily, to the advantage of its lower mass. Other considerations such as gas solubility in hydrazine propellant and leakage from the subsystem were rated equal for the two gases. The selection of helium as the pressurizing gas results in the following mass savings:

	Thor/Delta	Atlas/Centaur
Probe spacecraft	0.32 kg (0.7 lb)	1.18 kg (2.6 lb)
Orbiter spacecraft	0.23 kg (0.5 lb)	0.95 kg (2.1 lb)

Propellant Management Technique (Task PP-2)

A tradeoff was performed to determine the most desirable technique for controlling the orientation and location of propellant as well as separation of propellant and pressurization gas within the common propellant/pressurant storage tanks. The techniques considered were:

1) Use of centrifugal forces due to spacecraft rotation

2) Use of bladders or diaphrams

3) Use of surface tension devices

The centrifugal force technique requires no additional components, thus it is lighter and simpler than the diaphram design and less costly than the surface tension device. Centrifugal force has been used successfully for propellant settling for many years on all Hughes' earth satellites.

Applying this technique to the Pioneer Venus missions required analysis of the various mission phases. The most sensitive condition is encountered during the Venus orbit with the spacecraft spinning at 5 rpm.

To maintain proper propellant orientation, the centrifugal acceleration available for propellant settling must dominate the surface tension forces and adversely directed external accelerations. The ratio of external forces to surface tension forces is represented by the Bond number and is used to determine the acceleration required for propellant settling, shown in Figure 3-3 as a function of propellant mass remaining in each tank.

The available settling acceleration is shown in Figure 3-4 as a function of spin speed for several propellant masses. It is easily seen that the available acceleration exceeds the required setting acceleration for propellant load greater than 0.014 kg (0.03 lb.) per tank. Since this load occurs at essentially the end of the orbiter mission, a positive margin exists throughout the mission and no propellant unsettling is expected. The available margin is shown in Figure 3-5 as a function of remaining propellant.

During the Venus orbit, the expected external accelerations due to thruster firings will consist of a pure couple during attitude control maneuvers, and axial thrust (parallel to the spin axis) to compensate for aerodynamic drag and solar perturbations. Since the axial thrust will be provided by the thrusters at the orbit-insertion-motor-end of the spacecraft, the resulting acceleration is not adversely directed with respect to the tank orientation and will, in fact, help to further settle the propellant.

Another mission phase requiring consideration is the initial spacecraft spinup after separation from the launch vehicle. This condition is unique to the Atlas/Centaur launch vehicle and was not discussed in the study task report, since the Thor/Delta mission used spin table spinup prior to separation and the Atlas/Centaur spinup requirements were not defined at that time.

During the period immediately following separation of the spacecraft from the Atlas/Centaur launch vehicle, the spacecraft exists as a free body with no significant gravitational forces acting on it except for low level accelerations associated with separation tip-off rotations.

Spacecraft spinup will be performed by operating the spinup thrusters. A volume of propellant, sufficient to provide an interim spacecraft spin rate for propellant settling, is located in plenums upstream of each spinup thruster. Availability of this propellant for use is assured by maintaining the latching valves closed during the launch-boost phase. After operating the thrusters for sufficient time to use the propellant provided in the plenums, the thrusters are shut down. The spacecraft is allowed to dwell at the achieved spin rate to assure proper propellant orientation in the tanks and the spinup thrusters operated again to provide the required nominal spin rate. The operating sequence for this maneuver is as follows:

1) Operate spinup thrusters (two) for approximately 8 sec to use up the propellant in the plenums

2) Dwell at the resulting spin rate for 40 sec to settle the propellant in the tanks

3) Operate spinup thrusters (two) to obtain the nominal spin speed of 25 rpm

The interim spin rate during the dwell mode is 5.3 rpm for the probe spacecraft and 7.8 rpm for the orbiter spacecraft. The estimated time required to settle propellant in the tank is 3.7 sec from a randomly oriented propellant mass, with only one spinup thruster operating (a failure mode condition). Thus, allowing 40 sec for propellant settling provides a settling time margin which is more than adequate. This technique has been used successfully in four Intelsat IV launches.

Propellant Tradeoff (Task PP-3)

The review of available propulsion subsystems, using various state of the art propellants was conducted. The factors considered were:

1) Subsystem weight
2) Complexity
3) Reliability
4) Availability
5) Thermal control requirement
6) Operation (continuous and pulsing)

The total impulses required for the Pioneer Venus multiprobe and orbiter missions were on the order of 48,900 to 71,200 N-sec (11,000 to 16,000 lb-sec). The thrust range of 4.4 to 35.6 N (1 to 8 lb) was established. The upper thrust limit was established to assure control granularity for maneuvers. The lower limit was based on the time required to perform maneuvers; constrained due to thermal considerations associated with the spacecraft attitudes required for midcourse, orbit insertion, and other maneuvers.

The basic subsystems considered were:

1) Cold gas
2) Heated gas
3) Liquid monopropellant
4) Liquid bipropellant
5) Electric propulsion

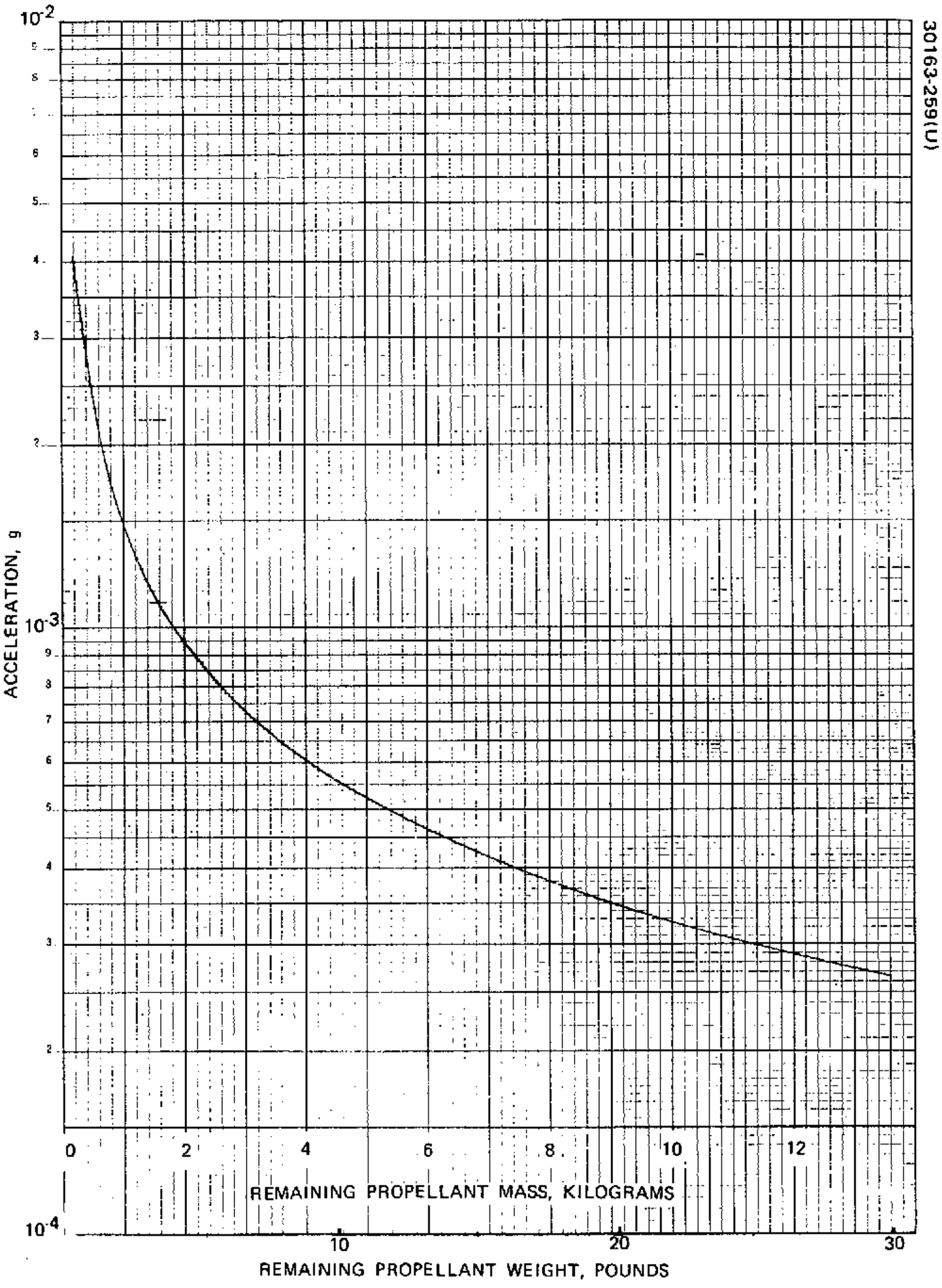


FIGURE 3-3. ACCELERATION REQUIRED VERSUS PROPELLANT MASS

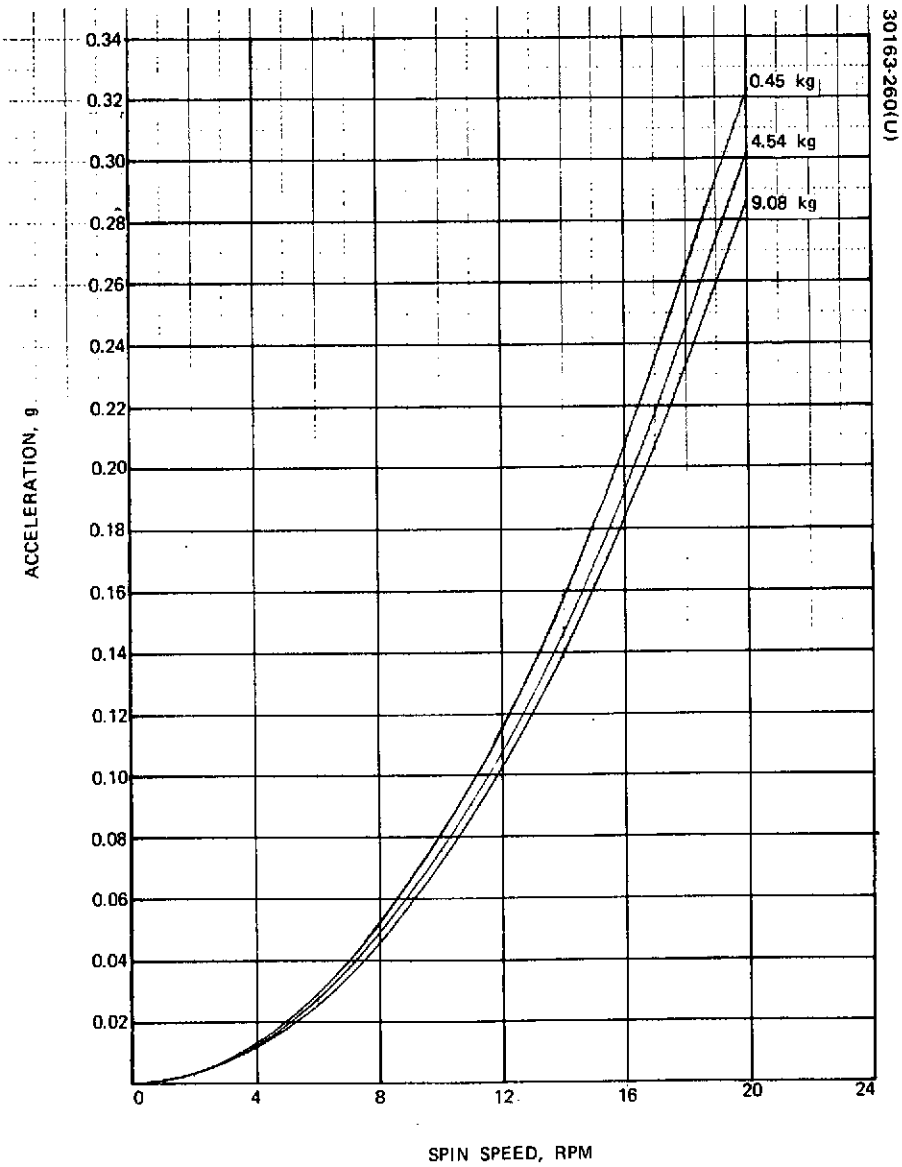


FIGURE 3-4. AVAILABLE SETTLING ACCELERATION VERSUS SPIN SPEED

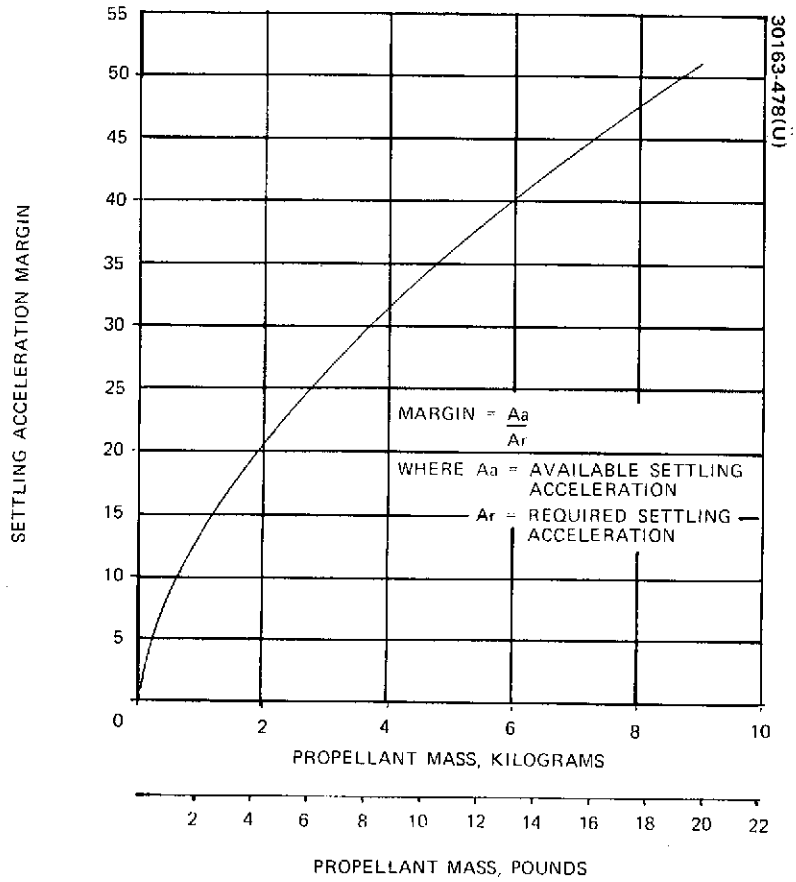


FIGURE 3-5. PROPELLANT SETTLING ACCELERATION MARGIN VERSUS PROPELLANT MASS REMAINING IN TANK

The approximate combined mass of the propellant, tankage and thrusters for each of these subsystems is shown in Table 3-9. The selection of the monopropellant hydrazine subsystem resulted from elimination of the other subsystems based on the above-listed criteria. A summary of these conclusions is as follows:

- Cold gas

  Too heavy due to low performance and high pressure storage requirements

  Too susceptible to leakage problems

- Heated gas

  Nitrogen

    Too heavy due to low performance and high pressure storage requirements

    Too susceptible to leakage problems

  Ammonia

    Too complex (control of 815°C (1500°F) gas)

    Reliability (gas generator failure causes loss of entire subsystem)

- Liquid bipropellant

  Complexity (separate tanks, lines, valves; control of mixture ratio variations)

  No operational thrusters available in required thrust range

- Electric propulsion

  High electric power requirement

  Inability to operate in pulsing mode

  No present operational thruster in required thrust range

- Hydrogen peroxide (monopropellant)

  Lower performance than hydrazine

  Limited materials for subsystem construction

  More difficult subsystem passivation

TABLE 3-9. APPROXIMATE PROPULSION SUBSYSTEM WEIGHT FOR THE CANDIDATE PIONEER VENUS PROBE AND ORBITER SPACECRAFT PROPULSION SUBSYSTEMS

Subsystem Type	Mass, kg (lb)	
	Probe, Inert Propellant Total	Orbiter
Cold gas:		
N_2	136.1 (300)	226.8 (500)
CH_4	99.8 (220)	136.1 (300)
Heated gas:		
N_2 (1500°F)	99.8 (220)	136.1 (300)
NH_3 (1500°F)	34.1 (75)	40.8 (90)
Monopropellant:		
N_2H_4	34.1 (75)	43.1 (95)
H_2O_2	40.8 (90)	52.2 (115)
Bipropellant:		
MMH-N_2O_4	31.8 (70)	40.8 (90)
Electric:		
Not including required power supply	4.5 (10)	5.4 (12)

Thruster Arrangement Tradeoff (Task PP-4)

The maneuvers required for the Pioneer Venus missions may be performed by thrusters placed on the spacecraft in any of several arrangements. This study first considered the constraints on the mission maneuvers, Table 3-10. The fundamental thruster arrangements were then evaluated in building-block fashion to determine the pure maneuver capability of each, as well as the cross coupling effect into related maneuver modes. Each arrangement was additionally evaluated to determine the effects of thruster misalignment, the loss of one thruster on the pure maneuver, and the effects on power and thermal control on maneuvers performed with each arrangement. A summary of this evaluation is presented in Table 3-11.

Based on the building-block evaluation, thruster arrangements were selected consisting of two axial and four radial thrusters for the probe spacecraft and three axial and four radial thrusters for the orbiter spacecraft as shown in Figure 3-6. The additional axial thruster on the orbiter spacecraft is required to minimize the induced nutation during maneuvers while spinning at 5 rpm in Venus orbit, and to provide redundancy for the primary maneuver mode.

Radial thrusters are placed in pairs at the spacecraft center of mass locations associated with selected mission times; with 180 deg displacement about the spin axis between each pair. When pulsed as a pair in phase with the spin rate, a radial velocity increment is generated. Operating one thruster of a pair in a continuous mode provides either spinup or spindown, depending on the thruster selected. Using two pairs of thrusters provides redundant capability for both of these maneuver types. Different station numbers of radial thruster pairs provides a backup mode for performing precession maneuvers.

Axial thrusters are located at both ends of the spacecraft and provide thrust vectors parallel to and offset from the spin axis. These thrusters on the probe spacecraft are located 180 deg with respect to one another about the spin axis and generate a pure couple when pulsed simultaneously. This is the primary operating mode for precession maneuvers. A redundant capability is available by pulsing either axial thruster singly. Velocity increments in the axial direction are available by operating the appropriate axial thrusters in the continuous mode.

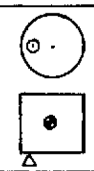
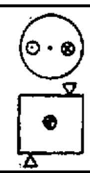
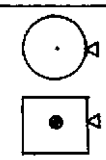
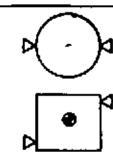
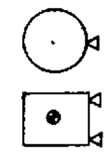
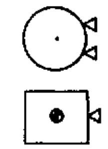
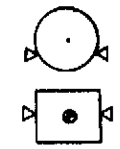
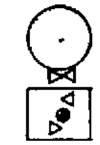
It is intended that large ΔV maneuvers (greater than 10 m/sec) be performed with axial thrusters to take advantage of the higher performance associated with continuous mode operation. Small ΔV maneuvers will be performed by either radial or vector mode maneuvers. These maneuver types are described in Figure 3-7 and they afford ample maneuver redundancy.

Orbital ΔV corrections require special consideration due to reduced attitude control stiffness at the lower spin speed. Initial orbital period and periapsis altitude corrections may be performed with axial thrusters at either end of the spacecraft prior to despinning to 5 rpm spin speed. During normal

TABLE 3-10. MISSION CONSTRAINTS ON MANEUVERS

Midcourse	Required ΔV at arbitrary attitude, to be executed to within 4 percent magnitude and 4 deg directional error
Spin speed (Thor/Delta)	Third stage injection at 90 rpm, large probe separation at 15 rpm, small probe separation at 71 rpm, orbiter insertion at 30 rpm, orbital operations at 5 rpm
Attitude	Spin axis never within 15 deg sunlin; minimize ΔV imparted by large reorientations after final midcourse (<0.1 m/sec)
Targeting	Small probe targeting ΔV perpendicular to small probe separation attitude; probe bus retarding ΔV nearly parallel to probe bus entry attitude
Orbital insertion	N_2H_4 preburn in spin axis direction at orbital insertion
Orbital	Required ΔV always in spin axis direction (ecliptic normal); limit attitude error to <2 deg and nutation to <3 deg after ΔV correction at 5 rpm

TABLE 3-11. THRUSTER ARRANGEMENT TRADEOFF EVALUATION

Parameter \ Fundamental Jet Schemes									
Pure maneuver modes	None	Precession (plus nutation)	Axial ΔV	Radial ΔV	Precession (plus nutation)	Radial ΔV	Radial ΔV, spinup, spindown	Spinup, spindown	Precession (plus nutation)
Compound maneuver modes	1) Precession plus axial ΔV 2) Axial ΔV plus residual nutation plus attitude error	1) Precession plus axial ΔV (two ways) 2) Bidirectional axial ΔV plus residual nutation plus attitude error	1) Precession plus axial ΔV (two ways) 2) Axial ΔV plus plus attitude error (two ways)	None	1) Precession plus radial ΔV (two ways) 2) Radial ΔV w/ 180 deg mode, band-bang precession	1) Precession plus radial ΔV (two ways)	1) Radial ΔV plus spinup 2) Radial ΔV plus spindown	1) Radial ΔV w/ 180 deg mode, alternate spinup and spindown 2) Radial ΔV plus spinup 3) Radial ΔV plus spindown	1) Radial ΔV plus spinup plus precession 2) Radial ΔV plus spindown and precession 3) Spinup plus attitude error plus residual nutation 4) Spindown plus attitude error plus residual nutation 5) Radial ΔV, 180 deg mode
Dominant effect of thruster misalignment and/or mismatch on pure maneuver	-----	Spin speed change	Spin speed change, bounded attitude error	Secular precession, spin speed change	Spin speed change	Secular precession, spin speed change	Secular precession, spin speed error	Bounded attitude error	Spin speed change
Effect of one thruster loss on pure maneuver	No capability	Compound mode[1] available	Compound mode[2] available	No capability	Compound mode[1] available	Compound mode[1] with substantial precession	Compound mode (1 or 2) with substantial spin change; loss of spinup or spindown capability	Loss of spinup or spindown capability	Compound mode (1 or 2) with radial ΔV and substantial spin change
Primary characteristics for ΔV maneuvers	• Maximum 180 deg initial precession • Continuous • Less time • Less propellant • One jet sufficient for entire maneuver • Bounded attitude error	• Maximum 90 deg initial precession • Continuous • Less time • Less propellant • One jet sufficient for entire maneuver • Bounded attitude error	• Maximum 180 deg initial precession • Continuous • Less time • Less propellant • One jet sufficient for entire maneuver • Minimum attitude perturbation during continuous firing	• Maximum 90 deg initial precession • Pulsed • 1/8 duty cycle • More propellant • Requires other jet for precession • Susceptible to secular precession	• Maximum 90 deg initial precession • Pulsed 180 deg mode • 1/4 duty cycle More propellant • Jet pair sufficient for entire maneuver • Susceptible to secular precession	• Maximum 90 deg initial precession • Pulsed • 1/4 duty cycle • More propellant • Jet pair sufficient for entire maneuver • Susceptible to secular precession	• Maximum 90 deg initial precession • Pulsed • 1/4 duty cycle • More propellant • Requires other jet for precession • Susceptible to secular precession	• Maximum 90 deg initial precession • Pulsed 180 deg mode • 1/4 duty cycle • More propellant • Requires other jet for precession • Susceptible to secular precession	• Maximum 90 deg initial precession • Pulsed 180 mode • 1/4 duty cycle • More propellant • Jet pair sufficient for entire maneuver • Susceptible to secular precession; most cross coupling
Effect on thermal control during ΔV maneuvers	Spin axis attitude near sunline (if required for first TCM) does not pose thermal control problem for near earth condition. For subsequent TCMs, thermal attitude constraints limit axial jet use to sun angles 90 ± 25/45 deg.			← Spin axis attitude 90 ±5 deg from sunline during TCMs does not pose any thermal control problem. →					
Effect on power during ΔV maneuvers	Battery supply adequate for short term spin axis attitudes near sunline during TCMs			← Negligible impact on solar panel power available. →					

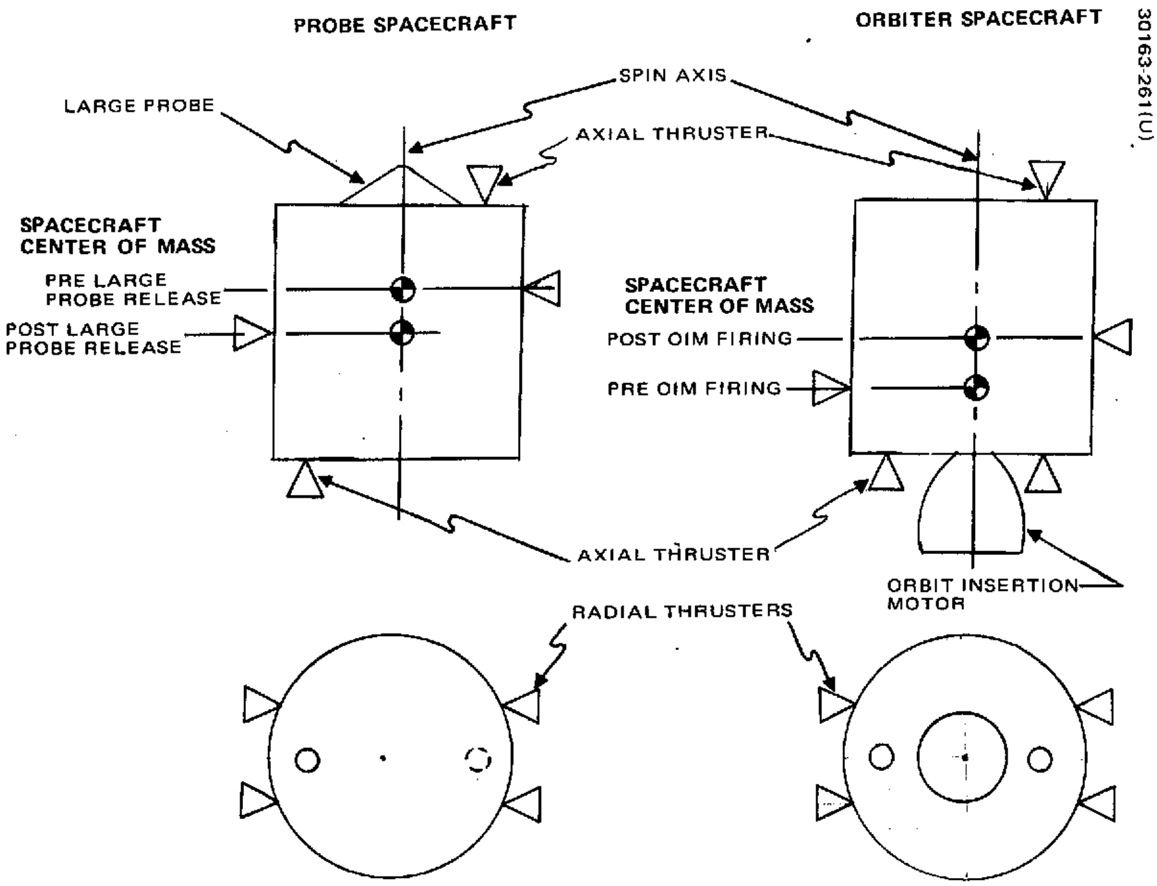


FIGURE 3-6. THRUSTER ARRANGEMENT SELECTION

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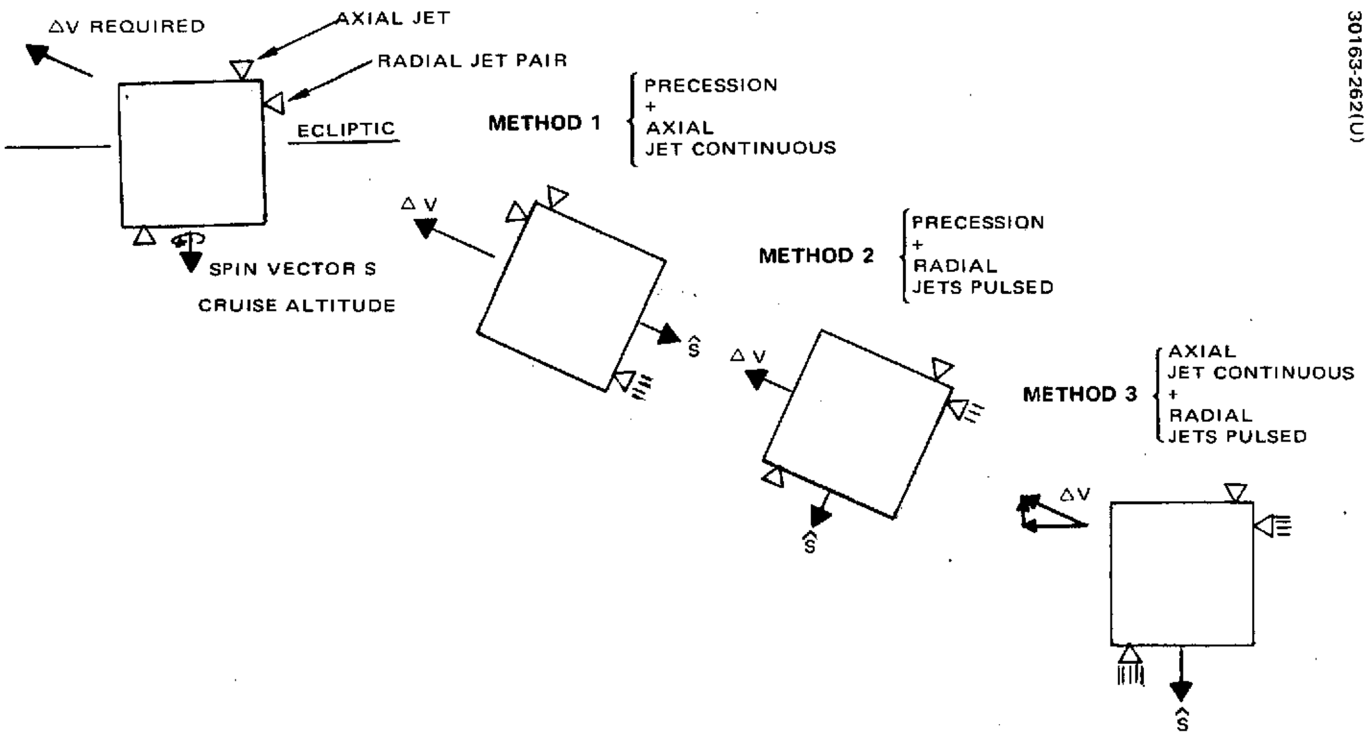


1 AXIAL JET MODE - AXIAL JET LOCATED AT BOTH ENDS OF SPACECRAFT; REQUIRES MAXIMUM 95 DEG OPEN LOOP PRECESSION PLUS AXIAL JET CONTINUOUS BURN; SPIN AXIS NEVER CLOSER THAN 15 DEG TO SUN LINE; MINIMUM PROPELLANT AND MANEUVER TIME'

2 RADIAL JET MODE - RADIAL JET PAIR THRUSTS THROUGH CRUISE CENTER OF MASS; REQUIRES MAXIMUM 15 DEG PRECESSION, PLUS RADIAL JET PAIR PULSED; USED FOR V DIRECTIONS NEAR SUNLINE; ATTITUDE AND SPIN SPEED MONITORED AND LARGE MANEUVER PARTITIONED IF NECESSARY.

3 VECTOR MODE - NO CHANGE IN ATTITUDE REQUIRED; AXIAL JET CONTINUOUS THRUST VECTORED WITH RADIAL JET PULSED THRUST; USED FOR SMALL MANEUVERS.

FIGURE 3-7. MIDCOURSE MANEUVER EXECUTION METHODS orbital operations at the low spin speed, compensation for solar and drag perturbations only requires axial thrust at the insertion motor end of the spacecraft (to decrease periapsis and increase apoapsis altitudes). The pair of axial thrusters at that end of the spacecraft provides balanced (nominally) thrust to execute these maneuvers with minimal attitude perturbations. Each of the pair of axial thrusters is redundant to the other when used in the 180 deg bang-bang mode (pulsing single thruster at twice spin frequency to limit attitude disturbances at 5 rpm).

The Hughes 22.2 N (5 lb_f) thruster was selected for use at all thruster locations. This thrust level satisfies the requirements to perform trajectory correction maneuvers in a reasonably short duration as well as provide the maneuver granularity necessary for orbital operations.

For typical Thor/Delta launched spacecraft with the selected thruster arrangements, predicted errors associated with typical maneuver sequences are presented in Tables 3-12 and 3-13. Atlas/Centaur mission analyses will be reported in subsection 3.5.

Maneuvers while in Venus orbit warrant separate consideration due to the low spin speed and its effect on spacecraft attitude control. Thruster operation and predicted maneuver response is summarized in Table 3-14.

Propellant Utilization Effect on Center of Mass (Task PP-5)

The multiprobe mission and orbiter mission for the Thor/Delta and Atlas/Centaur launched spacecraft were evaluated to define the movement of spacecraft center of mass as propellant is used for spacecraft maneuvers. Figure 3-8 present spacecraft longitudinal center of mass as a function of spacecraft mass. The center of mass does not move radially from the spin axis, since the propellant tanks are equidistant from the spin axis and propellant is used uniformly from both tanks. On the orbiter spacecraft, the orbit insertion motor center of mass is located on the spin axis so as to prevent the introduction of a radial excursion of the spacecraft center of mass.

Since the completion of this task, the Thor/Delta design was changed from a three-tank propulsion subsystem to a two-tank design. The tank locations, as stated above, were located equidistant from the spin axis and 180 deg apart, similar to the Atlas/Centaur propulsion subsystem design.

Operational Flexibility (Task PP-6)

Operational plans were developed for the propulsion subsystem through the various phases of fabrication, test, integration and launch preparation. An analysis was made of the capability for modification of propellant loads and pressure levels, realignment of thrusters, replacement of components, removal of particulate contamination, and passivation of materials in contact with hydrazine propellant.

A summary of operations and related lead times is presented in Table 3-15.

TABLE 3-12. MANEUVER ERRORS – MULTIPROBE MISSION

Mission Phase	Errors			
	Velocity Vector Error	Final Attitude, deg	Residual Nutation, deg	Spin Change, rpm
Initial despin	0.025 m/sec	0.036	0.098	1.32
Initial reorientation	0.10 m/sec	3.2	0.50	0.2
First midcourse, axial mode	3%, 3.4 deg	14.2	0.35	5.9
First midcourse, radial mode	3%, 5.2 deg	12.0	0.003	25.1
Second midcourse, axial mode	3%, 2.1 deg	2.8	0.35	0.33
Second midcourse, radial mode	3%, 3.5 deg	2.8	0.003	1.4
Third midcourse	5%, 0.5 deg	0.15	1.28	0.1
Large probe reorientation	0.09 m/sec	4.5	0.70	1.3
Large probe despin	0.011 m/sec	0.014	0.11	0.64
Small probe reorientation	0.075 m/sec	1.22	0.28	2.3
Small probe targeting	4.7%, 0.75 deg	0.51	0.002	1.3
Probe bus E-20 reorientation	0.13 m/sec	2.89	0.73	0.16
Probe bus targeting reorientation	0.17 m/sec	3.23	0.84	0.58
Probe bus retardation	3% 0.78 deg	0.66	0.47	0.96
Probe bus E-18 reorientation	0.023 m/sec	1.07	0.80	0.03
Probe bus final ΔV	3.2% 0.77 deg	0.64	0.066	0.057
Final reorientation	0.014 m/sec	0.93	0.80	0.018

TABLE 3-13. MANEUVER ERRORS – ORBITER MISSION

Mission Phase		Errors			
		Velocity Vector Error	Final Attitude, deg	Residual Nutation, deg	Spin Change, rpm
Initial despin		0.036 m/sec	0.039	0.054	1.4
Initial reorientation		0.10 m/sec	3.4	0.58	0.2
First midcourse	axial mode	3%, 3.4 deg	14.4	0.48	6.7
	radial mode	3%, 5.1 deg	13.3	0.004	27.8
Despin		0.018 m/sec	0.005	0.02	1.3
Second midcourse	axial mode	3%, 2.3 deg	2.8	0.76	0.6
	radial mode	3%, 2.8 deg	3.0	0.005	2.4
Third midcourse		4.0%, 1.8 deg	0.08	0.13	0.08
Reorientation for orbit insertion		0.10 m/sec	5.6	0.68	0.2
Reorientation for orbit operations		0.14 m/sec	5.6	0.70	0.2
Typical initial orbit correction		2.2%, 0.70 deg	0.01	0.08	0.2
Spindown for orbit operations		0.02 m/sec	0.23	0.37	1.05
Typical in-orbit ΔV (0.5 m/sec)		2.2%, 1.10 deg	1.7	1.9	0.04
Typical in-orbit attitude change		0.010 m/sec	0.17	0.51	0.0016

TABLE 3-14. ON-ORBIT MANEUVER PERFORMANCE

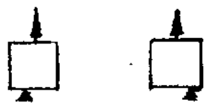
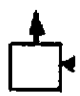
Maneuver	Mode	Comments	Performance
ΔV	Continuous 	Axial jet pair fired continuously	0. 152 m/sec/sec (0. 5 ft/sec/sec), 2. 2 deg maximum nutation (3 percent thrust mismatch)
	1 0 deg bang-bang 	Single axial jet pulsed	0. 0024 m/sec/pulse (0. 008 ft/sec/pulse), 0. 46 deg maximum nutation (30 ms pulse)
Attitude precession		Axial jet diagonal pair pulsed simultaneously at spin frequency	0. 84 deg/pulse, 0. 92 deg maximum nutation (30 ms pulse)
		Single axial jet pulsed at spin frequency	0. 42 deg/pulse, 0. 46 deg maximum nutation (30 ms pulse)
		Radial jet pair pulsed	$\pm$0. 05 deg/pulse, 0. 055 deg maximum nutation (30 ms pulse)
Spin speed		Single radial jet fired continuously	$\pm$0. 68 rpm/sec, 0. 5 maximum nutation

TABLE 3-15. OPERATIONAL FLEXIBILITY

Item	Lead Time
Propulsion Subsystem	
• Procurement	
1) Increase tank volume (new tank)	26 weeks
2) Increase number of tanks per subsystem	26 weeks
• Subsystem level (assembly and test)	
1) Repair substandard weld	1 day
2) Replace malfunctioning component	2 days
3) Subsystem flush	1 day
• Spacecraft level (test)	
1) Subsystem flush	1 day (latest flush ≥ 1 week prior to launch)
2) Replace malfunctioning component	1 to 1.5 weeks*
Orbit Insertion Motor Subsystem	
• Modify case (increase volume): Thiokol	14 months
Aerojet	12 months
• Reduce propellant load within 10 percent of nominal	3 months
• Reduce propellant load greater than 10 percent nominal:	
1) Modify igniter	1 month**
2) Modify case: Thiokol	10 months
Aerojet	10 months

*Varies dependent on degree of spacecraft disassembly required for access, and availability of X-ray and welding equipment. Shipment of X-ray and welding equipment required if replacement is to be done outside of Hughes facilities.

**Assumes modification is done prior to motor acceptance firing.

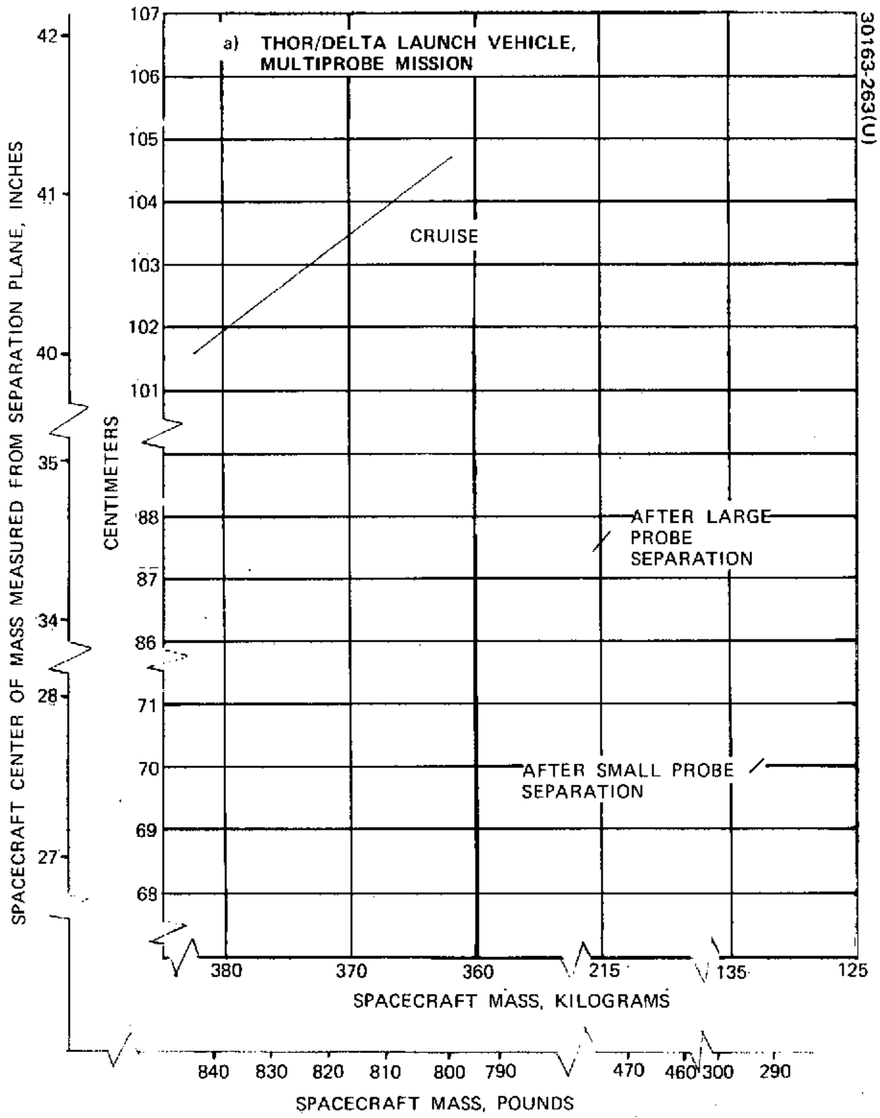


FIGURE 3-8. SPACECRAFT BUS MASS AND CENTER OF MASS CHANGE VERSUS PROPELLANT UTILIZATION

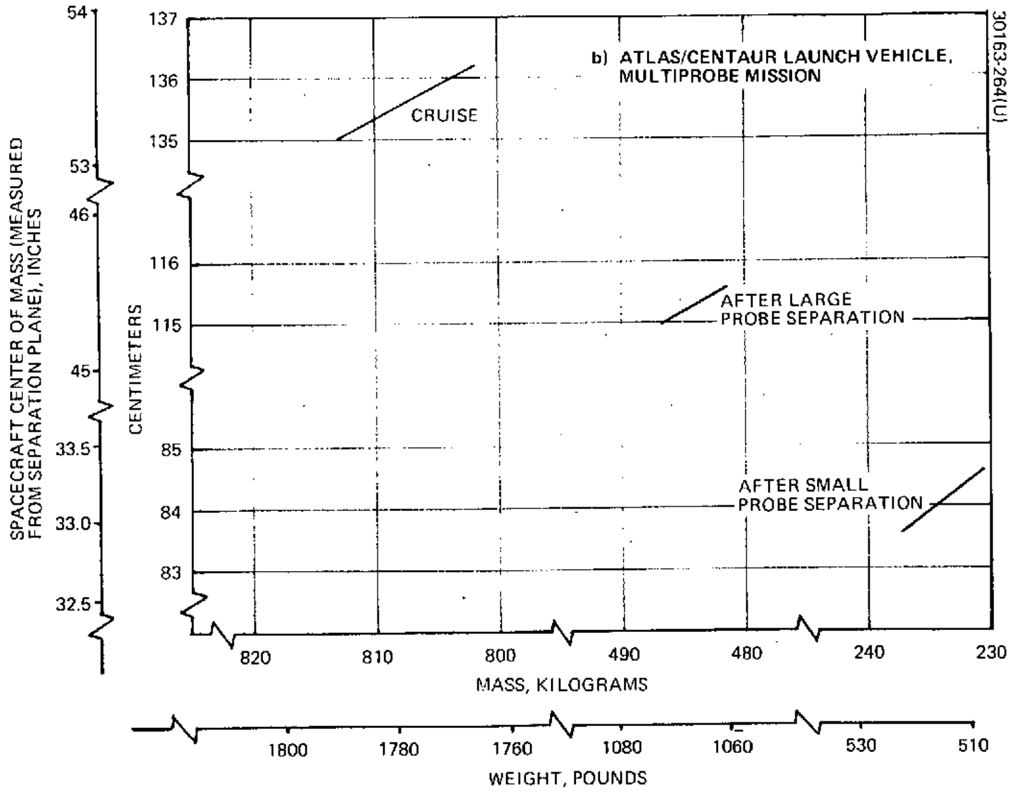


FIGURE 3-8 (continued). SPACECRAFT BUS MASS AND CENTER OF MASS CHANGE VERSUS PROPELLANT UTILIZATION

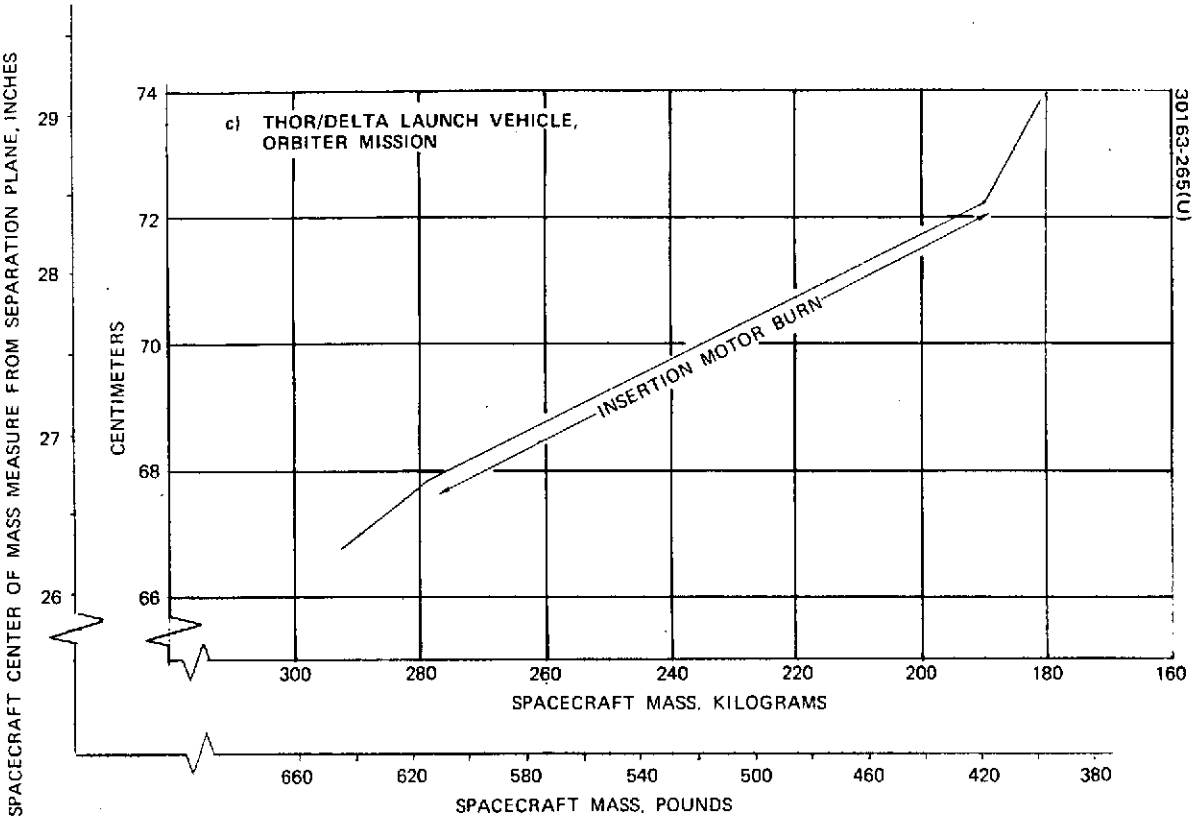


FIGURE 3-8 (continued). SPACECRAFT BUS MASS AND CENTER OF MASS CHANGE VERSUS PROPELLANT UTILIZATION

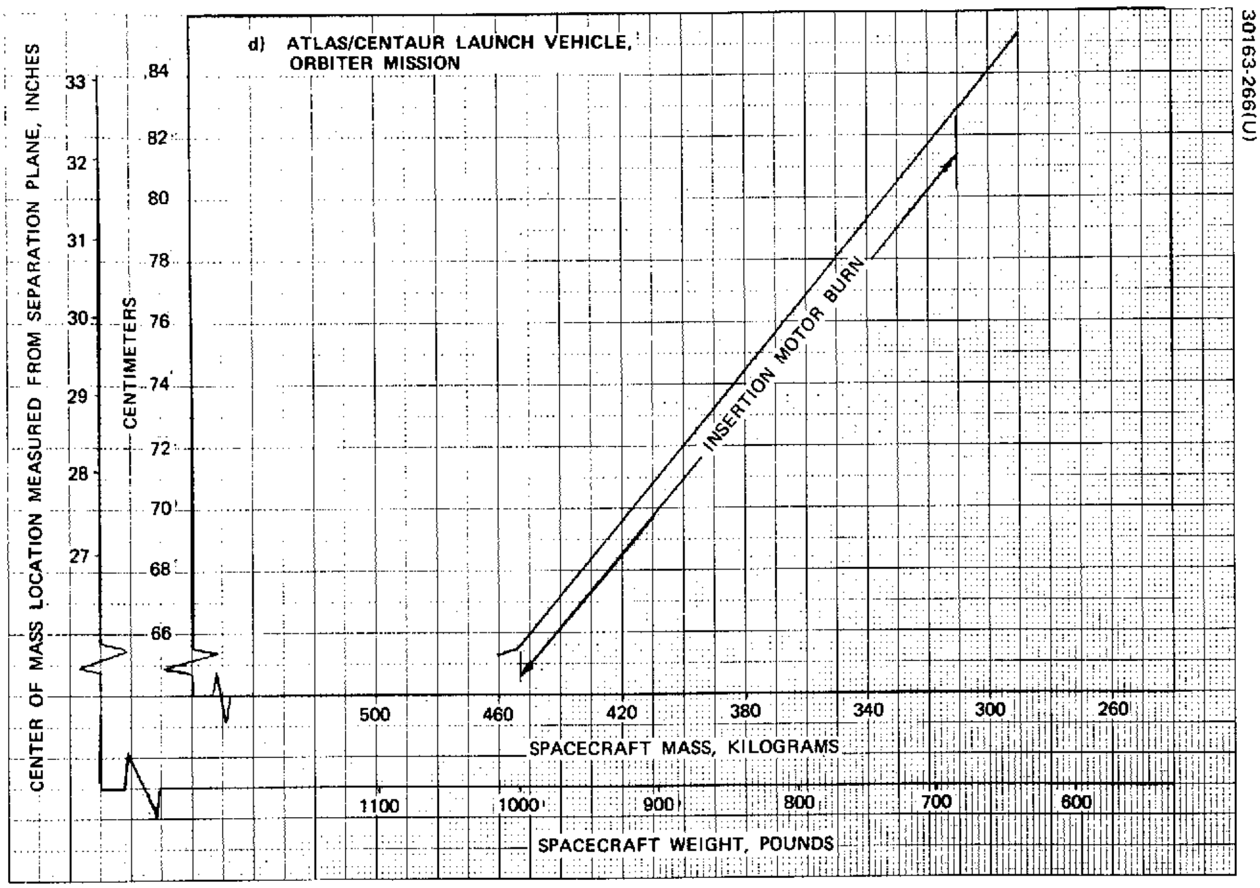


FIGURE 3-8 (continued). SPACECRAFT BUS MASS AND CENTER OF MASS CHANGE VERSUS PROPELLANT UTILIZATION

TABLE 3-16. PROPULSION SUBSYSTEM FUNCTIONAL POWER REQUIREMENTS

1. Pressure transducer (1 unit/spacecraft) 0.25 W at 28 Vdc

2. Thruster propellant valve* (6 units, probe bus; 7 units, orbiter)

   a) Coil Resistance - ohms

Minimum	Maximum	Temperature, °C (°F)
40.2	42	4 (40)
49.6	52	60 (140)
62.6	65.6	138 (280)

   b) Coil Current - amperes

Minimum	Maximum	Temperature, °C (°F)	Voltage, dc
0.524	0.548	4 (40)	22
0.424	0.444	60 (140)	22
0.335	0.352	138 (280)	22

   c) Torque Motor Impedance

   135 ohms at 50 Hz and 21°C (70°F)

   d) Voltage Operating Range, Vdc

Minimum actuate	20.5	
Minimum dropout	1.7	
Maximum applied (variable)	50.0	dropping to 43 in 30 minutes and to 40 in 120 minutes
Maximum applied (sustained)	40.0	

   e) Torque Motor Inductance (at 75 mA, dc 21°C (70°F)

   300 mH minimum
   500 mH maximum

3. Latch valve (2 units/subsystem)**

   Current drain less than 2.5 A at 27 Vdc at 21°C (70°F)

4. Temperature sensors***

   *8 units, probe bus, 9 units, orbiter) 0.09 W each at 28 Vdc

*Signal duration dependent on maneuver requirements

**Signal duration is 50 ms

***Continuous power applied

Power and Thermal Insulation Requirements (Task PP-7)

This study was conducted to define the thermal insulation and heater power required to maintain the propulsion subsystem temperatures within design limits, and to describe the functional power requirements for operating the subsystem valves and telemetry devices.

Electrical power is required to operate the latching valves, propellant valves, pressure transducer, and temperature transducers. The transducers are energized continuously, while the valves are energized through commandable valve drivers. Power requirements for these functions are presented in Table 3-16.

Maximum nonoperating, propulsion subsystem temperatures are influenced by solar heating associated with the spacecraft attitude with respect to the sunline. Minimum temperatures for propellant tanks and thrusters result from eclipses experienced in Venus orbit, and shadowing of the thrusters during spacecraft maneuvers.

The propulsion subsystem thermal design utilizes superinsulation blankets to cover all components and interconnecting tubing in order to thermally couple the subsystem to the thrust tube and equipment shelf; and protect against the solar vacuum environment. Coupling to the equipment shelf satisfactorily maintains the subsystem below the 60°C (140°F) maximum temperature limit.

The predicted steady state temperatures for the propellant tanks are shown in Table 3-17 for the near Earth and near Venus conditions. These represent the minimum and maximum expected steady-state conditions, respectively.

The orbiter apoapsis eclipse represents the most severe transient condition encountered by the tanks. Tank temperature versus time is shown in Figure 3-9 as a function of the propellant load in each tank, for the Thor/Delta mission, Type II trajectory, north periapsis. These eclipses are expected late in the orbiter mission with approximately 0.68 kg (1.5 lb) of propellant in each tank. A small temperature margin is available for this condition.

Each thruster will be conductively insulated from the spacecraft structure by attachment to a fiberglass/epoxy plate to minimize temperature soak back to the valve after firing. A radiation heat shield around each thruster is also used to force rejection to space, of the thruster heat of operation.

Thermal control of the thrusters for the Pioneer Venus mission requires consideration of heater requirements due to the long eclipses encountered during apoapsis and the shadowing resulting from maneuver attitudes not experienced in earth orbital missions. The following analyses were made utilizing the Intelsat IV thruster design. The steady state thruster

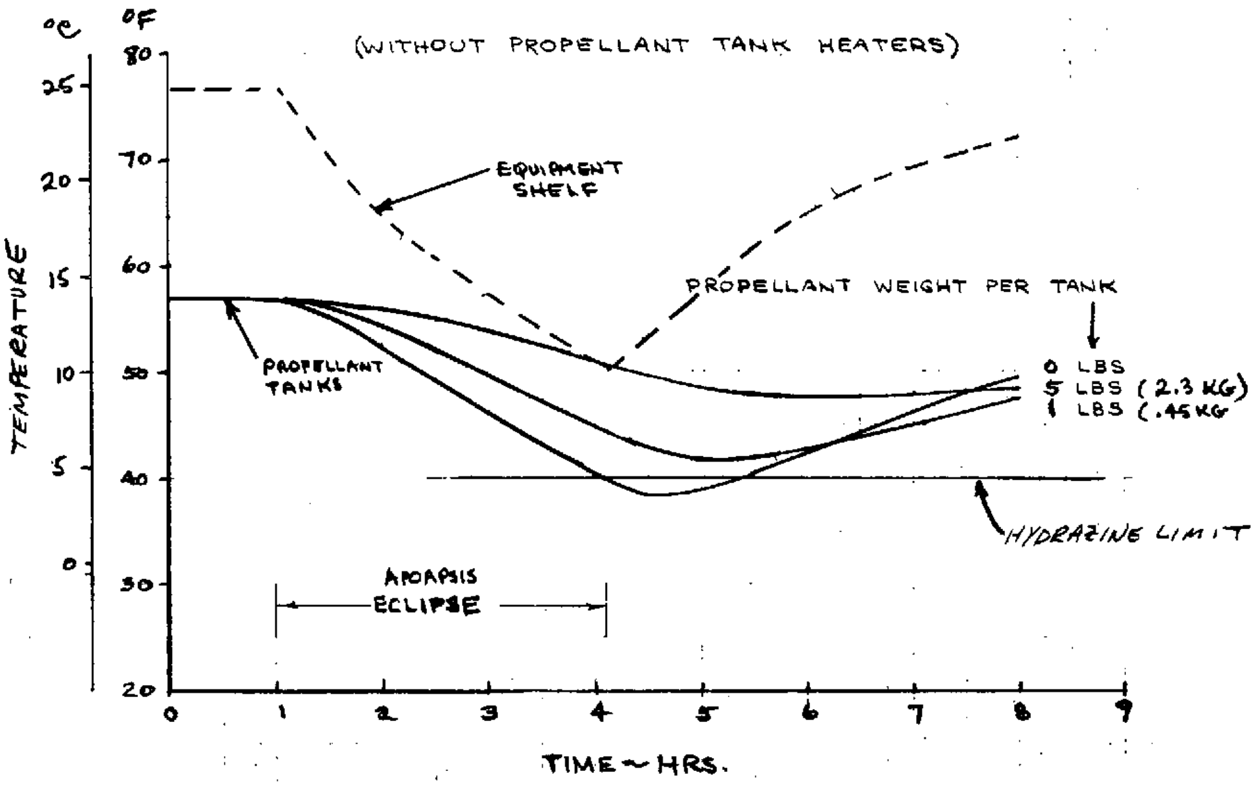


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FIGURE 3-9. ORBITER TEMPERATURE VERSUS TIME DURING ECLIPSE - WITHOUT PROPELLANT TANK HEATERS

TABLE 3-17. PROPELLANT TANK STEADY STATE TEMPERATURES

Design Condition	Temperature °C (°F)
Orbiter	
Near earth	9 (48)
Orbit insertion	15 (59)
Probe Bus	
Near earth	7 (44)
Venus approach	12 (54)

temperatures for the radial thruster positions are shown in Figure 3-10 as a function of valve heater size. Transient temperatures experienced during apoapsis eclipse are shown in Figure 3-11.

The axial thruster thermal design posed a more difficult case due to varied coupling to the equipment shelf as a function of mounting location. A worst case situation was assumed consisting of steady shadowing. The steady state axial thruster temperatures, are shown in Figure 3-12 as a function of heat input to the catalyst bed.

The thruster design to be used for the Pioneer Venus missions is identical to the design qualified for use on the Intelsat IVA program, and utilizes reduced thermal isolation between the catalyst bed and valve. Applying the above thermal analyses to this design, the thruster assembly heater power requirement has been defined as a valve heater with 0.5 W required for thrusters mounted at radial positions, and 1.25 W required for thrusters mounted at axial positions.

Latch Valve Arrangement (Task PP-8)

This study was performed to determine a suitable latch valve arrangement for the propulsion subsystem with the aim of achieving high reliability using as few valves as possible.

A two latch valve configuration was selected for the Pioneer Venus application.

Subsystem reliability was found to be high with no latch valves. Due to the method chosen for initial spinup of the spacecraft after separation from the Atlas/Centaur booster, a minimum of two latch valves is required to assure adequate propellant availability.

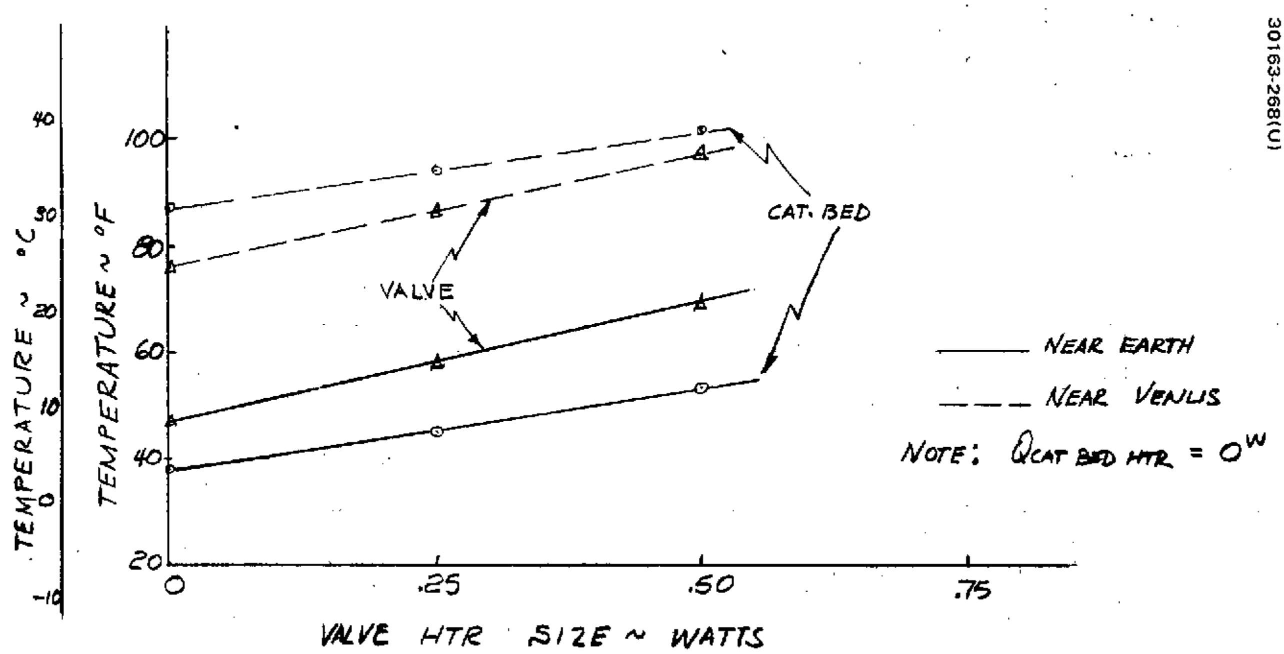
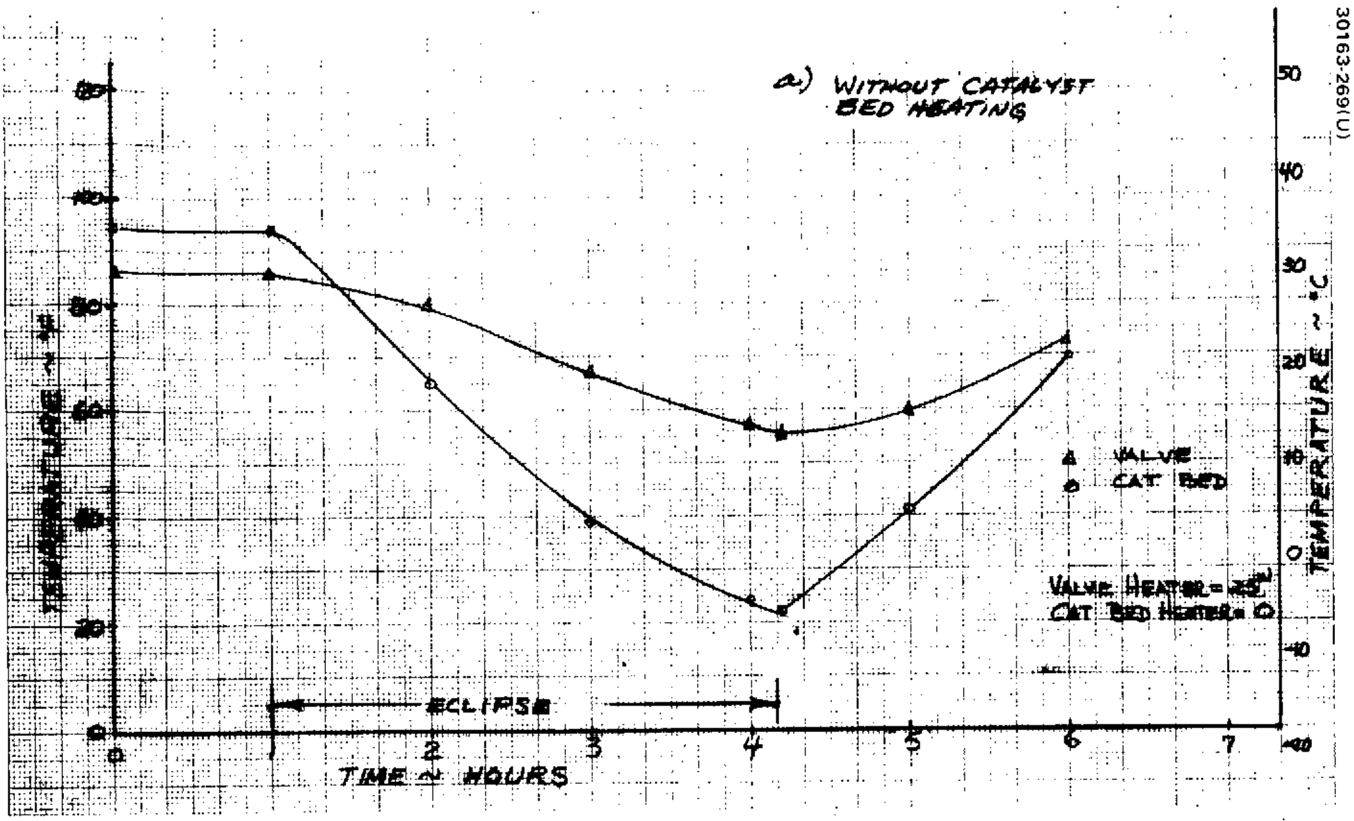


FIGURE 3-10. STEADY STATE RADIAL THRUSTER TEMPERATURES



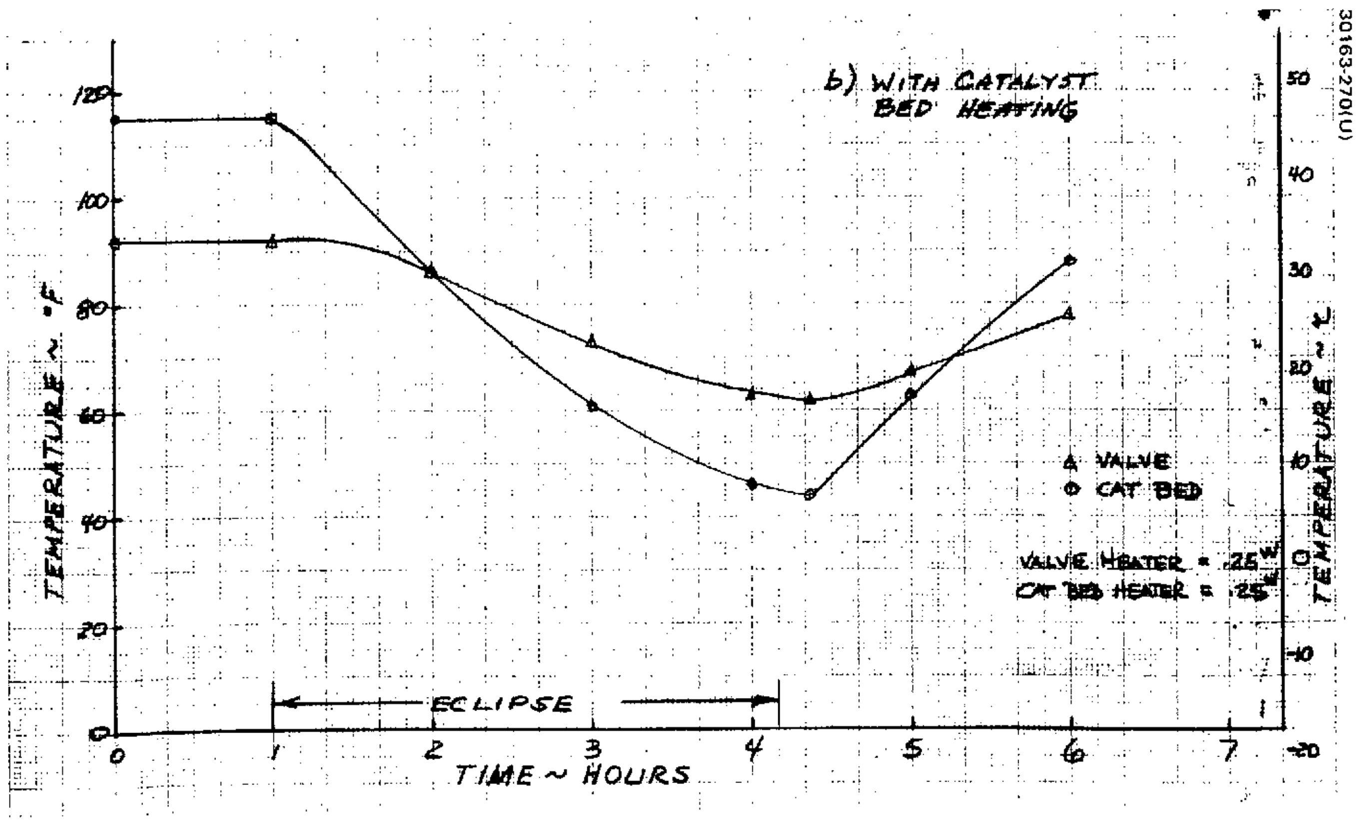


FIGURE 3-11. RADIAL THRUSTER TEMPERATURE VERSUS TIME DURING ORBITER APOAPSIS ECLIPSE

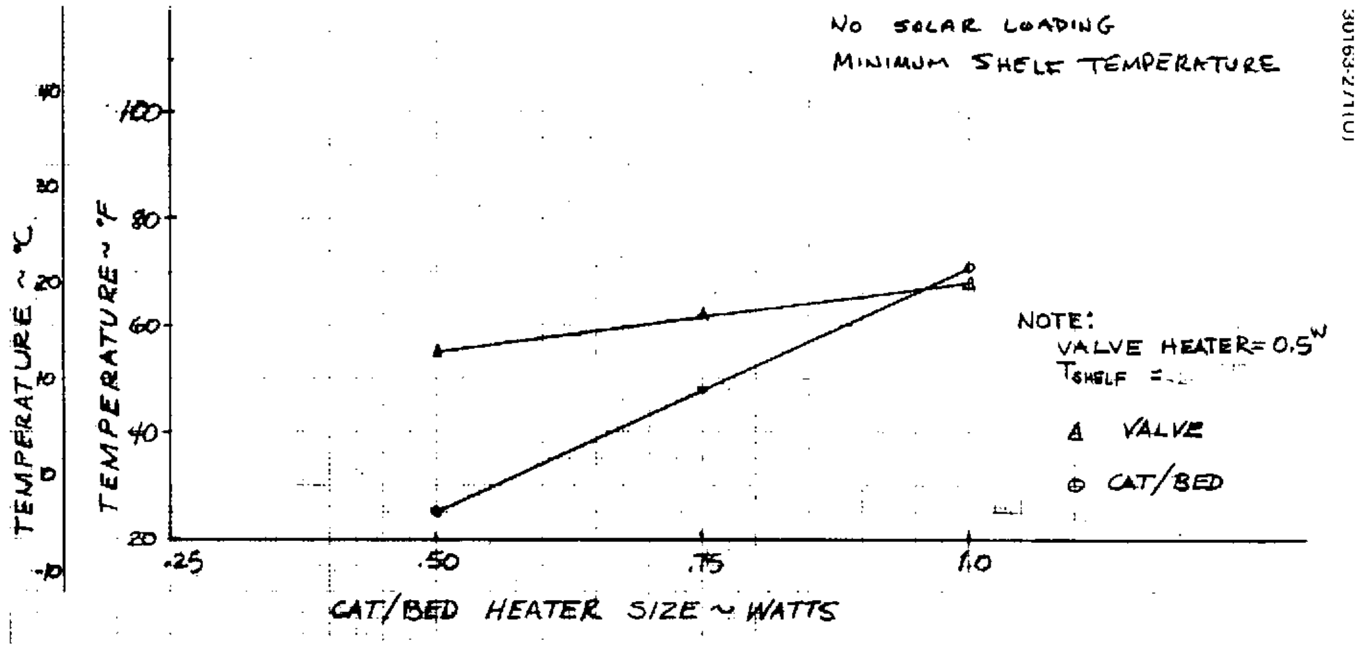


FIGURE 3-12. AXIAL THRUSTER STEADY STATE TEMPERATURES

An additional argument for incorporating latch valves is the long communications transit time to the spacecraft when it is near Venus. This increases reaction time to a situation involving failure of a thruster valve to shut down. Assurance of positive shutdown may be obtained by the addition of latch valves, in series with the thruster valves, which are cycled closed at the end of each maneuver sequence.

Figure 3-13 presents the propulsion subsystem schematics considered in this study.

Figure 3-14 shows component (latch valve) mass, cost, and reliability as a function of the number of valves used. The two valve configuration precludes the single point failure possible with the single valve design. Additional valves increase mass and cost while providing only slight reliability increase; therefore, the two valve configuration was selected as the baseline design. The operational concept of closing the latch valves at the end of each maneuver sequence was also adopted.

Thruster Operating Requirements (Task PP-9)

This study was performed to establish thruster operating requirements consistent with the mission maneuver sequence and with the thruster arrangement selected. Specifically, the requirements for number of starts, continuous mode duration, number of pulses and pulsing duty cycle were defined for axial and radial thruster positions under nominal, off nominal, and failure mode conditions, for Atlas/Centaur and Thor/Delta missions.

The continuous and pulsing requirements for the nominal and off nominal missions are summarized in Tables 3-18 and 3-19.

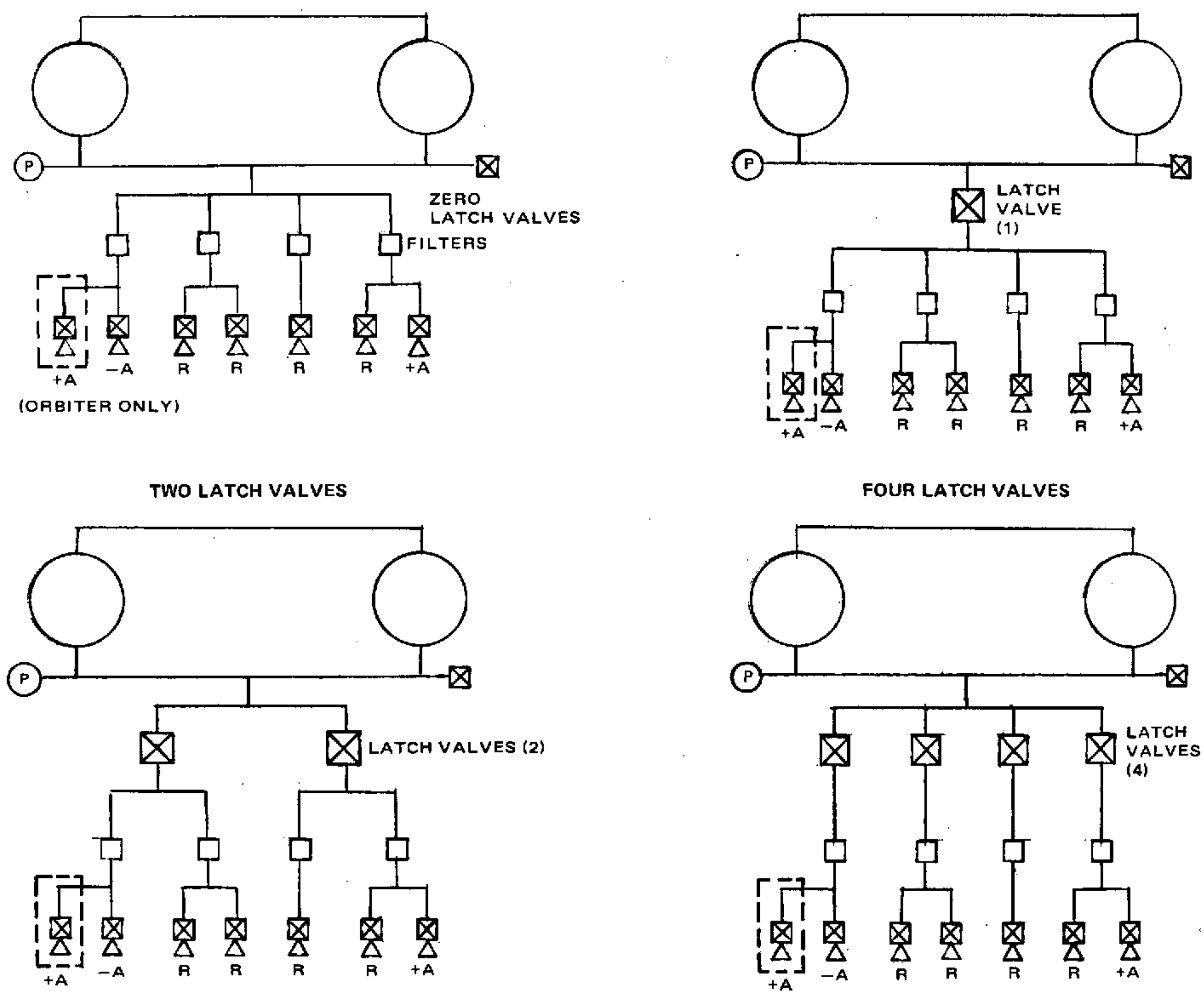
Maneuver requirements fall into three general categories:

1) Velocity increments
2) Spin speed control
3) Attitude control

Continuous mode thruster operation is planned for large velocity increment maneuvers as well as spin speed control maneuvers. Pulsing operation is planned for small velocity increment maneuvers and all attitude control maneuvers. The nominal mission refers to the case where the spacecraft is precessed to the required attitude and an axial thruster is fired in a continuous mode for the first trajectory correction maneuver.

The off nominal maneuvers refer to the case where the velocity increment vector for the first trajectory correction falls within ±15 deg of the sun line. In this case, it is planned that a pair of radial thrusters, pulsed in phase with the spin frequency, provide the necessary impulse.

Failure conditions (i.e., an inoperative thruster) have been considered whereby backup operating modes have been defined to perform all required maneuvers.



R: RADIAL THRUSTER
−A: AXIAL THRUSTER AT FORWARD END OF SPACECRAFT
+A: AXIAL THRUSTER AT AFT END OF SPACECRAFT

FIGURE 3-13. PROPULSION SUBSYSTEM CANDIDATE CONFIGURATIONS

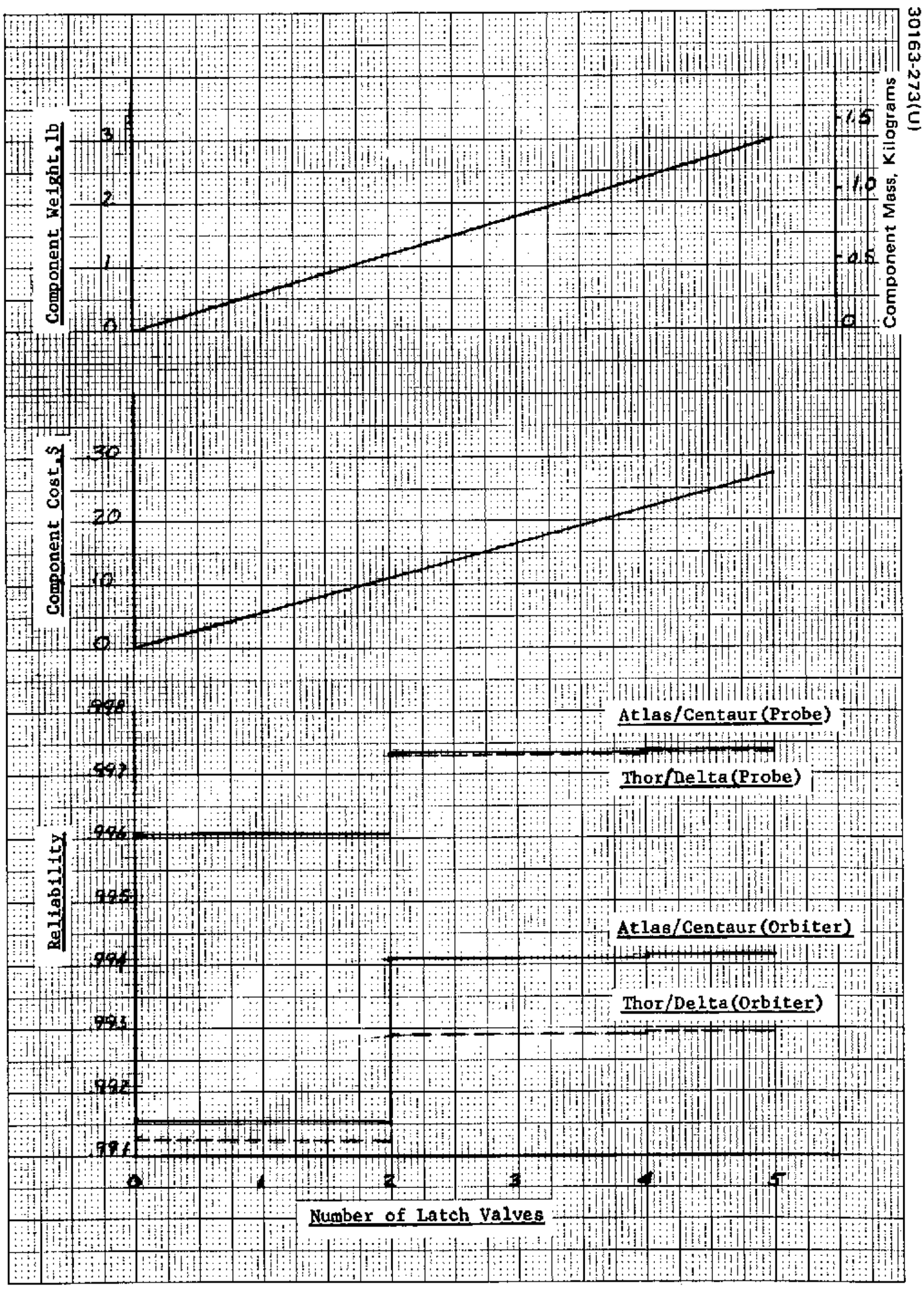


FIGURE 3-14. COMPONENT COST, MASS, AND RELIABILITY VERSUS NUMBER OF LATCH VALVES

TABLE 3-18. AXIAL THRUSTER OPERATING SUMMARY

	Continuous				Pulsing			
	Thor/Delta		Atlas/Centaur		Thor/Delta		Atlas/Centaur	
	Probe Bus	Orbiter	Probe Bus	Orbiter	Probe Bus	Orbiter	Probe Bus	Orbiter
Maximum number of starts[1]	4	35	4	36	10	38	8	35
Maximum duration (single start)	1015.3[2] sec	765.1[2] sec	300.4[2] sec	288.5[2]	454 pulses	326 pulses	294 pulses	237 pulses
Maximum duration (mission)	1430.6 sec	1234.1 sec	565.6 sec	1084.8 sec	2307 pulses	2171 pulses	1119 pulses	1112 pulses
Maximum propellant[1] kg (lb)	14.6 (32.10)	12.7 (27.98)	6.8 (15.09)	12.1 (26.68)	2.3 (5.07)	1.7 (3.64)	1.7 (3.70)	1.6 (3.48)

[1] All orbiter TCMs performed by aft-mounted axial thruster pair. All probe bus TCMs and bus targeting performed by the same axial thruster.

[2] First TCM performed by a single axial thruster (probe bus and orbiter).

TABLE 3-19. RADIAL THRUSTER OPERATING SUMMARY

	Continuous				Pulsing			
	Thor/Delta		Atlas/Centaur		Thor/Delta		Atlas/Centaur	
	Probe Bus	Orbiter	Probe Bus	Orbiter	Probe Bus	Orbiter	Probe Bus	Orbiter
Maximum number of starts	5	4	4	3	3 4[1]	2 3[1]	4 5[1]	2 3[1]
Maximum duration (single start)	107.5 sec	37.3 sec	137.7 sec	46.6 sec	46 pulses 5744[1] pulses	22 pulses 4317[1] pulses	557 pulses 1440[1] pulses	49 pulses 1389[1] pulses
Maximum duration (mission)	262.1 sec	115.0 sec	319.8 sec	148.3 sec	98 pulses 5790[1] pulses	29 pulses 4346[1] pulses	682 pulses 2122[1] pulses	57 pulses 1446[1] pulses
Total propellant kg (lb)	1.1 (2.41)	1.0 (2.26)	3.9 (8.49)	0.6 (1.24)	0.08 (0.18) 6.7 (14.69[1])	0.03 (0.06) 5.0 (11.09[1])	0.9 (2.05) 3.0 (6.57[1])	0.09 (0.19) 2.1 (4.54[1])

[1] Assumes first TCM performed by a pair of radial thrusters in pulse mode

A wide range of spacecraft spin rates will be experienced during the Pioneer Venus missions. A common "on" time (defined as the time from valve-open command to the valve-closed command) will be used for all orbiter and probe spacecraft maneuvers except those performed while spinning at 5 rpm in Venus orbit. The "off" time (time between pulses) will vary, according to the spin rate at the time of each thruster use. This technique was used successfully on all Intelsat IV and Telesat missions. The duty cycle will then vary from 0.117 sec ON/0.728 sec OFF at the maximum spin rate of 71 rpm, to 0.117 sec ON/3.883 sec OFF at 15 rpm. For the on-orbit mission phase, the spacecraft will rotate at 5 rpm. A thruster ON time of 0.030 sec has been selected to provide maneuver granularity. The duty cycle at 5 rpm will be 0.030 sec ON/11.970 sec OFF.

The thruster has demonstrated its ability to operate over a wide range of duty cycles and no difficulty is anticipated with those described above. The 0.117 sec ON time was used on the Intelsat IV program and was selected on Pioneer Venus to utilize existing performance data and minimize the testing required to generate performance data unique to this program.

Summarizing the bounds of thruster operating requirements:

• Longest single pulse train:	5744 pulses
• Longest continuous firing duration:	1260 sec*
• Maximum mission-total number of pulses:	43,564 pulses*
• Maximum mission-total propellant (pulse mode):	15.24 kg* (33.5 lb)
• Maximum mission-total propellant (continuous mode):	14.59 kg (32.1 lb)

* Required for failure mode operation

Thruster performance predictability has been defined for two cases:

I: Predicting performance from nominal curves

II: Predicting performance from acceptance test data on each unit.

Predictability values are as follows:

	Case I	Case II
Continuous mode thrust:	±4%	±2%
Pulse mode cumulative impulse vector magnitude:	±5%	±3%
Pulse mode cumulative impulse vector angle:*	±6 msec	±4 msec

* Pulse trains greater than 25 pulses, and ON time = 0.117 sec

Finally, the temperature range for thruster operation has been defined, based on existing qualification status of the Hughes thruster, as follows:

•	Propellant:	4.44 to 60°C (40 to 140°F)
•	All hardware except for catalyst bed and nozzle:	4.44 to 60°C (40 to 140°F)
•	Catalyst bed and nozzle:	-6.67°C (20°F) minimum

A summary of Hughes 22.24 N (5 lbf) thruster qualification status is presented in Table 3-20.

3.3 THOR/DELTA BASELINE DESCRIPTION

The baseline propulsion subsystem for the probe spacecraft consists of a monopropellant liquid hydrazine, pressure blowdown design as shown in Figure 3-15. Propellant and pressurant are stored together in common tanks to provide a low mass, reliable design using flight proven design concepts. The Telesat tank design was selected to accommodate the propellant loads required for both the multiprobe and orbiter missions providing design commonality between the two spacecraft configurations. The tank volume is 18,040 cm^3 (1100 $in.^3$) and results in a pressure blowdown ratio of 2.33 when loaded with a propellant mass of 20.73 kg (45.7 lb), based on the maneuver schedule of Table 3-1.

Interconnecting manifolds equalize the pressure between tanks and distribute the propellant through appropriate filters to six thruster assemblies arranged in two groups; one axial and a pair of radial thrusters in each group. A bistable latching valve in the manifold feeding each thruster group permits isolation of that group in the event of propellant control valve leakage. A pressure transducer in the liquid manifold provides a telemetered reading of

TABLE 3-20. HUGHES 22.24 N (5 lb_f) THRUSTER QUALIFICATION STATUS

	Intelsat IV						Basic Bus Program[1]	
	S/N HT001		S/N HT002		S/N HT003			
	Continuous	Pulsing	Continuous	Pulsing	Continuous	Pulsing	Continuous	Pulsing
Maximum number of starts	119	172	16	270	115	184	2	259
Maximum duration (single start)	4,020 sec	1,500 pulses	5,400 sec	5,000 pulses	2,700 sec	1,500 pulses	135 sec	14,850[2] pulses
Maximum duration (total qualification)	27,372 sec	34,280 pulses	20,769 sec	31,420 pulses	55,736 sec	31,363 pulses	190 sec	100,000 pulses
Total propellant kg (lb)	235.4 (519)		228.2 (503)		233.1 (514)		2.0 (4.5)	142.3 (313.7)

[1] Thruster modified to accommodate increased pulsing requirements.

[2] Development tests. Qualification requires only 7200 pulses.

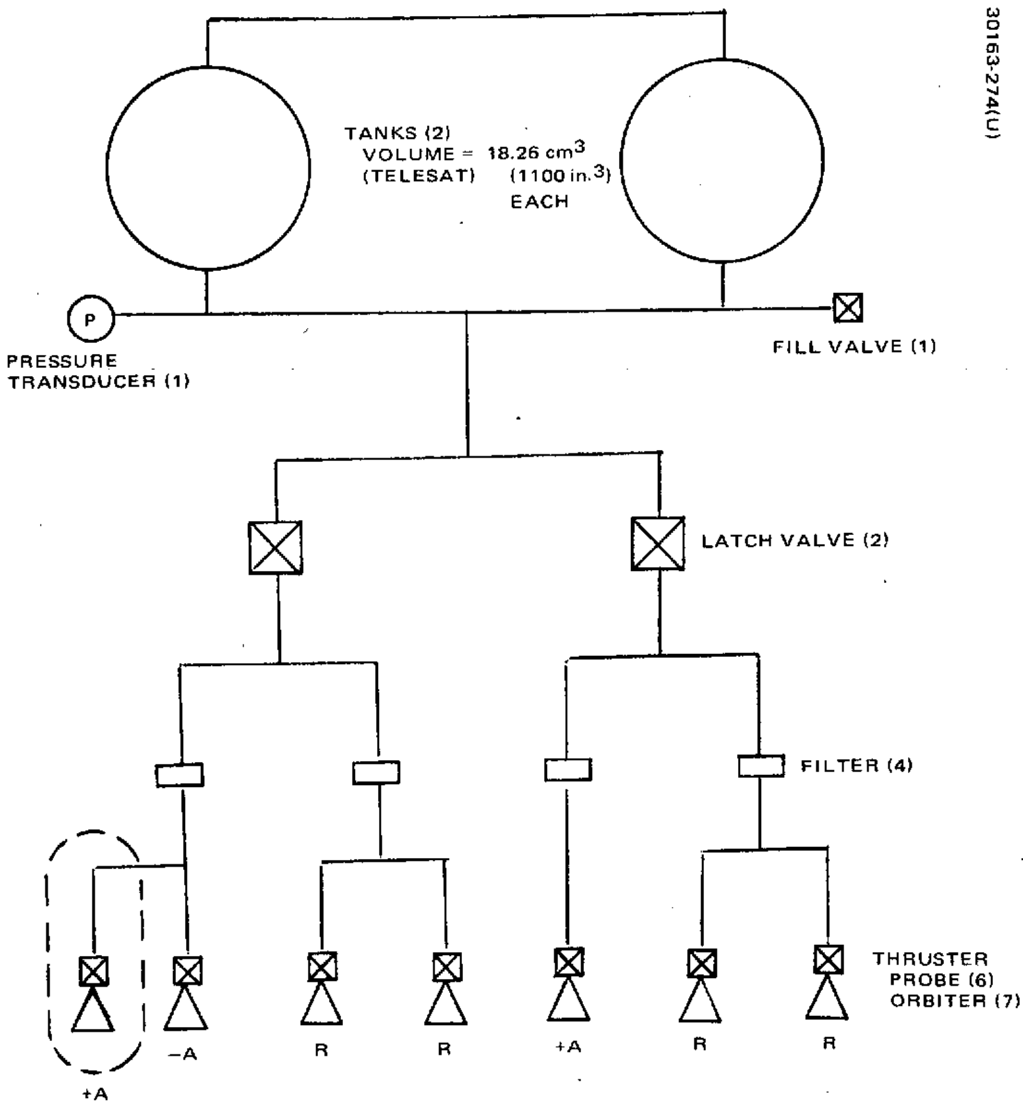


FIGURE 3-15. THOR/DELTA LAUNCH VEHICLE BASELINE PROPULSION SUBSYSTEM SCHEMATIC

TABLE 3-21. THOR/DELTA PROPULSION SUBSYSTEM MASS AND COMPONENT DERIVATION

Component	Probe Spacecraft		Orbiter Spacecraft		Component Derivation
	Number of Units	Subsystem Mass, kg (lb)	Number of Units	Subsystem Mass, kg (lb)	
Propellant tanks	2	3.13 (6.9)	2	3.13 (6.9)	
Thrusters	6	1.63 (3.6)	7	1.91 (4.2)	Intelsat IV
Propellant valves	6	1.36 (3.0)	7	1.59 (3.5)	Intelsat IV, Telesat
Latching valves	2	0.54 (1.2)	2	0.54 (1.2)	Intelsat IV
Pressure transducer	1	0.23 (0.5)	1	0.23 (0.5)	Intelsat IV, Telesat
Fill valve	1	0.14 (0.3)	1	0.14 (0.3)	Intelsat IV, Telesat, Surveyor
Filters	4	0.54 (1.2)	4	0.54 (1.2)	Intelsat IV, Telesat
Tubing	AR	0.32 (0.7)	AR	0.45 (1.0)	Intelsat IV, Telesat
Fittings	10	0.45 (1.0)	10	0.45 (1.0)	Intelsat IV, Telesat
Propellant valve heater	6	0.05 (0.1)	7	0.05 (0.1)	Intelsat IV, Telesat
Temperature sensors	8	0.09 (0.2)	9	0.09 (0.2)	Intelsat IV, Telesat
Thruster insulation	6	0.46 (1.0)	7	0.55 (1.2)	Intelsat IV
Total subsystem		8.94 (19.7)		9.67 (21.3)	

AR = As Required

subsystem internal pressure. Propellant flow is controlled by electrical signal to the torque motor-operated, dual-seat valve supplied with each thruster. The propellant decomposes in the thruster and the hot gases exhaust through the nozzle to produce the required thrust. The liquid gas interface in each tank is determined by the 1 g local gravitational force on the ground and by the centrifugal forces associated with the spacecraft rotation during the mission.

Propellant and pressurant may be loaded through a single fill and drain valve, a concept proved successful on the Intelsat IV, Telesat, and basic bus programs.

The propulsion subsystem is basically an all-welded system fabricated from 6Al - 4V titanium (a technique developed on Intelsat IV and used successfully on all subsequent Hughes satellites). Weld interfaces with some stainless steel components are accomplished with titanium to stainless steel diffusion bonded, coextruded transition joints. Thus, the only mechanical joints in the system are the redundantly sealed fill and drain valve and propellant control valves for each thruster.

All components have been flight proven on the Intelsat IV or Telesat satellite or qualified for flight on Intelsat IV. A detailed summary of component derivation is presented in Table 3-21, along with component masses which form a basis for the propulsion subsystem dry mass of 8.94 kg (19.7 lb). Photographs of existing components are shown in Figures 3-16 through 3-21.

The subsystem requires thermal control to maintain temperatures within the following allowable limits:

Propellant tanks, lines, and all components except for thruster catalyst beds and nozzles:	4.4°C (40°F) to 60°C (140°F)
Catalyst beds and nozzles:	-6.7°C (20°F) minimum

A passive thermal control design was selected to maintain the temperatures of propellant tanks, lines, and components within allowable units by placing these items under insulating blankets. Thruster temperatures will be maintained above minimum by the use of commandable heaters on the thruster propellant valves. Radial thruster valve heaters will require 0.5 W and axial thruster valve heaters 1.25 W.

The orbiter spacecraft propulsion subsystem is identical to the probe spacecraft subsystem described above except for the addition of one axial thruster at the orbit insertion motor end of the spacecraft. This is required to minimize induced nutation as well as provide redundant maneuver capability while in orbit about Venus.

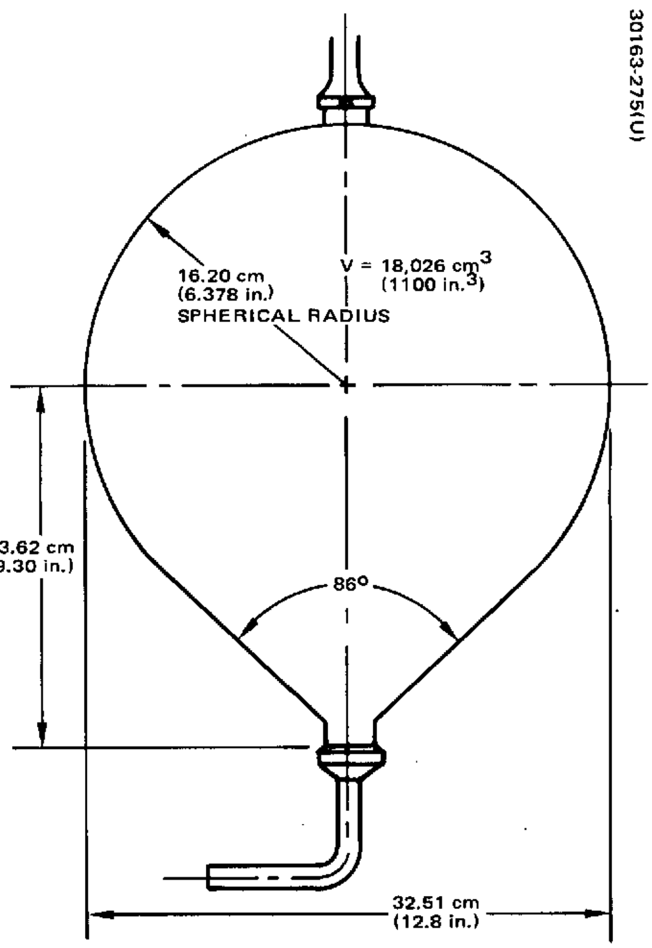
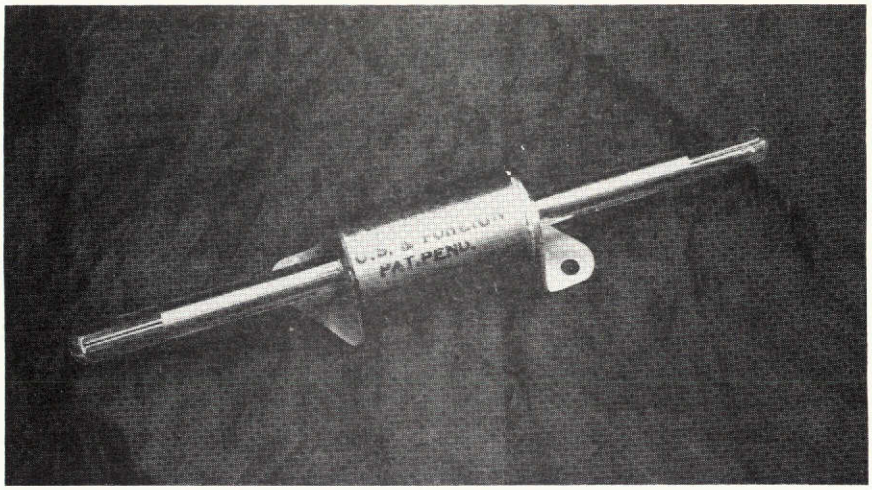
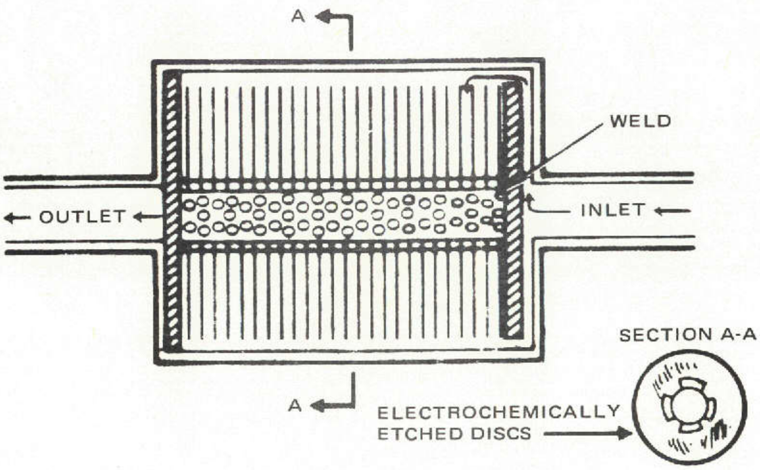


FIGURE 3-16. PROPELLANT TANK CONFIGURATION THOR/DELTA BASELINE



a) ASSEMBLED CONFIGURATION (PHOTO A27890)



b) CROSS SECTION VIEW

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FIGURE 3-17. LIQUID FILTER ASSEMBLY

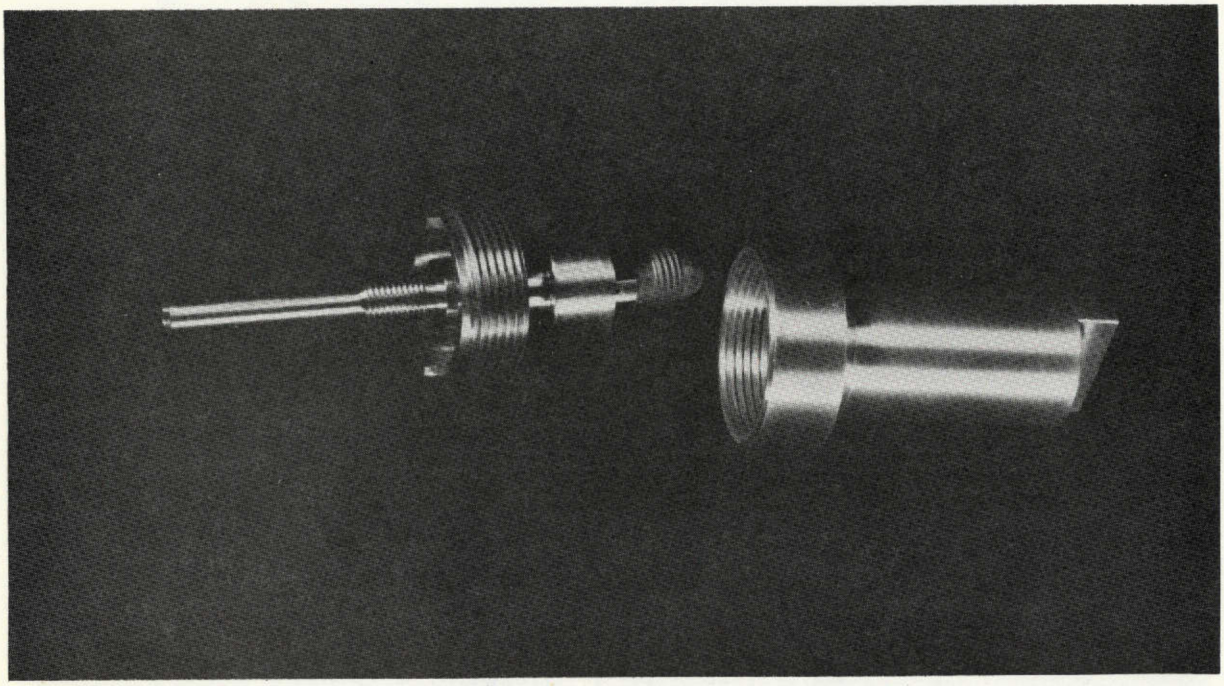
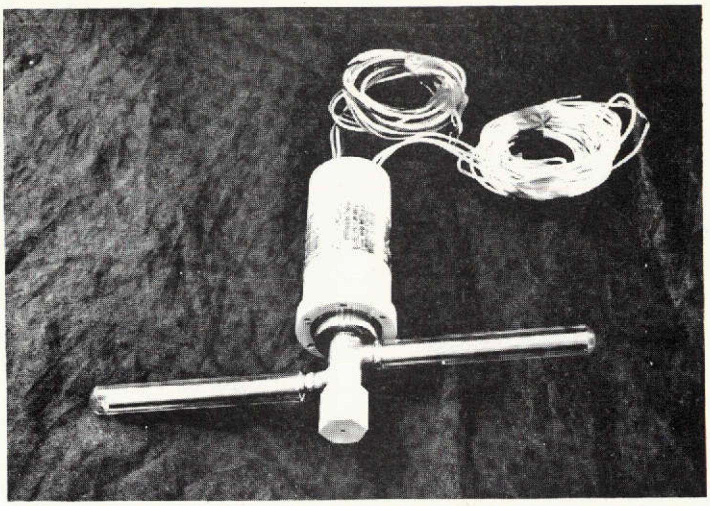
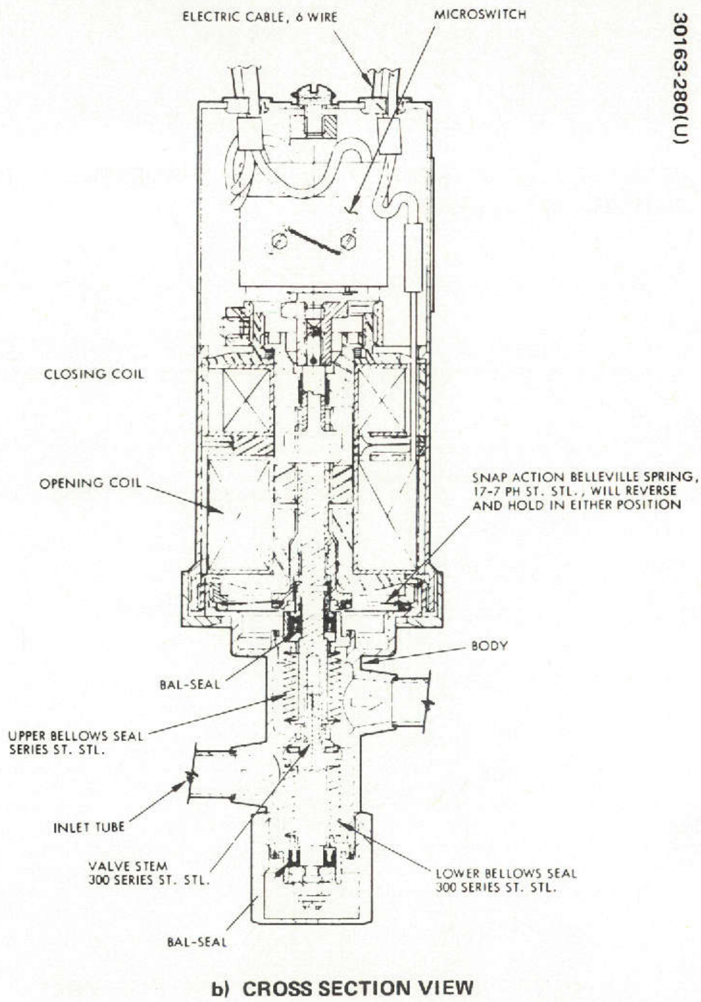


FIGURE 3-18. FILL AND DRAIN VALVE ASSEMBLY (PHOTO A29088)



a) ASSEMBLED CONFIGURATION (PHOTO A27887)



b) CROSS SECTION VIEW

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FIGURE 3-19. PROPELLANT LATCHING VALVE

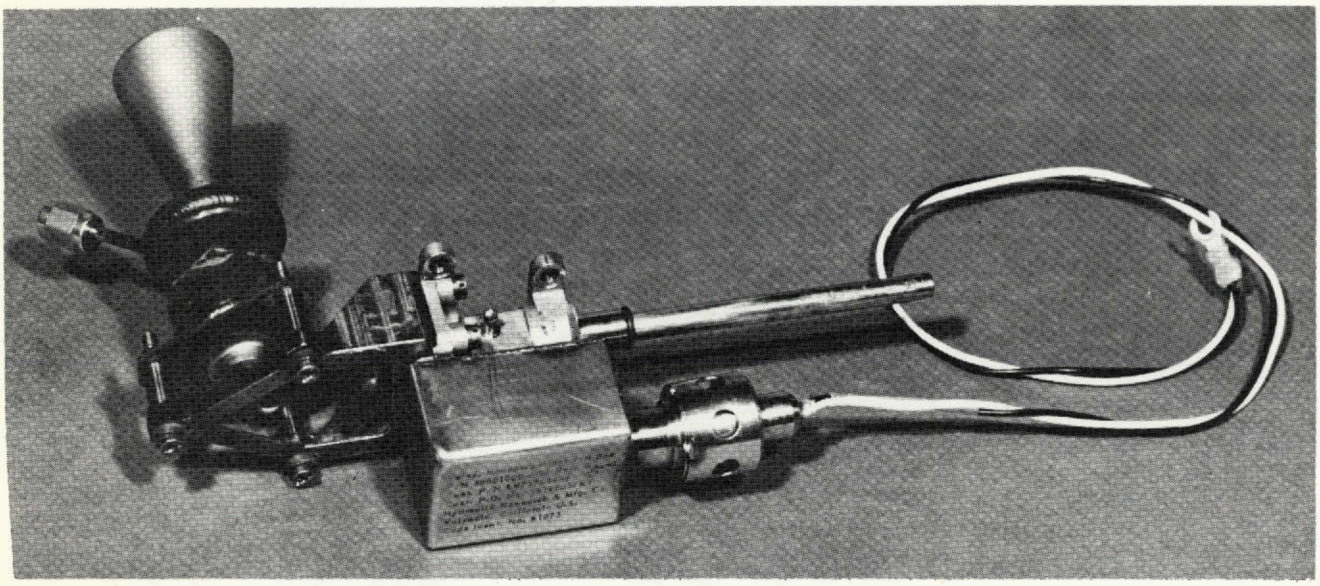

FIGURE 3-20. HUGHES 22.2 N (5lbf) THRUSTER ASSEMBLY (PHOTO ES77132139)

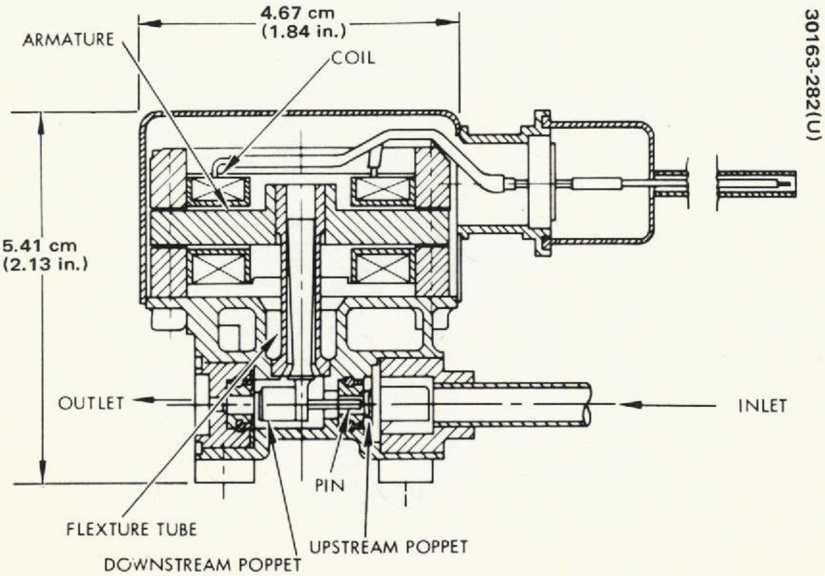

FIGURE 3-21. PROPELLANT CONTROL VALVE CROSS SECTION

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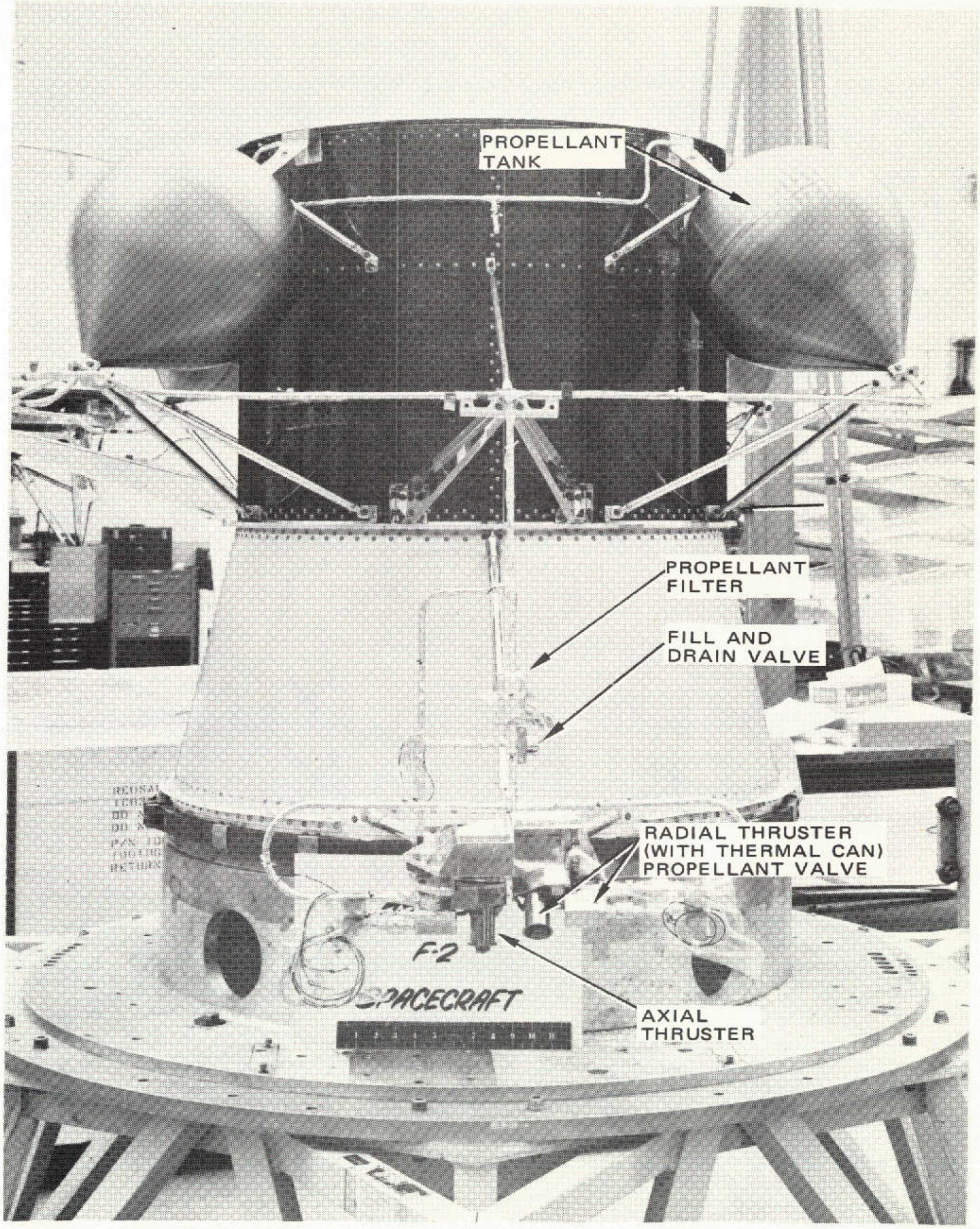

FIGURE 3-22. RCE SUBSYSTEM FOR TELESAT (PHOTO 72-12130)

A propellant mass of 24.31 kg (53.6 lb) was determined, based on the maneuver schedule of Table 3-3. This results in a blowdown ratio of 3.02 when using the Telesat tank as discussed above. Subsystem mass for this design is summarized in Table 3-21 and totals 9.67 kg (21.3 lb). The component derivation is identical to the probe spacecraft propulsion subsystem.

Figure 3-22 shows a subsystem assembly made for the Telesat satellite. This subsystem was assembled on the spacecraft structure, a technique similar to that proposed for the Pioneer Venus program.

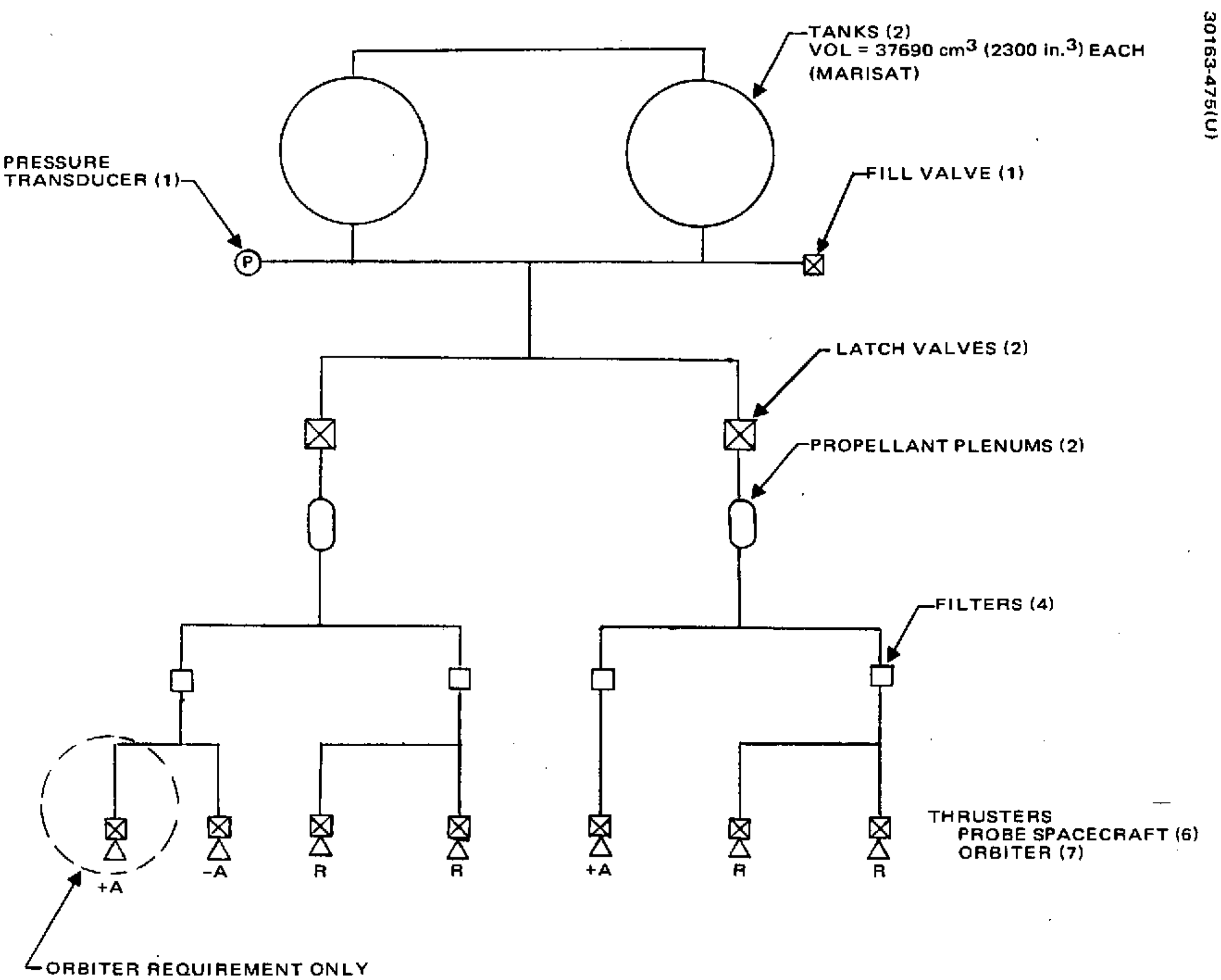

FIGURE 3-23. ATLAS/CENTAUR LAUNCH VEHICLE BASELINE PROPULSION SUBSYSTEM SCHEMATIC

3.4 ATLAS/CENTAUR BASELINE DESCRIPTION

The baseline propulsion subsystem for the orbiter and probe spacecraft consists of a monopropellant liquid hydrazine pressure blowdown design as shown in Figure 3-23. The propellant and pressurant are stored together in common tanks to provide a low mass, reliable design using flight proven design concepts. The tank design selected for these configurations is shown in Figure 3-24. The liquid/gas interface in each tank is determined by the 1 g local gravitational force on the ground and by the centrifugal forces associated with spacecraft rotation during the mission. Tank volume is 37690 cm^3 (2300 in^3), adequate to accommodate the propellant requirements for either the orbiter or the multiprobe missions. This tank was qualified for use on the Hughes Marisat Satellite Program. The tank is constructed of 6AL4V titanium and features a conispherical shape which has been used on Intelsat IV, Telesat, and basic bus satellites. This shape allows propellant expulsion on the ground as well as in the spinning environment encountered during the mission with a minimum of external connections. For the propellant loads required, the pressure blowdown ratio is 1.42 for the multiprobe mission, and 1.54 for the orbiter mission. Both operating regimes are consistent with satisfactory thruster performance over the entire blowdown range.

Interconnecting manifolds equalize the pressure between tanks and distribute the propellant uniformly from both tanks to the thrusters. Thrusters are arranged into two groups; one axial and two radial thrusters in each group on the probe spacecraft; and an additional axial thruster (in one group only) on the orbiter spacecraft.

A bistable latch valve, Figure 3-19, in the manifold feeding each thruster group permits isolation of that group in the event of propellant control valve leakage. A 116 cm^3 (7.1 in^3) plenum, located immediately downstream of each latch valve, provides sufficient propellant without the centrifugal force of spin to attain an initial, interim, spacecraft spin rate for the two-stage spin-up maneuver following spacecraft separation from the launch vehicle. The latch valves are closed during the launch-boost phase to assure availability of the propellant in each plenum. A more detailed description of the maneuver is presented in subsection 3.2 under the discussion of Propellant Management Techniques (Task PP-2). This technique has been used successfully for four Intelsat IV flights, and is expected to present no problems when used for this application.

A potentiometer type pressure transducer in the liquid manifold provides a continuous telemetered reading of subsystem internal pressure. This data is useful in several ways: the pressure level is used as an aid in predicting thruster performance, and the pressure history may be used to estimate propellant consumption.

Propellant flow is controlled by electrical signal to the valve supplied with each thruster, Figures 3-20 and 3-21. Each valve provides series redundant proepllant shutoff by utilizing dual, in-line, tungsten carbide poppets and seats. Valve operation is controlled by a single torque motor actuator in each valve. Contaminating material trapped on either seat does not prevent the

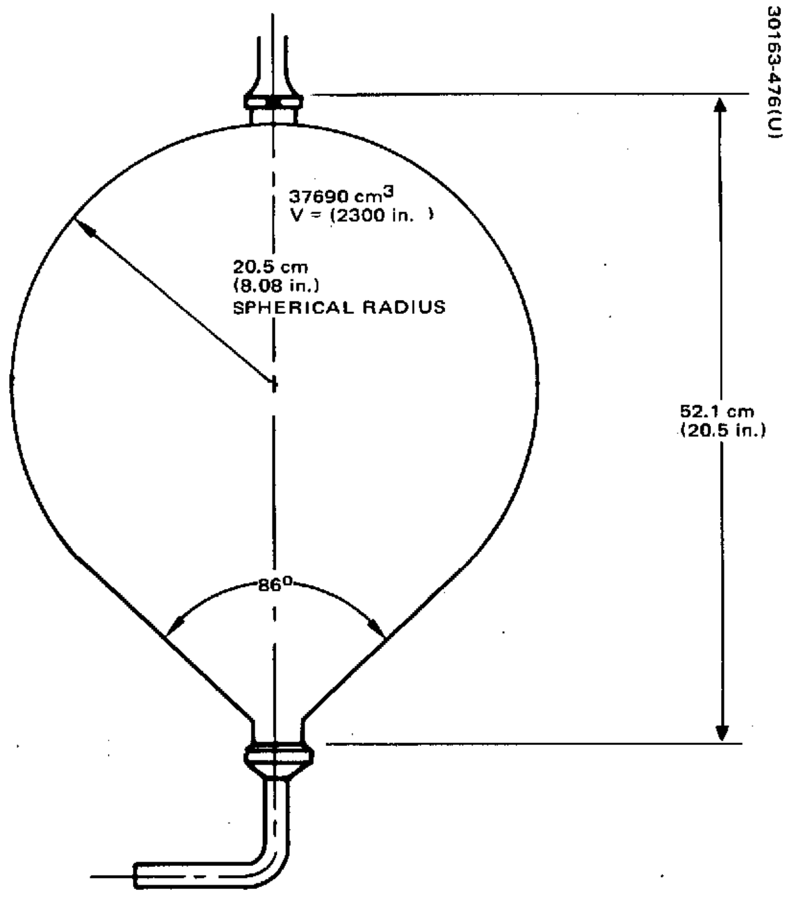

FIGURE 3-24. ATLAS/CENTAUR PROPELLANT TANK CONFIGURATION

TABLE 3-22. ATLAS/CENTAUR PROPULSION SUBSYSTEM MASS AND COMPONENT DERIVATION

Component	Probe Spacecraft		Orbiter Spacecraft		Component Derivation
	Number of Units	Subsystem Mass, kg (lb)	Number of Units	Subsystem Mass, kg (lb)	
Tanks	2	5.35 (11.8)	2	5.35 (11.8)	Marsat
Thrusters	6	1.63 (3.6)	7	1.91 (4.2)	Intelsat IV
Propellant valves	6	1.36 (3.0)	7	1.59 (3.5)	Intelsat IV Telesat Basic Bus
Latching valves	2	0.54 (1.2)	2	0.54 (1.2)	Intelsat IV Telesat Basic Bus
Pressure transducer	1	0.23 (0.5)	1	0.23 (0.5)	Intelsat IV Telesat Basic Bus
Fill valve	1	0.14 (0.3)	1	0.14 (0.3)	Surveyor Intelsat IV Telesat Basic Bus
Filters	4	0.54 (1.2)	4	0.54 (1.2)	Intelsat IV Telesat Basic Bus
Tubing	AR	0.32 (0.7)	AR	0.45 (1.0)	Intelsat IV Telesat Basic Bus
Fittings	10	0.45 (1.0)	10	0.45 (1.0)	Intelsat IV Telesat Basic Bus
Propellant valve heaters	6	0.05 (0.1)	7	0.05 (0.1)	Intelsat IV Telesat Basic Bus
Propellant tank heaters	2	0.27 (0.6)	2	0.27 (0.6)	Basic Bus (modified)
Temperature sensors	6	0.09 (0.2)	9	0.09 (0.2)	Intelsat IV Telesat Basic Bus
Thruster insulation	6	0.45 (1.0)	7	0.54 (1.2)	Intelsat IV Telesat Basic Bus
Propellant plenums	2	0.14 (0.3)	2	0.14 (0.3)	Intelsat IV Basic Bus
Total		11.57 (25.5)		12.29 (27.1)	

AR — As required

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The subsystem requires thermal control to maintain temperatures within allowable limits as follows:

Propellant tanks, lines, and all components except for thruster catalyst beds and nozzles:	4.4°C (40°F) to 60°C (140°F)
Catalyst beds and nozzles:	-6.67°C (20°F) minimum

A passive thermal control design was selected to maintain the temperatures of lines, and components within allowable units by placing these items under insulating blankets. Although propellant tanks are located under these blankets, heaters, requiring 5 W, will be necessary on the conical portion of each tank to maintain the propellant temperature above 4.4°C (40°F) in the near earth environment. Thruster temperatures will be maintained above minimum by the use of commandable heaters on the thruster propellant valves. Radial thruster-valve heaters will require 0.5 W and axial thruster valve heaters 1.25 W.

4. ORBIT INSERTION MOTOR SUBSYSTEM

4.1 SUBSYSTEM REQUIREMENTS

The orbit insertion motor subsystem must provide the necessary impulse to insert the spacecraft into an elliptical orbit about the planet Venus.

The impulse required for this maneuver varies as a function of the launch date, the Earth-Venus cruise trajectory selected, the spacecraft mass, the Venus orbit periapsis altitude and latitude, as well as other influencing factors which were evaluated in detail under Mission Analysis in Volume 3.

The baseline missions considered for this study were:

1978 launch, Type II trajectory, 26° N periapsis (Thor/Delta)

1978 launch, Type II trajectory, 50° S periapsis (Atlas/Centaur)

In both cases, it was assumed that the spacecraft mass at launch was equal to the maximum capability of the launch vehicle under consideration. Propulsion subsystem propellant was consumed to perform midcourse corrections based on the associated 99 percent launch vehicle dispersion errors, thus allowing computation of spacecraft mass at the point of orbit insertion. Accordingly, the insertion maneuver requires a velocity increment of 1069.8 m/sec to a spacecraft mass of 279.7 kg (616.6 lb) for the Thor/Delta launched spacecraft; and 1070 m/sec to a spacecraft mass of 454.4 (1001.8 lb) for the Atlas/Centaur launched spacecraft.

The insertion motor subsystem must withstand the vibration environments of Table 4-1 for each respective launch vehicle.

Insertion motor thrust must not accelerate the spacecraft in excess of 10 g to preclude influencing the spacecraft structural design.

The insertion motor case temperature during firing and the following soak-back period must not exceed 316°C (600°F). Motor temperature requirements during cruise between Earth and Venus will be determined as a function of the motor subsystem design selection.

TABLE 4-1. ORBIT INSERTION MOTOR QUALIFICATION VIBRATION LEVELS (THOR/DELTA)

Sinusoidal (Frequency sweep at 2 oct/min)		
Axis	Frequency, Hz	Acceleration, g, 0-peak
Thrust	5 to 15	2.3
	15 to 21	6.8
	21 to 250	2.3
	250 to 400	4.5
	400 to 2000	7.5
Lateral	5 to 14	2.3
	14 to 250	1.5
	250 to 400	4.5
	400 to 2000	7.5

Random (Test duration = 4 min/axis)			
Axis	Frequency Hz	PSD Level, g^2/Hz	g_{rms}
All axes	20 to 300	+3 dB/oct	9.2
	300 to 2000	0.045	

TABLE 4-1. ORBIT INSERTION MOTOR QUALIFICATION VIBRATION LEVELS (ATLAS/CENTAUR) (Continued)

Sinusoidal (Frequency sweep at 2 oct/min)		
Axis	Frequency, Hz	Acceleration, g, 0-peak
Thrust	5 to 8.5	1.5 cm (0.6 inch) d.a.
	8.5 to 250	2.3
	250 to 400	3.75
	400 to 2000	7.5
Lateral	5 to 8	1.1 cm (0.45 inch) d.a.
	8 to 250	1.5
	250 to 400	3.0
	400 to 2000	7.5

Random (Test duration = 4 min/axis)			
Axis	Frequency Hz	PSD Level g^2/Hz	g_{rms}
All axes	20 to 150	+6 dB/oct	9.3
	150 to 2000	0.045	

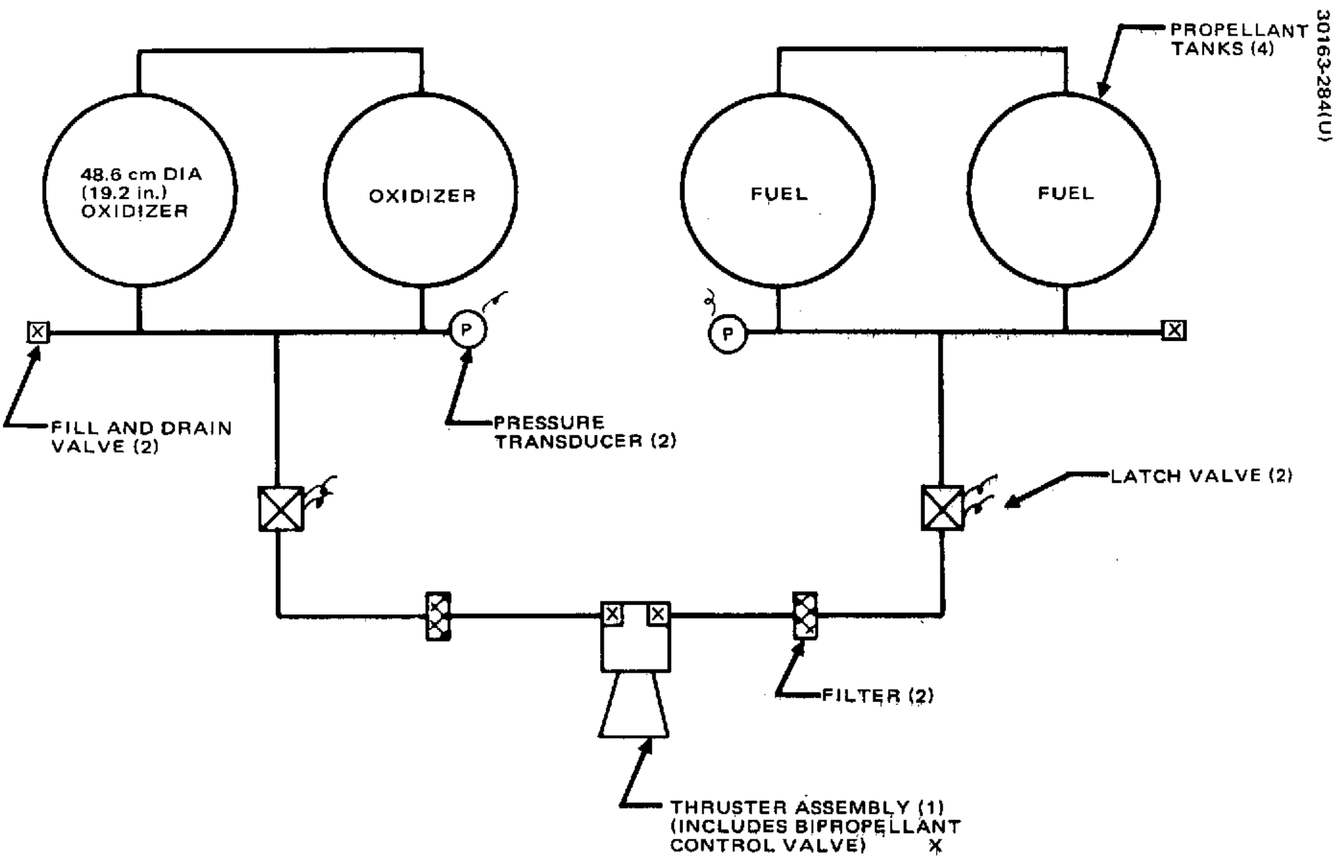

FIGURE 4-1. LIQUID BIPROPELLANT ORBIT INSERTION MOTOR SCHEMATIC

4.2 SUBSYSTEM TRADES

Several trade studies were performed to determine the most suitable type of subsystem to accomplish this mission, to define the details of the selected subsystem type, and the thermal control required. Trade study results are summarized in Table 3-6.

Criteria for subsystem selection included low mass, low program cost, high reliability, high performance, and space storage for 200 days prior to operations.

Propellant Tradeoff (Task PP-3)

This study considered the relative merits of a solid propellant rocket motor versus a liquid bipropellant motor assembly to perform the orbit insertion maneuver. The results of this study supported the selection of the solid propellant motor concept on the basis of lower cost, use of flight-proven motor designs with some modification, and comparable subsystem mass (notwithstanding the slightly lower performance of the solid propellant).

The liquid bipropellant subsystem shown in Figure 4-1, used propellants consisting of Aerozine-50 fuel and nitrogen-tetroxide. A Marquardt Corporation motor was selected which provides a thrust of 444.8 N (100 lb). A higher thrust, Rocketdyne motor, was rejected because of higher hardware mass. Subsystem mass is presented in Table 4-2 with tank size optimized for the propellant load. The cost of the liquid propellant subsystem, including qualification tests, was estimated at approximately $1.9M. The corresponding solid propellant motor subsystem cost was estimated at $0.4M.

Another liquid bipropellant design considered for the Atlas/Centaur mission option was the Symphonie system by Messerschmitt-Bolkow-Blohm GMBH, modified to fit the Pioneer Venus thrust tube envelope, and using a 400 N thrust engine. A summary of system design is shown in Figure 4-2, and performance parameters are presented in Table 4-3. Assuming attainment of the 305 sec I_{sp} quoted, the total subsystem mass required is 157.3 kg (346.8 lb). This agrees with the mass of the stretched TE-M-521 motor described in subsection 4.4.

The solid motor design approach was selected based on the use of existing solid motor design technology, providing shorter maneuver time at a program cost savings estimated of approximately $0.5M.

Power and Thermal Insulation Requirements (Task PP-7)

This study was performed to determine the functional and heater power requirements, as well as the thermal insulation requirements for the orbit insertion motor.

Power is required to fire the squibs in the igniter assembly. A minimum of 4.5 A is required at 24 ± 5 V for 0.010 sec.

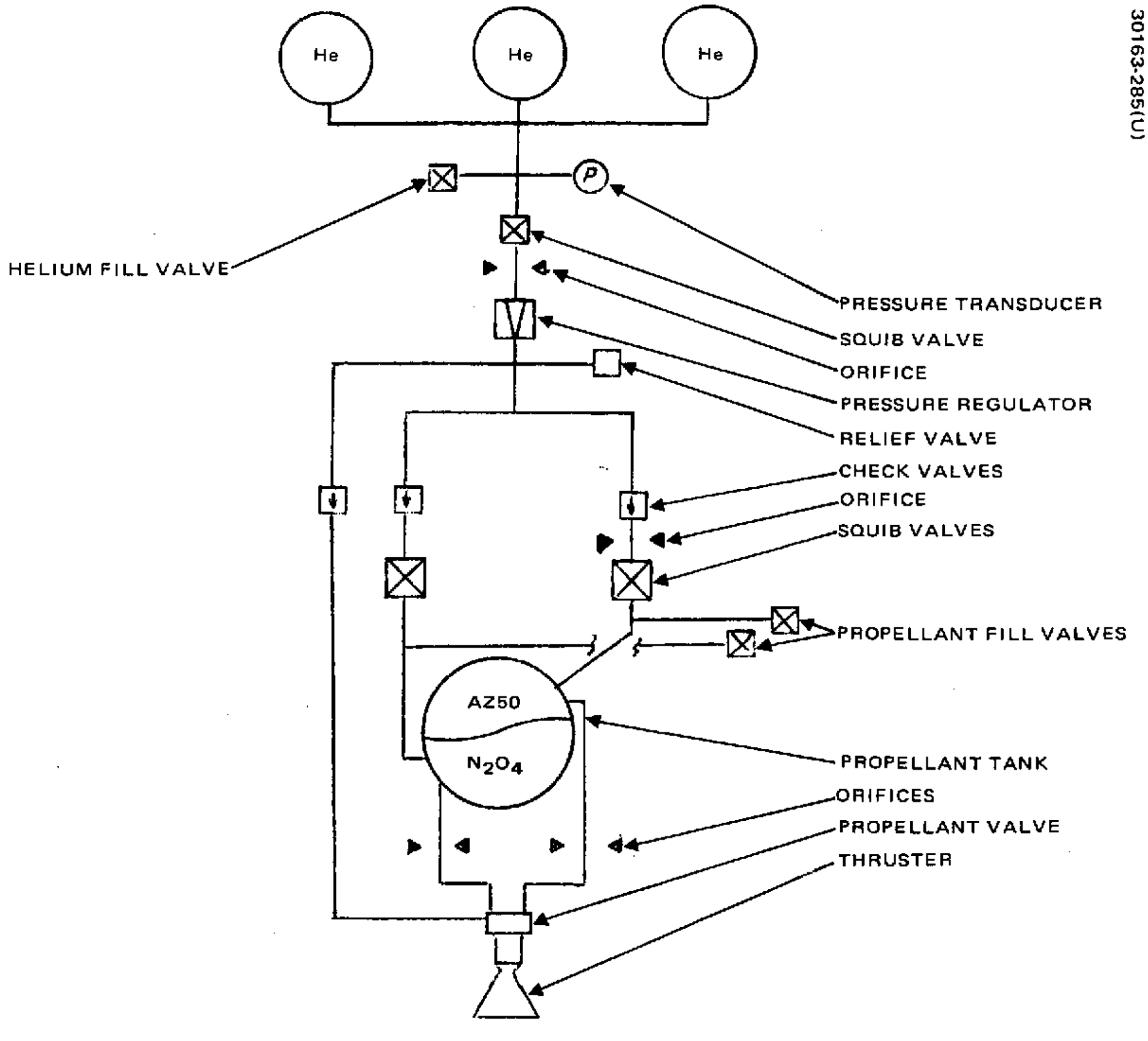

FIGURE 4-2. SYMPHONIE BIPROPELLANT ORBIT INSERTION MOTOR

TABLE 4-2. MASS LIST FOR THE LIQUID BIPROPELLANT ORBIT INSERTION MOTOR

	Quantity	Unit Mass, kg (lb)	Total Mass, kb (lb)	
Engine, 444.8 N (100 lb) thrust	1	2.27 (5.0)	2.27	(5.0)
Tanks	4	2.74 (6.05)	10.98	(24.2)
Filters	2	0.23 (0.5)	0.45	(1.0)
Fill valves	2	0.23 (0.5)	0.45	(1.0)
Latch valves	2	0.27 (0.6)	0.54	(1.2)
Pressure transducers	2	0.18 (0.4)	0.36	(0.8)
Thermal sensors	10	0.01 (0.02)	0.09	(0.2)
Lines and fittings			0.91	(2.0)
Heaters			0.45	(1.0)
Structure			2.27	(5.0)
Gas			0.54	(1.2)
Propellant margin*			3.63	(8.0)
Total dry weight			22.94	(50.6)
Propellant			137.89	(304.0)
Total weight			160.83	(354.6)

* Assume 2.5 percent for extended maneuver duration and mixture ratio shifts.

TABLE 4-3. SYMPHONIE BIPROPELLANT SYSTEM DESIGN AND PERFORMANCE PARAMETERS FOR ORBIT INSERTION
(1978 Launch, Type II Trajectory, 56° S Periapsis)

• Subsystem mass, kg (lb)	
Dry	18.5 (40.8)
Propellant	138.8 (306.0)*
Total subsystem	157.3 (346.8)
• Thrust, N (lbf)	392 (88.1)
• Vacuum I_{sp}, sec	305
• Maneuver time, sec	1059
• Propellants	
Oxidizer	N_2O_4
Fuel	Aerozine 50
• Status	Qualified. Requires modification to relocate helium pressurizing gas tanks to meet envelope dimension requirements

* Maximum propellant capability is 148 kg (326.3 lb)

Thermal control of the orbit insertion motor was influenced strongly by the large variation in solar intensity during the transit between Earth and Venus. Another influencing factor was the requirement for occasional spacecraft attitude changes exposing the motor to increased solar heating.

The basic thermal design consists of radiation isolation from the spacecraft by application of multilayer Kapton superinsulation between the spacecraft thrust tube and the motor. The exposed portion of the motor nozzle is covered with 2 mil aluminized teflon. This approach protects the spacecraft from overheating during and immediately after motor firing, and reduces the heat loss from the spacecraft after motor firing.

To maintain minimum temperature, $4^{\circ}C$ ($40^{\circ}F$) near Earth, heater power is required. A 5 W heater was therefore placed at the motor nozzle throat. This approach was used successfully on the apogee motors of the Intelsat IV and Telesat spacecraft. It is intended that this heater be switched off during Earth/Venus transit when it is no longer needed.

The expected motor temperature, at the time of the orbit insertion maneuver, is $25^{\circ}C$ ($77^{\circ}F$).

<u>Orbit Insertion Motor Selection (Task PP-10)</u>

Based on the selection of the solid propellant motor for the orbit insertion maneuver, this study was performed to select a baseline motor from the currently available solid propellant motor designs, and to define any required modifications thereto.

This study considered the:

1978 launch, Type II trajectory, north periapsis (Thor/Delta)

1978 launch, Type I trajectory (Atlas/Centaur) missions

The data in Table 4-4 presents the capability of the available motors. The curves of Figure 4-3 show the available velocity increment as a function of spacecraft initial mass for each motor. Points I and II on this figure represent the design points required for the Thor/Delta and Atlas/Centaur missions considered.

Selecting motors with the highest propellant I_{sp} and lowest burnout mass for each design point, an off loaded Thiokol TE-M-521 motor was selected for the Thor/Delta mission and an off loaded Thiokol TE-M-616 motor was selected for the Atlas/Centaur mission. Details of the modifications required are presented in subsection 4.3. Available velocity increments as a function of spacecraft mass is presented for the selected, modified, motors in Figure 4-4.

The trajectory selection for the Atlas/Centaur mission has since been changed to a 1978 launch, Type II trajectory, 50° S periapsis, resulting in selection of an on loaded, modified TE-M-521 (Thiokol) motor selection. Details of this baseline design are presented in subsection 4.4.

TABLE 4-4. MANEUVER CAPABILITY OF AVAILABLE MOTORS

Motor	Mass		ΔV = 1069.8 m/s (3510 fps) (Thor/Delta)				ΔV = 1708.0 m/s (5604 fps) (Atlas/Centaur)			
			Total Spacecraft Mass		Orbiter Payload		Total Spacecraft Mass		Orbiter Payload	
	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb
Aerojet SVM-1	91.1	200.8	247.5	545.7	156.4	344.9	172.2	379.7	81.1	178.9
Aerojet SVM-2	164.0	361.6*	449.2	990.4	286.1	630.8	312.9	689.9	149.8	330.3
Aerojet SVM-2 (mod)	164.0	361.6*	457.3	1008.1	293.5	647.0	317.8	700.7	154.0	339.6
Aerojet SVM-5	316.8	698.4*	901.3	1987.0	584.5	1288.6	627.9	1384.3	311.1	685.9
UTC FW-4S	299.6	660.5*	858.9	1893.6	559.3	1233.1	598.1	1318.6	298.5	658.1
UTC FW-5	291.7	643.0*	825.2	1819.3	533.6	1176.3	574.3	1266.2	282.7	623.2
Thiokol TE-M-442	266.5	587.6*	709.5	1564.1	442.9	976.5	496.3	1094.2	229.8	506.6
Thiokol TE-M-521	124.5	274.4	358.4	790.1	233.9	515.7	249.1	549.2	124.6	274.8
Thiokol TE-M-616	362.4	799.0	1074.1	2367.9	711.6	1568.9	745.7	1643.9	383.2	844.9
Hercules X-248	232.6	512.7*	597.1	1316.4	364.6	803.7	420.6	927.3	188.1	414.6
Hercules BE-3-A	98.9	218.0	268.4	591.7	169.2	373.0	187.8	414.1	93.8	206.7
Hercules BE-3-B	111.6	246.0	3059.0	674.4	194.3	428.4	213.7	471.2	107.7	237.4

*Too large to be considered for practical motor modification for the Thor/Delta mission.

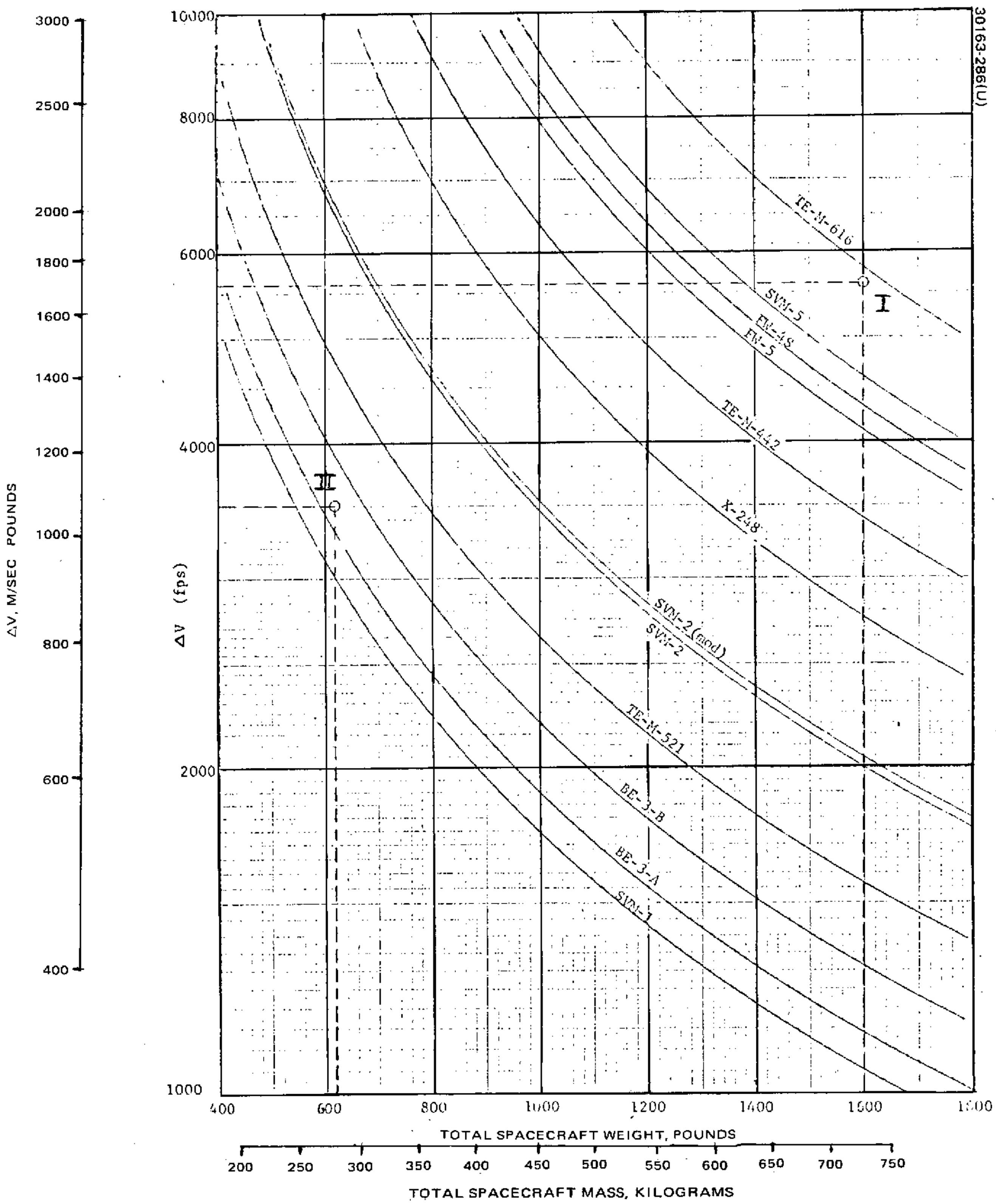

FIGURE 4-3. VELOCITY INCREMENT VERSUS SPACECRAFT MASS

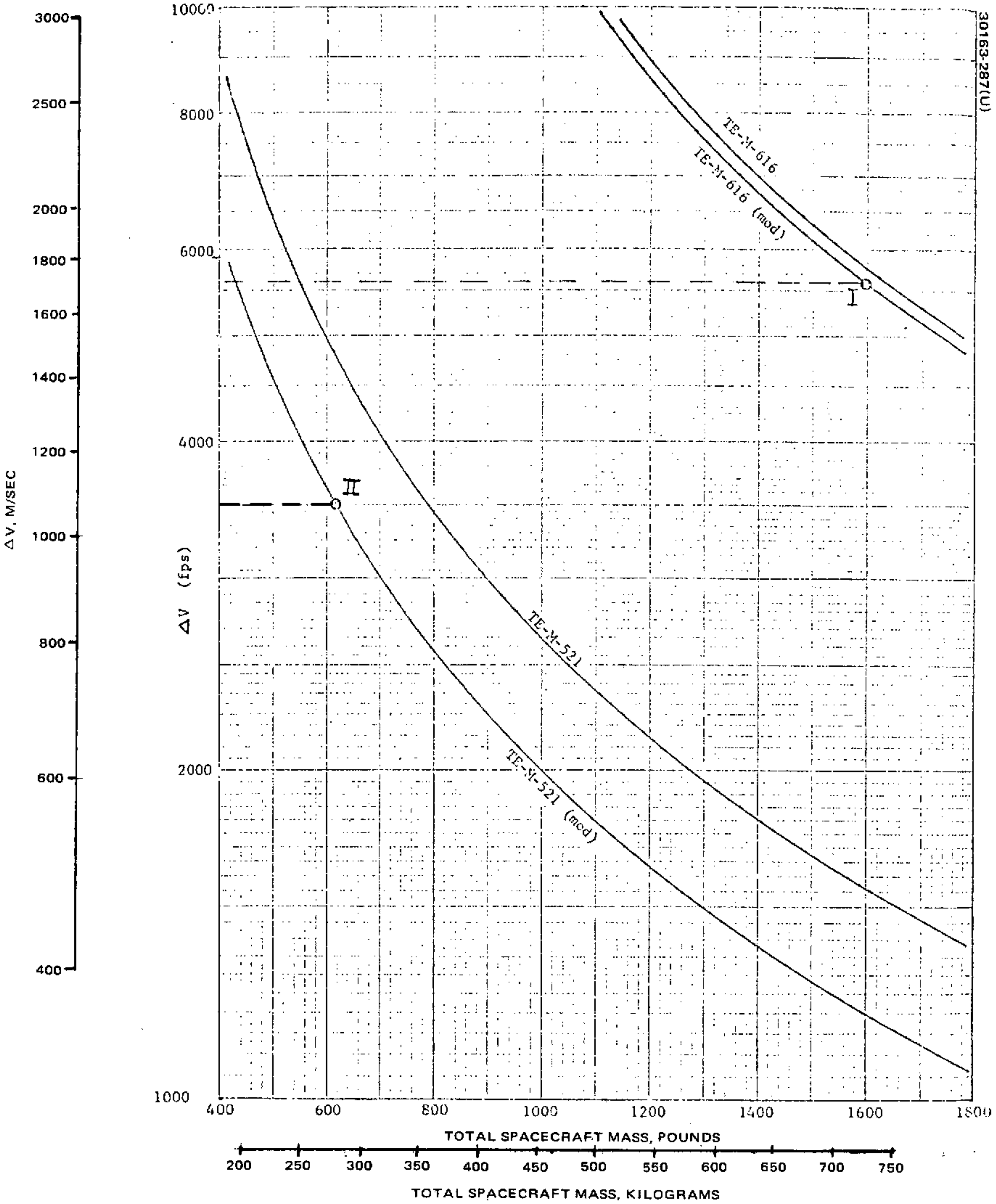

FIGURE 4-4. VELOCITY INCREMENT VERSUS SPACECRAFT MASS FOR MODIFIED SELECTED MOTORS

Another area of consideration under this study was the ability of solid propellant to withstand the space storage, up to 200 days, during Earth-Venus transit prior to motor operation.

Space aging test results by Thiokol Chemical Corporation and United Technology Corporation, on component materials as well as loaded motors, agree that the aging mechanism consisted of the outgassing of volatiles from the surface of exposed materials, resulting in degradation of elastic properties. Most of the outgassing occurred within a few days after entering the vacuum environment. Thus, a surface effect was experienced with no property change to the propellant bulk other than due to the normal aging phenomenon which occurs irrespective of the space or earth-ambient environment. Both companies have concluded that the propellant is capable of withstanding space storage without degraded performance.

4.3 THOR/DELTA BASELINE DESCRIPTION

The baseline orbit insertion motor subsystem for the Thor/Delta mission consists of a Thiokol, Model TE-M-521 motor modified for the required propellant load.

The propellant and expended inert requirement of 88.72 kg (195.6 lb) constitutes a 21 percent propellant offload from the existing TE-M-521 design. A modification consisting of shortening the cylindrical portion of the titanium case was instituted to optimize the motor design for the propellant load.

Note that the TE-M-521 motor design was derived by stretching the TE-M-479 motor (also flight proven). Thus a motor size between these two designs, as required here, is considered a minor modification by the motor vendor. The motor is shown in Figure 4-5.

Motor design parameters, existing as well as modified for the Pioneer Venus mission, are presented in Table 4-4. It should be noted that the minimum operating environmental temperature of -6.7°C (20°F) was imposed based on Hughes judgment and experience. Previous motor qualification has been accomplished at -17.8°C (0°F).

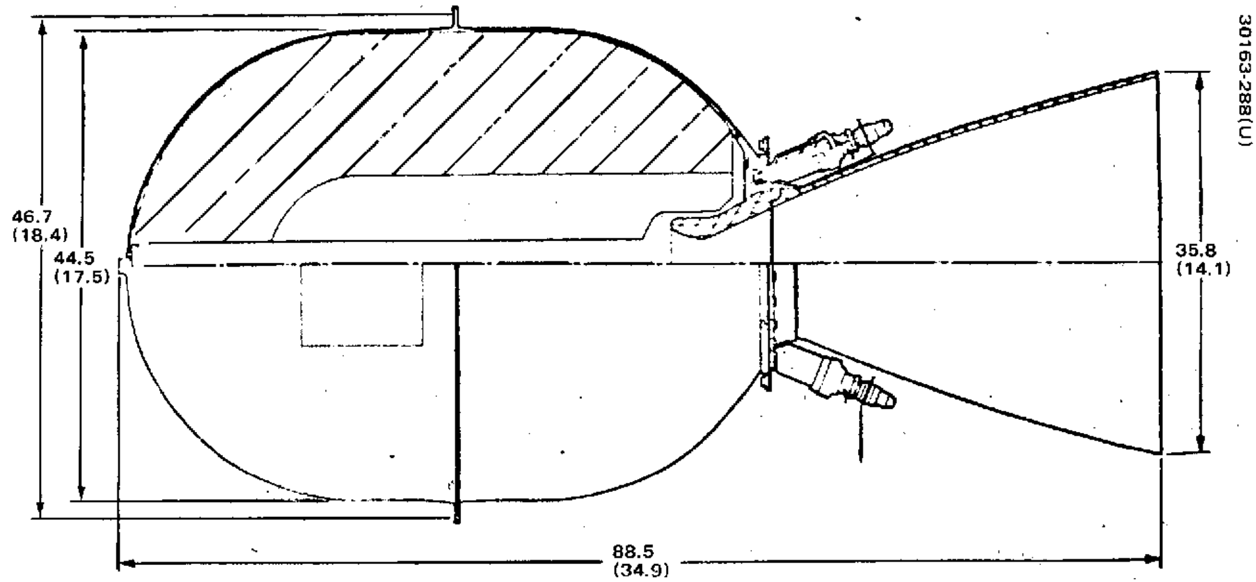

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FIGURE 4-5. THOR/DELTA BASELINE TE-M-521 MOTOR

4.4 ATLAS/CENTAUR BASELINE DESCRIPTION

The baseline orbit insertion motor for the Atlas/Centaur mission consists of a Thiokol Model TE-M-521 motor modified for the required Pioneer Venus propellant load, Figure 4-6. The selection criteria is given in Section 4.2.

The propellant plus expended inert mass requirement of 143.3 Kg (316.0 lb) exceeds the present, existing design for this motor. In order to carry the required expendables, the motor requires a case modification. The present case is constructed of 6Al 4V titanium and consists of a cylindrical section between two hemispherical ends. The modification consists of increasing the cylindrical length approximately 13.0 cm (5.1 in.), with corresponding modification to the internal motor insulation and nozzle design to accommodate the increased mass flow.

The propellant is a carboxyl-terminated polybutadiene formulation cast into an internal burning, eight point star configuration. The high density TP-H-3135 propellant (88 percent solids content) will replace the lower energy TP-H-3062 propellant previously used in this motor. The increased performance is reflected in the expendable mass requirement stated above. This propellant is used in the Model TE-M-616 (Canadian Technology Satellite) motor and has also been successfully tested in the existing TE-M-521 motor.

The material used for the liner between the propellant and insulation is TL-H-304, which has been used successfully in many previous applications. The insulation material is an asbestos-filled polyisoprene (designated TI-R-300) which features a low sensitivity to moisture pickup and reproducible thermal properties.

The nozzle has a contoured configuration with an area expansion ratio of 50. The nozzle incorporates a body fabricated from vitreous silica phenolic and a Graph-I-Tite G-90 throat insert.

A single, head-end mounted pyrogen igniter with dual initiators will be used.

A summary of motor design and performance parameters is presented in Table 4-5.

A throat heater is provided to maintain the motor above the minimum temperature of 4° C (40° F). The heater power requirement for the Atlas/Centaur mission is 20 W. The heater design incorporates redundant circuits for reliability purposes.

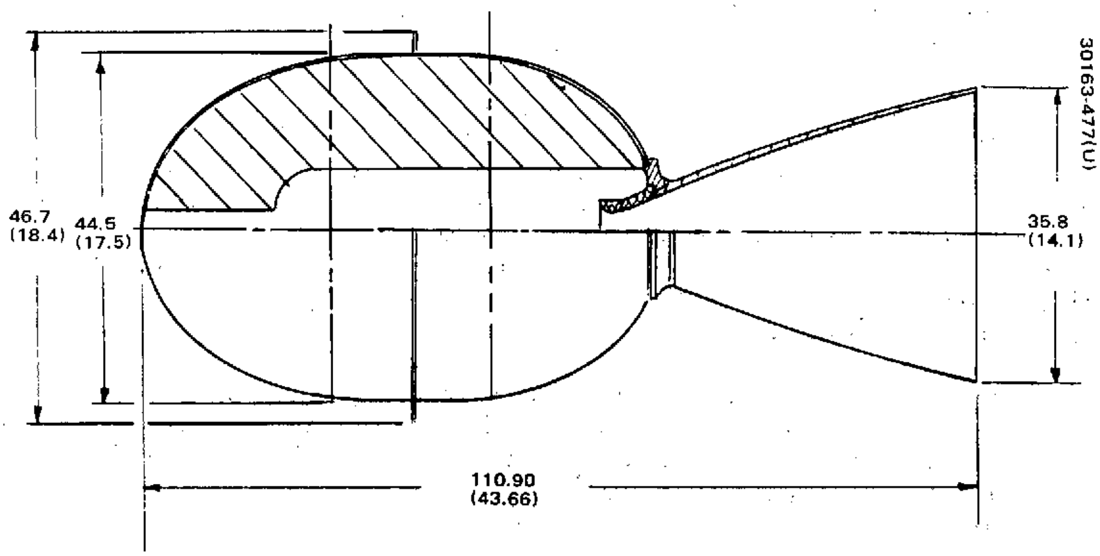

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FIGURE 4-6. ATLAS/CENTAUR BASELINE TE-M-521 ORBIT INSERTION MOTOR

TABLE 4-5. TE-M-521 MOTOR DESIGN PARAMETERS ATLAS/CENTAUR

Performance	
Average thrust	20,527 N (4615 lb)
Motor delivered Isp	289 sec
Mass	
Total propellant mass	141.8 Kg (312.7 lb)
Burned-out motor mass	12.6 Kg (27.7 lb)
Expended inert mass	1.5 Kg (3.3 lb)
Total motor mass	155.9 Kg (343.6 lb)
Mass fraction	0.91
Case	
Material	6Al 4V Titanium
Nozzle	
Body material	Vitreous Silica Phenolic
Throat insert material	Graph-I-Tite G-90
Area expansion ratio	50
Liner	
Material	TL-H-304
Insulation	
Material	TI-R-300
Propellant	
Designation	TP-H-3135
Configuration type	Internal burning, 8 point star
Igniter	
Type	Pyrogen