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FINAL REPORT

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LANDSAT/MMS PROPULSION MODULE DESIGN FINAL REPORT	
AUTHOR(S) J. M. MANSFIELD, F. G. ETHERIDGE, J. INDRIKIS	

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ABSTRACT

EVALUATIONS ARE PRESENTED OF ALTERNATIVE LANDSAT FOLLOW-ON LAUNCH CONFIGURATIONS TO DERIVE THE PROPULSION REQUIREMENTS FOR THE MULTIMISSION MODULAR SPACECRAFT (MMS). TWO BASIC TYPES WERE ANALYZED INCLUDING USE OF CONVENTIONAL LAUNCH VEHICLES AND SHUTTLE-SUPPORTED MISSIONS. IT WAS CONCLUDED THAT TWO SIZES OF MODULAR HYDRAZINE PROPULSION MODULES WOULD PROVIDE THE MOST COST-EFFECTIVE COMBINATION FOR FUTURE MISSIONS OF THIS SPACECRAFT. CONCEPTUAL DESIGNS OF THE SELECTED PROPULSION MODULES WERE PERFORMED TO THE DEPTH PERMITTING DETERMINATION OF MASS PROPERTIES AND ESTIMATED COSTS.

FOREWORD

This report is provided in accordance with the requirements of Contract NAS5-23524. The data and analyses were prepared by the Space Division of Rockwell International for the Goddard Space Flight Center of the National Aeronautics and Space Administration. The report is printed in three volumes:

I. Task 4.3 - Trade Studies

II. Task 4.4 - Concept Design

III. Appendix - Cost Analyses

The following individuals contributed to this report: J. Indrikis, E. Katz, R. Yee, F. Etheridge, and J. Mansfield.

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INTRODUCTION

The Multimission Modular Spacecraft (MMS) is being developed by the Goddard Space Flight Center to achieve cost savings in future unmanned earth orbiting space projects through the utilization of a Shuttle-compatible standardized modular spacecraft. One of the early missions being considered which might utilize this approach is a follow-on to the current Landsat. If adopted, this mission would potentially be the first MMS application to require a propulsion subsystem. The Space Division of Rockwell International has performed a series of analysis and design tasks to define a modular propulsion subsystem concept which will be compatible with the MMS and satisfy the Landsat follow-on mission propulsion requirements.

The initial portion of this effort concentrated on the evaluation of alternative Landsat follow-on launch configurations to establish the propulsion requirements and the performance of trade studies of the propulsion subsystem elements to select the most cost effective sizing approach to meet the variations in requirements. This report summarizes these analyses which were utilized in the preparation of conceptual designs of the propulsion module.

SUMMARY AND CONCLUSIONS

Two basic types of Landsat follow-on missions were analyzed to derive the propulsion requirements. The first involves launch and delivery to the operational orbit by a conventional launch vehicle such as the Delta 3910. In this mode, the MMS propulsion subsystem must provide for correction of initial orbit errors, periodic adjustments to compensate for aerodynamic drag, and back-up attitude control for special situations. The analyses of this mode have concluded that utilizing a combination of 0.2-lbf and 5.0-lbf hydrazine thrusters in in a blowdown system, the requirements for a three-year mission can be met by a total of 61.1-lb of hydrazine without any allowances for reserves or unspecified contingencies.

The other mission approach involves the use of the Shuttle to deliver the spacecraft to some intermediate altitude and the subsequent transfer to the operational altitude by the MMS propulsion. It was found that with the current formula for computing the relative portion of the Shuttle launch cost to be borne by a payload, the optimum altitude for both Shuttle deployment and retrieval is at an altitude which does not require a supplemental OMS propellant kit in the cargo bay. An altitude of 150 n.mi. was chosen to avoid the rapid buildup of orbit perturbations due to drag at lower altitudes. Computations were made of the propellant requirements for the orbit transfers including some allowance for nominal off-set c.g's, and again, utilizing 0.2-lbf and 5.0-lbf hydrazine thrusters in a blowdown system. When combined with the nominal three-year mission requirements previously derived, a total propellant quantity of 1027.6-lb was indicated for this mode.

The propulsion system analysis was initiated by identifying all subsystems and components required to synthesize the baseline propulsion modules for the two mission modes including a growth version which utilized interior volume of the basic MMS structure. Potential suppliers of key elements, thrusters and tank systems, were formally contacted for supporting technical data and Rough Order of Magnitude (ROM) cost data.

Utilizing the propellance requirements and mission modes derived in the mission analyses, representative propulsion systems were derived which were basically compatible with the system requirements including the constraints derived from the MMS and the Shuttle. This report describes eight potential configurations and variations thereof. The final conceptual designs are described in another volume.

1.0 DELTA MISSION

1.1 REQUIREMENTS

The Landsat Follow-On Observatory is injected into orbit at its operational altitude (380.6 n.mi) by a conventional launch vehicle such as the Delta 3910. The Multimission Modular Spacecraft (MMS) must provide

- orbit adjust capability to correct for launch vehicle injection errors.
- orbit maintenance to keep the repeating ground track within ± 2.7 n.mi. (± 5 km) for a period of three years.
- reaction control capability for initial stabilization plus three restabilization maneuvers of the observatory.
- operation in a safe hold mode necessary for emergency retrieval of the observatory.

For subsequent analyses the following mission orbital parameters, spacecraft and environment characteristics were assumed:

Orbit Parameters	
Epoch	1 October 1980 - Midnight
Altitude	380.67 n.mi. (705 km)
Eccentricity	0
Inclination	98.2
Geographic Longitude of Ascending Node	42.6° E
Spacecraft	
Weight	3564 lbs
Moments of inertia	I_x - 1652 slug ft^2
	I_y - 2472 slug ft^2
	I_z - 2158 slug ft^2
	I_{xy} - 119 slug ft^2
Dimensions	(See figure 1)
C_D	2.5
Magnetic Dipole Moment	5000 pole-cm

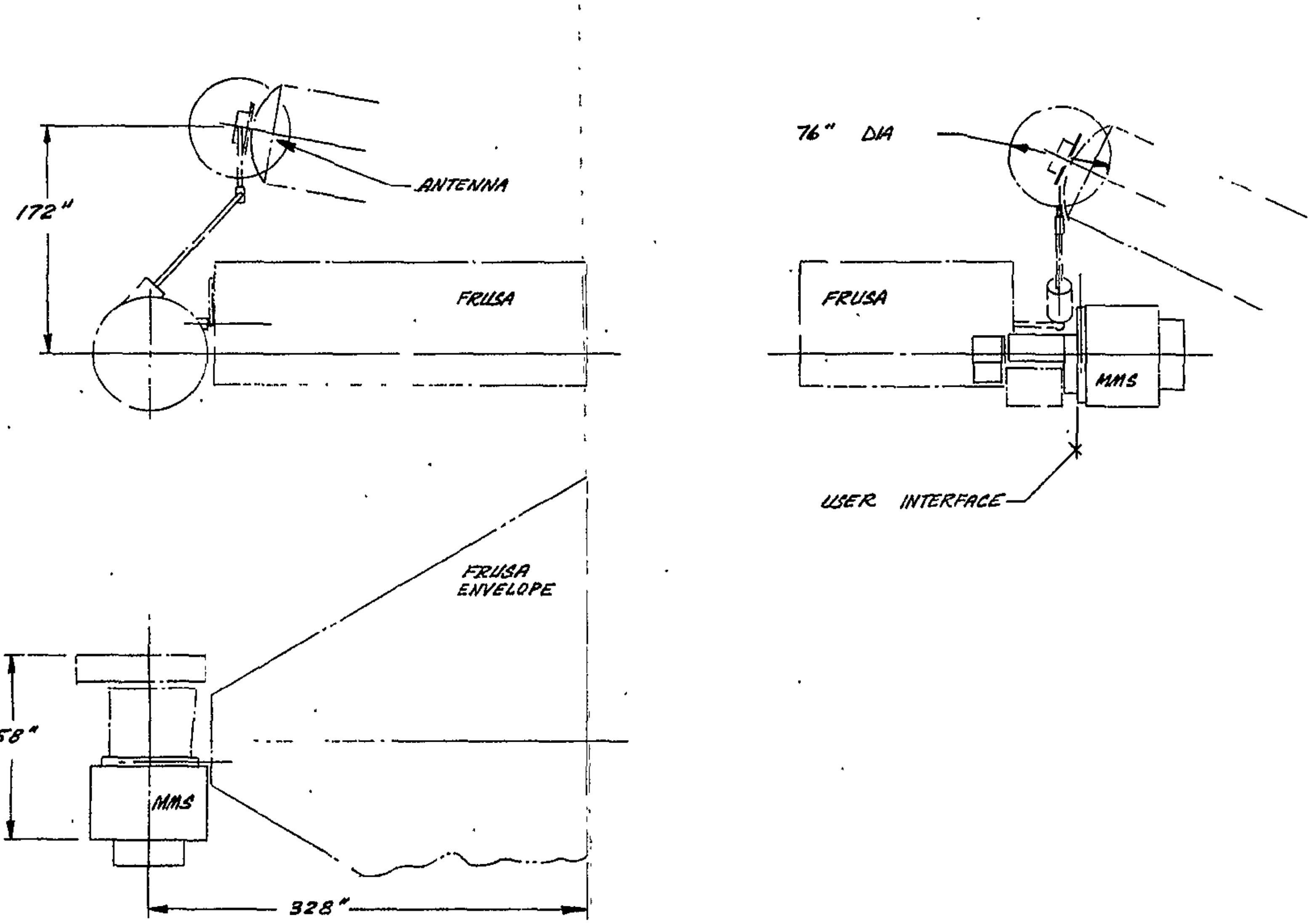


Figure 1. Spacecraft Dimensions

Environment

Atmospheric Density 1.1×10^{-15} slug/ft^3

Solar Pressure Constant 9.4×10^{-8} lb/ft^2

1.2 INJECTION ERROR CORRECTION

A conventional launch vehicle such as the two-stage Delta 3910 injects the observatory in its operational orbit. Certain inaccuracies in orbital parameters will occur as a result of off nominal operation of the launch vehicle. Typical two-stage Delta vehicle accuracies for circular orbits between 100 and 1000 n.mi. are as follows:

- orbit altitude (deviation from circular) $\leq$ 10 n. mi
- orbit inclination (deviation from desired) $\leq$ 0.05 deg.

These three sigma data obtained from Reference 1 are based on Hohmann transfer flight mode with second stage restart to circularize the orbit.

The reference document cautions the user that the above data should be used as general accuracy indicators only. Detailed analyses are performed for each specific mission, including the effects of individual mission requirements, to define more precisely the accuracy to be expected.

A comparison of the above injection error data was made with accuracy data obtained from actual missions flown by the Delta launch vehicle. It was concluded that the use of the accuracies presented in Reference 1 is probably conservative.

A ΔV of 26.9 fps would be required to correct a circular orbit altitude deviation of 10 n.mi. and an inclination deviation of 0.05 degrees. The propellant weight required to perform this maneuver can be calculated by

$$W_{PR} = W_o \left(1 - e^{-\frac{\Delta V}{g\, Isp}}\right)$$

where

W_o = initial weight = 3564 lbs.

I_{sp} = specific impulse = 230 sec.

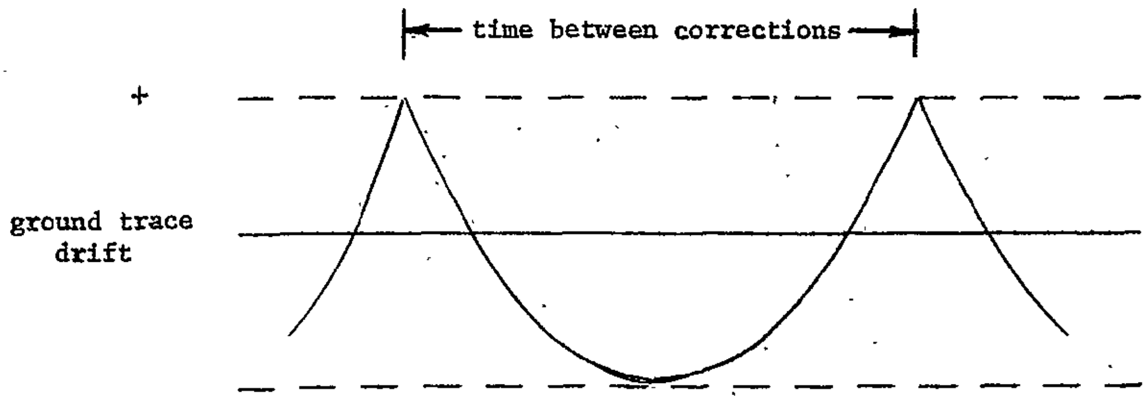
This value was used as an average for simplicity. Variations due to blowdown will be small and are considered to be within the uncertainty range of these preliminary estimates.

The maximum propellant weight to correct the orbit injection errors resulting from off nominal operation of the Delta launch vehicle is 12.9 lb. The orbit correction will be performed with at least one or two distinct thruster (5 lbs.) operations. The number of operations depend on the type of error to be corrected. For example, if perigee altitude is already at the operational orbit altitude, only a single burn maneuver would be required to bring down the apogee and circularize. It is not forseen that these orbit adjust maneuvers will normally require the pulse mode of thruster operations. At least one full orbit will be required after injection to provide sufficient tracking and orbit data to command the orbit adjust maneuvers.

1.3 ORBIT MAINTENANCE

The Multimission Modular Spacecraft Propulsion System must provide orbit maintenance (stationkeeping) capability so as to keep the repeating ground track within ± 2.7 n.mi (± 5 km).

The maximum time between stationkeeping corrections occurs when the drift rate just after correction is just enough to cause the ground track to drift to the opposite limit and have the perturbing forces turn it around at that point. The sketch below illustrates this concept.



Analytical relationships were used to estimate orbital drift and maintenance maneuver requirements. To verify the analytical calculations the Rockwell International GETOP program was used to propagate the mission orbit by numerical integration of the equations of motion. The GEOTOP perturbation model includes aspherical earth, solar and lunar gravity, solar radiation, and atmospheric drag. During the analysis, it was found that atmospheric drag was the primary, in fact, the only perturbing force that mattered as far as stationkeeping analysis was concerned. The drag force causes the semimajor axis to decrease resulting in an eastward drift of the ground track longitude.

The results of the analytical solution and GEOTOP were in close agreement. The more conservative of the two was retained for this study. The results of the orbit maintenance requirements can be summarized as follows:

Time between maneuvers	15.6 days
(or) 24 maneuvers per year	(rounding up to the next maneuver)

The altitude change due to drag over this 15.6 day cycle is $\Delta a = 0.324$ n.mi and hence the

$$\Delta V \text{ per maneuver} = \frac{V_o}{2} \frac{\Delta a}{a_o} = 1.04 \text{ fps}$$

or 25 fps/year

In three years 75 fps

$$\text{or} \quad W_{PR} = W_o \left(1 - e^{-\frac{\Delta V}{g I_{sp}}}\right) = 35.9 \text{ lb.}$$

This equates to approximately 0.5 lb/maneuver.

1.4 STABILIZATION MANEUVERS

The stabilization maneuvers are identified by coarse sun acquisition requirements. The initial conditions for the maneuver are random altitude and initial rates of up to two degrees per second about each control axis. Utilizing the MMS Propulsion System the maneuver must be completed within 10 minutes.

The mechanics of initial solar acquisition is quite complex and depend on a number of factors in the attitude control system not yet identified. The critical parameter is the time to complete the maneuver. Preliminary analysis indicates that this maneuver will be readily accomplished within the 10 minute time interval alloted.

Even with three RCS motors operating continuously during this maneuver not more than 1.6 lb of propellant would be expended. For this reason, more detailed analysis was assumed to be unwarranted.

1.5 SAFE HOLD OPERATION

The safe hold mode of operation consists of

- aligning the coarse sun sensor reference axis relative to the line from the spacecraft to the sun in less than 10 minutes
- maintaining spacecraft control for a period of 30 days
- transferring to inertial control mode in less than 5 minutes
- maintaining the spacecraft attitude for 1 hour to meet shuttle retrieval requirements

Rockwell International MIDAS program was used to determine the perturbing torques experienced by the spacecraft. The program is capable of accounting for solar radiation pressure, aerodynamic, magnetic dipole, and gravity gradient torques. The unbalanced torques averaged over one orbit due to the above phenomena are shown in Table 1.

Figure 2 illustrates the variation of perturbing torques for the three spacecraft axes as a function of time (single orbit).

Table 1. Average Torques Over One Orbit (Non-Return Mission)

Solar radiation pressure torque	ft-lb	2.02×10^{-4}
Atmospheric drag torque	ft-lb	6.97×10^{-4}
Magnetic dipole torque	ft-lb	1.61×10^{-4}
Gravitational torque	ft-lb	5.36×10^{-4}
Gravitational torque (1° attitude error)	ft-lb	0.89×10^{-4}

The average perturbing torques for the three spacecraft axes are

Yaw torque	11.03×10^{-4} ft-lb
Pitch torque	3.07×10^{-4} ft-lb
Roll torque	1.03×10^{-4} ft-lb

These include the gravitational torques that result from non-zero product of inertia as well as a 1° error in spacecraft attitude (all three axes).

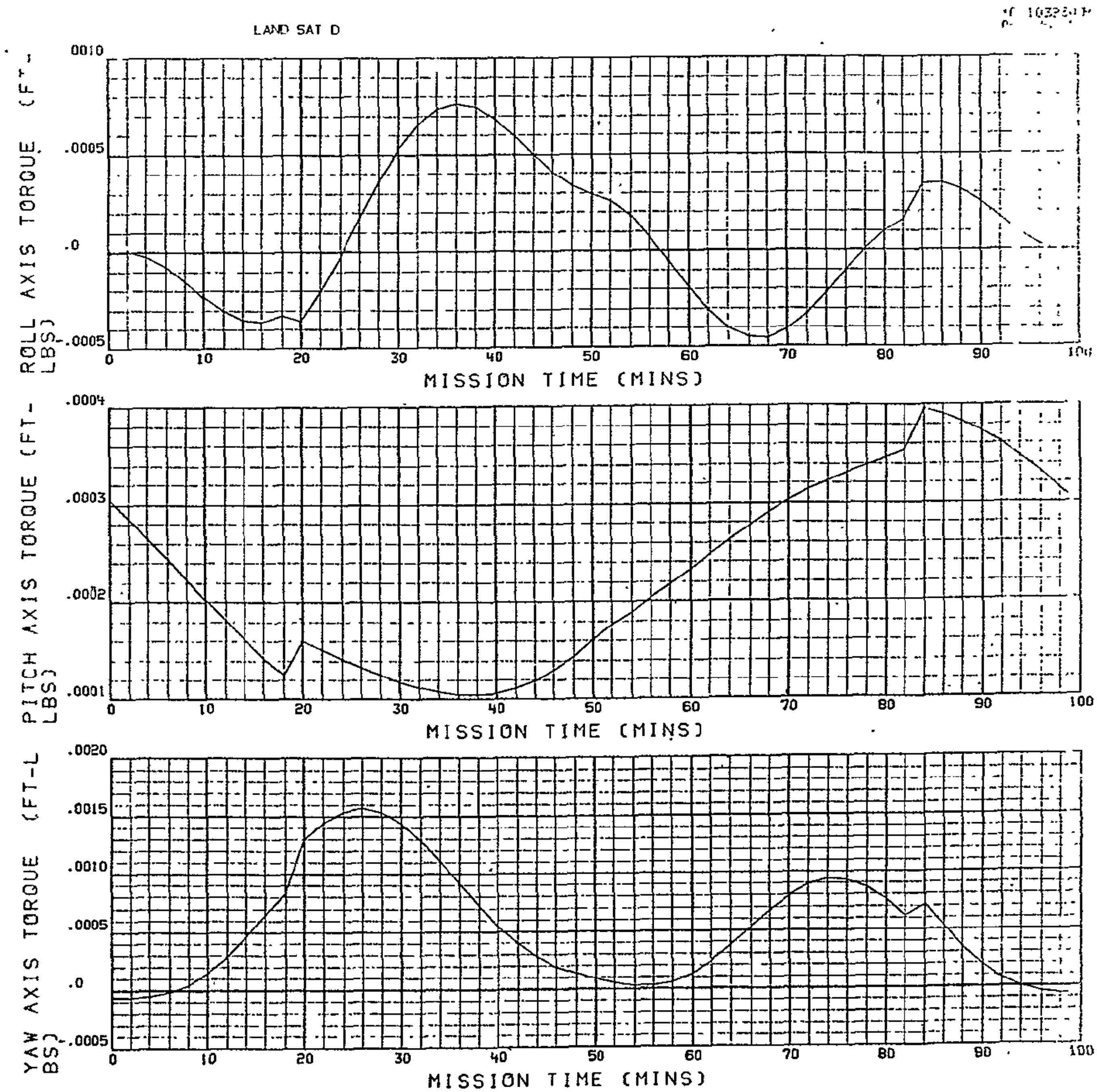


Figure 2. Landsat Perturbing Torques

Over a period of 30 days the momentum that has to be balanced by the RCS thrusters is

Yaw momentum	2858 ft-lb-sec
Pitch momentum	795 ft-lb-sec
Roll momentum	266 ft-lb-sec

The control moment available from the RCS thrusters (two thrusters used per axis) is 0.8 ft-lb in roll and approximately 2.2 ft-lb in pitch and yaw. Based on the above values

- yaw control will required 1300 sec of thruster operation
- pitch control will require 360 sec of thruster operation
- roll control will require 330 sec of thruster operation

The above times equate to approximately 3.5 lb of propellant to be expended.

The total propellant requirement for the safe hold mode is

- 1.6 lb for coarse sun acquisition (10 min)
- 3.5 lb for safe hold maintenance (30 days)
- 0.8 lb for transfer to inertial hold (5 min)
- trace for shuttle retrieval mode (1 hour)
- 5.9 lb total expenditure

1.6 SUMMARY

For the MMS mission where the spacecraft is injected into its operational orbit by a conventional launch vehicle the on-orbit propellant requirements are summarized in Table 2. The order of presentation is indicative of the sequence in propellant expenditure.

It should be noted that the total amount of 61.1 pounds shown in the table does not include any propellant for reserves or other contingencies.

Table 2. Propellant Requirement Summary

Orbit injection error correction	12.9 lb
Coarse sun acquisition maneuver	1.6 lb
Orbit maintenance (3 years)	35.9 lb
Three additional coarse sun acquisition maneuvers	4.8 lb
Safe hold mode	5.9 lb
Total	61.1 lb

2.0 SHUTTLE MISSION

2.1 REQUIREMENTS

The Landsat Follow-on Observatoty is injected into orbit by use of the Shuttle. This configuration is one that would be launched and possibly recovered by the Shuttle at an intermediate orbit altitude.

The Multimission Modular Spacecraft must provide

- orbit transfer capability to the operational altitude
- orbit transfer capability back to a parking orbit for Shuttle retrieval
- control authority during the ascent and descent trajectory
- on-orbit propulsion requirements as already defined for the Delta mission.

The concept of least overall cost to the Government is used to determine the most desirable parking orbit.

2.2 ORBIT TRANSFER

The thrust provided by the baseline MMS Propulsion System (approximately 20 lbs) results in a mission thrust-to-weight ratio of 0.006 - 0.002. Such low thrust-to-weight ratios during ascent to a higher orbit results in a special class of spiral trajectories. This class of ascent trajectories is bounded by the multiturn spirals resulting from very low thrust-to-weight ratios ($\leq 10^{-4}$) on one end and the two impulse Hohman transfer ellipses with a long coast period on the other.

A Rockwell International trajectory program was used to generate total velocity requirements as a function of vehicle thrust-to-weight ratio. The existence of "optimum" thrust-to-weight ratios with strong dependence on mission characteristics was identified (Figure 3). For these "optimum" thrust-to-weight ratios and mission combinations the finite burn velocity approached the minimum "impulsive burn" velocity requirement.

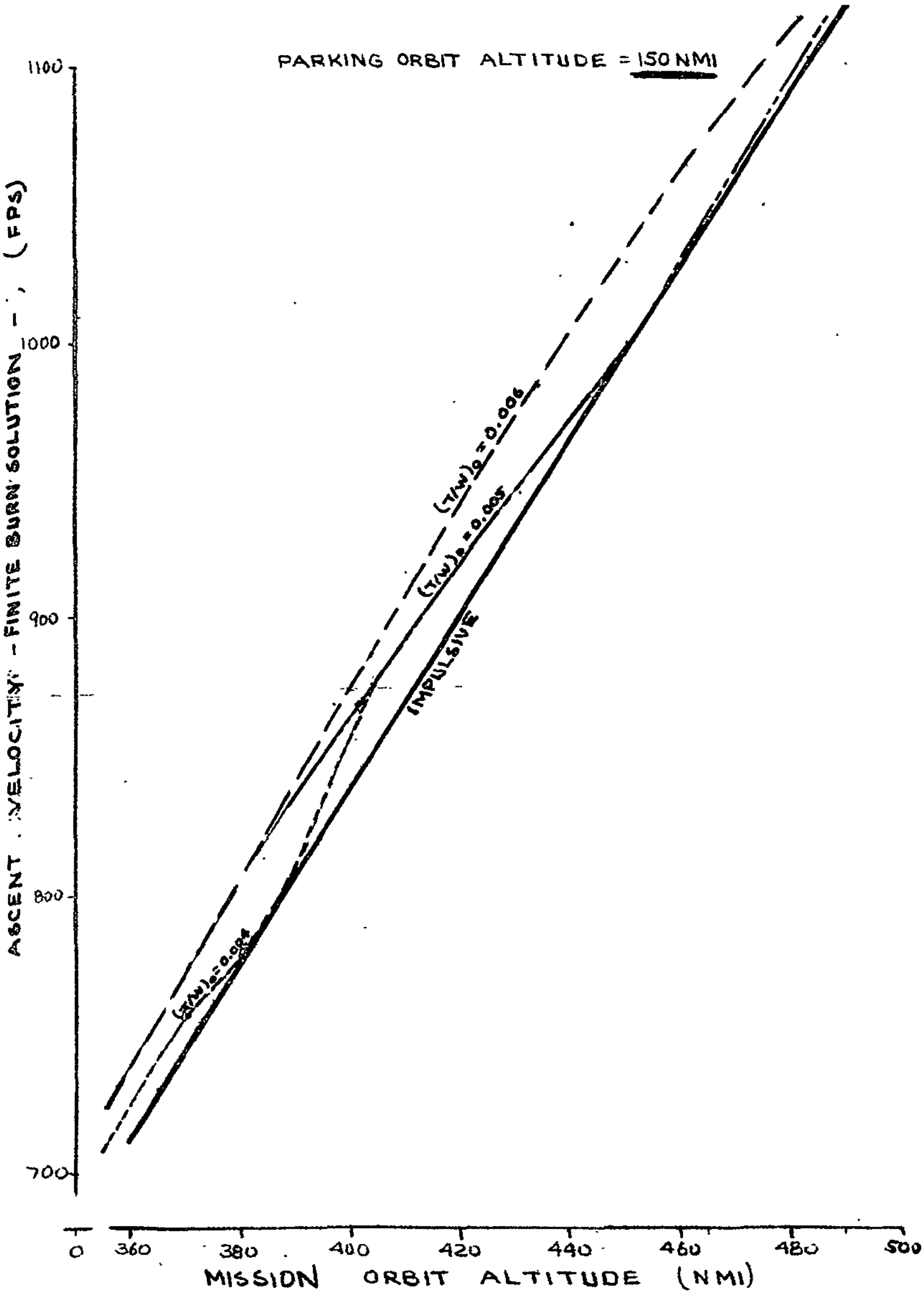


Figure 3. Orbit Transfer Velocity Variation With Thrust-to-Weight Ratio

It is suspected that this occurs when the transfer burn angular range is close to 360,720 and possibly 1080 degrees. Further analysis in this area would be desirable since such low thrust-to-weight ratios may be experienced for other proposed orbit transfer mission; for example, for the large space structures.

For the Landsat mission attitudė of 380.6 n.mi the difference between impulsive and finite burn velocity requirements at 0.006 thrust to weight ratio was approximately 4 percent. This factor was then used to bound all subsequent parking orbit/Landsat mission orbit combinations.

The orbital transfer velocity requirements are shown in Figure 4 as a function of parking orbit altitude. Both the impulsive and approximated finite burn requirements are shown. These velocity requirements converted to propellant weight needed are presented in Figure 5. Both the ascent only and the ascent and descent missions were analyzed. It was assumed that the velocity requirement for the ascent/descent mission is twice the velocity required for ascent only. Since the descent propellant has to be carried during ascent, this results in more than doubling the required propellant.

It should be noted that the propellant requirement thus calculated is for orbit transfer only. It does not include propellant required for vehicle control. This subject will be addressed in subsequent sections.

2.3 ATTITUDE CONTROL REQUIREMENTS

Trajectory analyses have shown that efficient transfers may be accomplished even though the MMS Propulsion System thrust-to-weight ratio (using (4) five pound thrusters with a blow-down ratio of 3:1) is very low (.002 - .006) by conventional standards. These analyses have also shown that the MMS would be in powered flight for nearly the entire transfer period. Figure 6 shows an example transfer orbit. The duration of the powered flight segments suggested a possible impact upon the vehicle's attitude control system (ACS) requirements. For this reason, a preliminary assessment was made of the MMS ACS specification, Reference 2, to evaluate the MMS ACS compatibility with the propulsive requirements.

The general conclusion was reached that compatibility exists in all respects between the provisions of Reference 2 and the MMS propulsive requirements for orbital transfer.

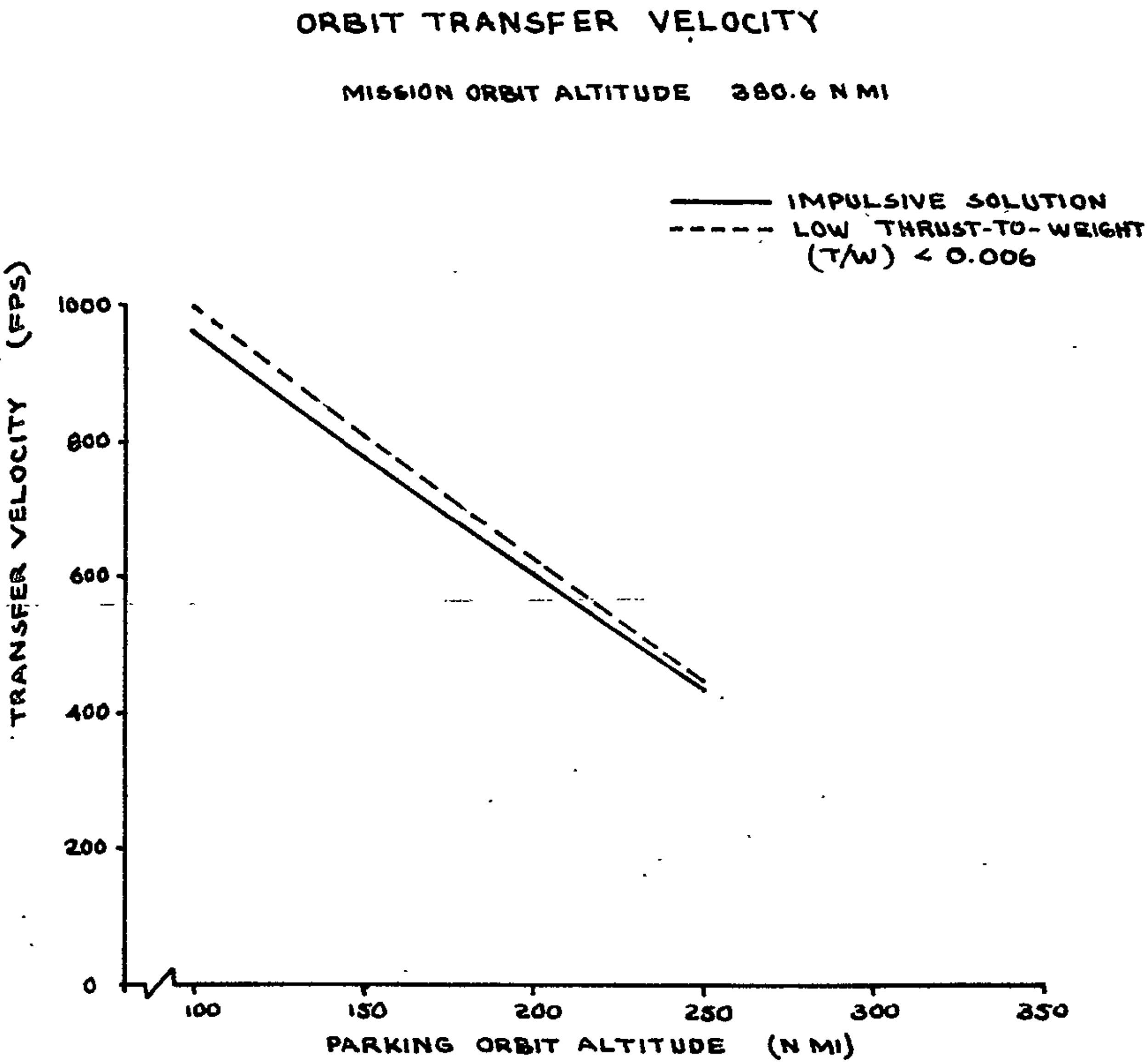


Figure 4. Orbit Transfer Velocity

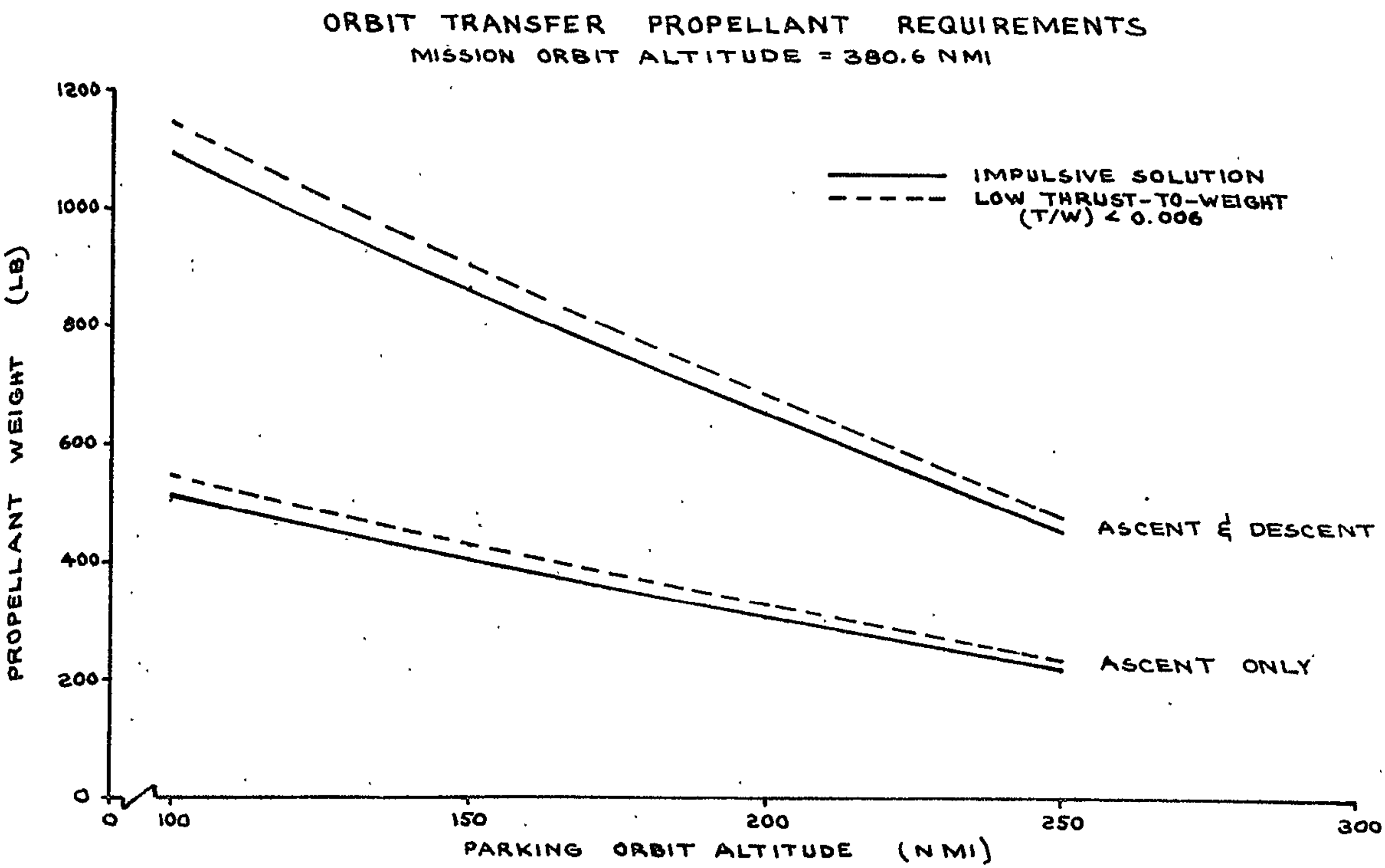


Figure 5. Orbit Transfer Propellant Requirements

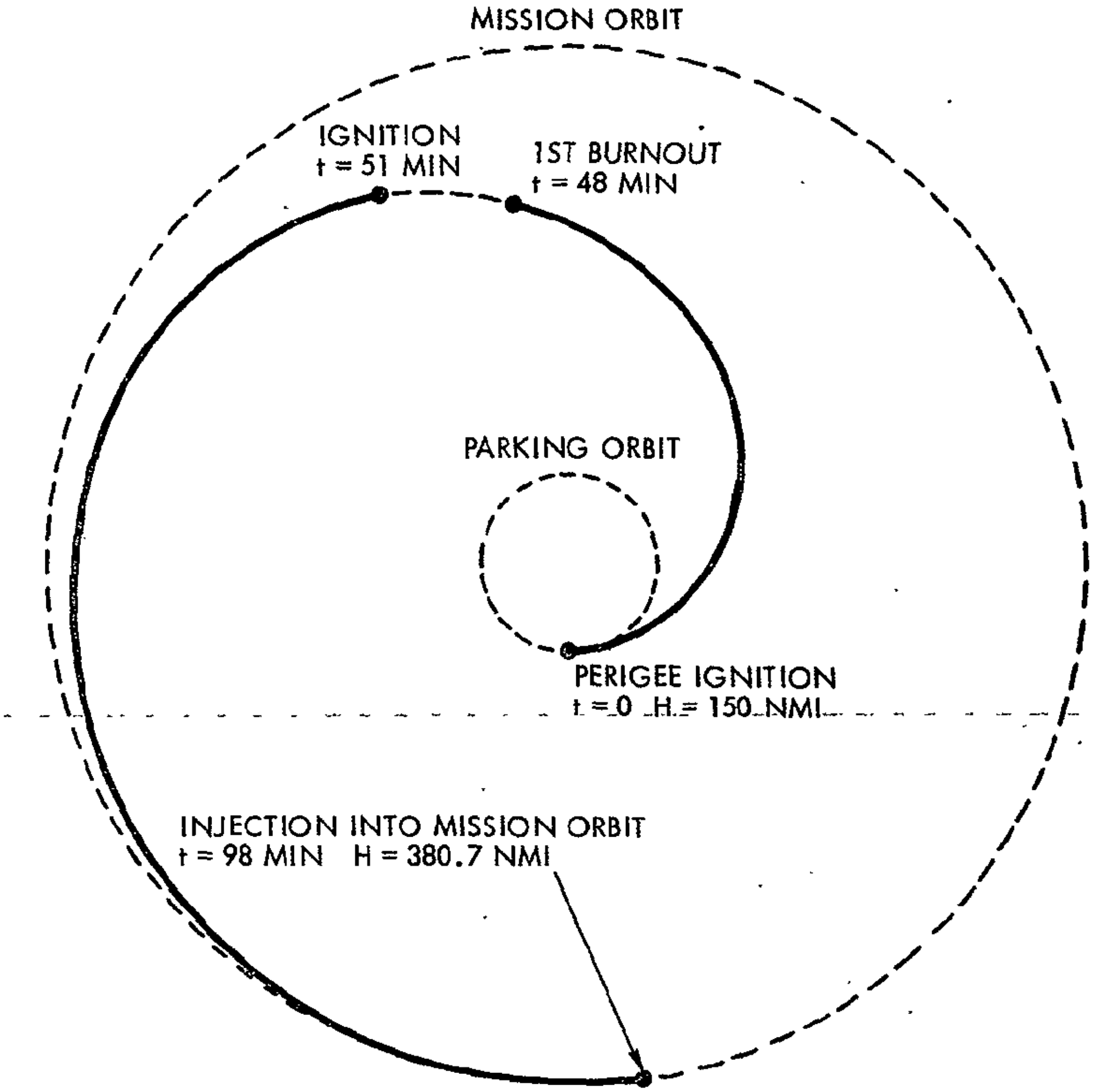


Figure 6. MMS Orbit Transfer

The following discussion will treat the various phases of the MMS/Landsat mission in respect to the interaction between propulsive and ACS requirements.

2.3.1 Pre-Launch

A series of post-separation flight commands must be generated for a time-sequenced program of various maneuvers/flight modes which the MMS will be required to execute subsequent to separation from the launch vehicle. The program will be a function of the predicated time profiles of thrust levels and tolerances, thruster alignments, and specific impulse. The MMS Modular Communications and Data Handling (C&DH) subsystem will have the capacity for storing these commands in the memory of its computer section.

2.3.2 Launch

In the case of a Shuttle launch where a number of Shuttle orbits may be performed prior to MMS separation, there may be a requirement to update the MMS flight program stored in the C&DH subsystem. There will be a signal interface between the Shuttle and the MMS which will allow the stored program to be updated.

2.3.3 Separation

At separation, the MMS ACS will be enabled to bring the reaction wheels up to speed. However, the thrusters should be inhibited until there is safe clearance between the MMS and the Orbiter. This clearance would probably be effected by a combination of a mechanical ejection device and by Orbiter maneuvers.

2.3.4 Post-Separation

During this period until perigee ignition, the MMS must be oriented to that attitude required for perigee ignition. In addition, any deviations in the MMS orbit from that pre-specified must be determined so that an update may be input to the C&DH subsystem. The ACS and C&DH subsystems will have the capacity to effect these functions.

Immediately following separation, the Acquisition Mode of the ACS will orient the MMS so that its solar array can generate power and so that the vehicle's attitude may be determined with respect to stellar, inertial, and earth-centered coordinates. The Slew Mode will then orient the vehicle, using reaction wheel torque, to align its X-axis locally horizontal (along the flight vector) and its Z-axis locally vertical. For up to several orbits (the exact number to be determined in later studies), the ground stations will track and precisely determine the MMS orbit. If warranted, an update command for program changes will be transmitted to the C&DH subsystem. At the appropriate time, the vehicle will be programmed to slew to the inertial position required for perigee ignition.

2.3.5 Perigee Burn

During this period, the four 5-pound (nominal) thrusters must provide accelerating thrust and control about the Y (pitch) and Z (yaw) axes. The X-axis (roll) control must be effected by the low-level thrusters. The thrust and specific impulse used to generate the sample trajectory is shown in Figure 7.

To satisfy these requirements, the ACS will execute the Orbit Transfer Mode. In this mode the computer within the C&DH subsystem will control the thruster duty cycles as necessary to provide the required change in velocity (ΔV) and the vehicle's orientation. The resulting duty cycles will account for thruster unbalance, thruster misalignments relative to the vehicle's center-of-gravity, and various disturbance torques. The required ΔV will be determined by ground processing and controlled by means of total thruster activation time.

2.3.6 Coasting

During the short (approximately 3 minutes) coasting period prior to apogee ignition, the MMS must be oriented to the appropriate attitude. The ACS will slew the vehicle to this attitude per the program stored in the C&DH.

2.3.7 Apogee Burn

Ignition must be commanded at the time and attitude required to assure successful insertion into the operational orbit. The requirements and provisions associated with this phase are fundamentally the same as those given previously for the Perigee Burn.

2.3.8 Orbit Adjust

Following the apogee burn-out, the resulting orbit of the MMS must be determined by ground tracking. If the orbit is outside acceptable tolerances, the vehicle must be commanded to perform an orbit adjust

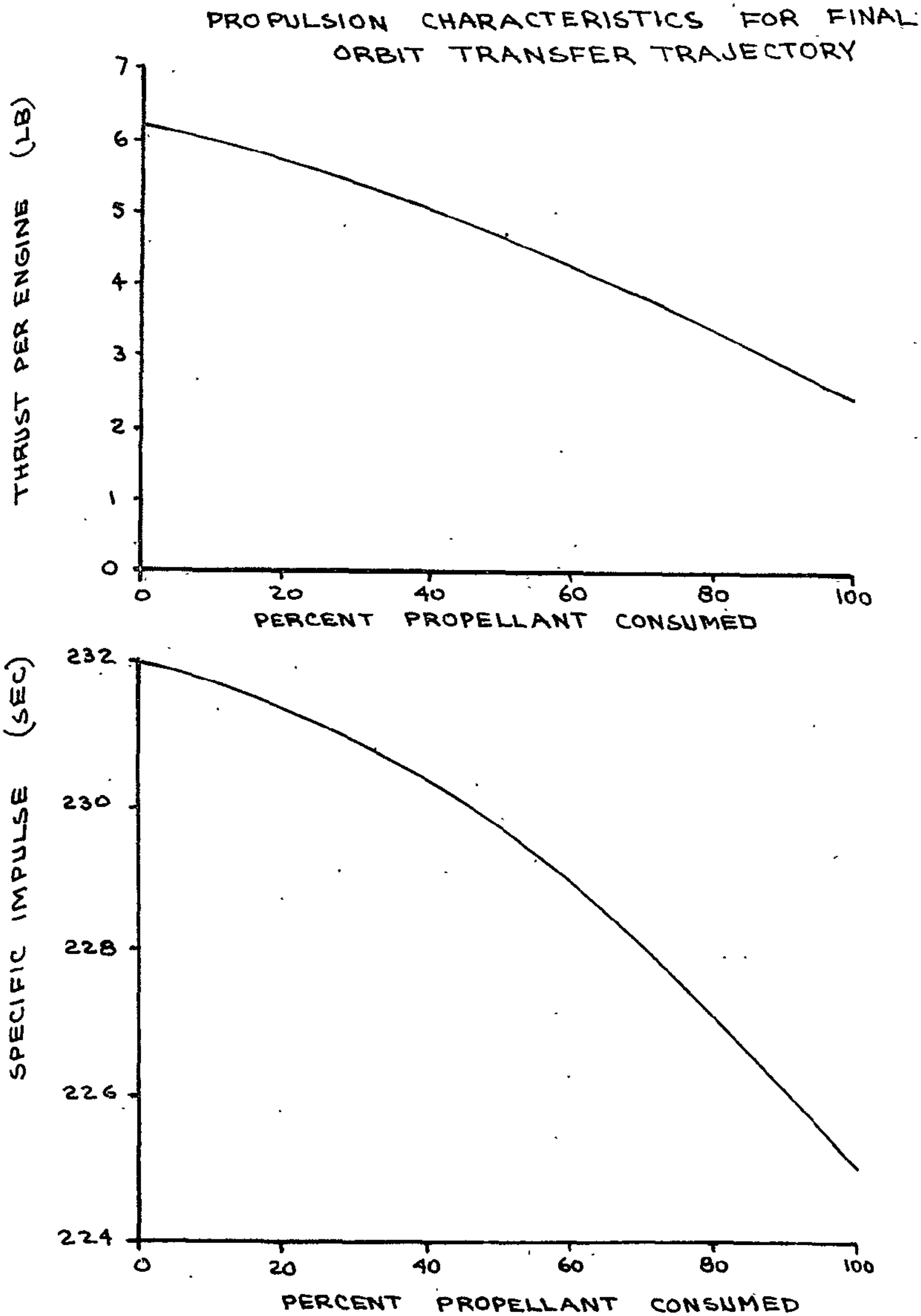


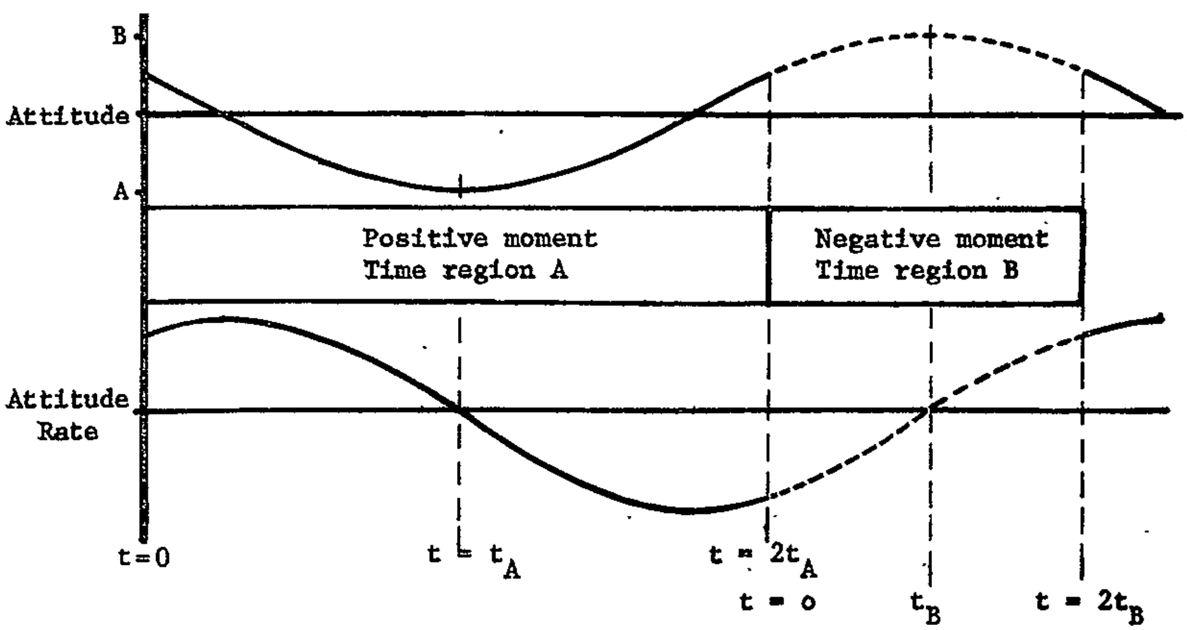
Figure 7. Propulsion Characteristics for Final Orbit Transfer Trajectory

maneuver. The C&DH subsystem will have the capacity to accept a ground generated program for the maneuver. The ACS would then execute a series of steps to slew the vehicle to the appropriate attitude for orbit-adjust burn. At the programmed time, the ACS will actuate the Orbit Adjust Mode and the thrusters will commence burning until the adjust ΔV has been achieved. Following this maneuver, the MMS would be slewed to the attitude pre-specified for operational orbit.

Propellant requirements for control authority during the long ascent burn are estimated in Section 2.4.

2.4 CONTROL AUTHORITY PROPELLANT REQUIREMENTS

For control authority during the orbit transfer operations a thruster pulsing mode will be employed. Unbalanced pitch and yaw moment (as a result of center of gravity offset, thrust misalignment, etc.) will be compensated by pulsing the appropriate orbit transfer thruster (5 lb). Roll motion will be limited by the use of the low level control thrusters (0.2lb). In all cases the vehicle attitude about the respective axis will be allowed to oscillate between $\pm$ 3 degrees. The attitude and attitude time histories will exhibit general sinusoidal characteristics as shown in the sketch below.



The two regions where positive and opposing or negative moment is experienced are indicated. By appropriately switching on and off the proper orbit transfer thrusters (5 lb) control of the vehicle around the pitch and yaw axis can be attained.

The moment or thruster switching time can be determined in the following manner.

The attitude and the attitude rate of the spacecraft around any of the three axis of rotation can be expressed as

$$\theta(t) = \theta_o + \dot{\theta}_o t + \int_o^t \dot{\theta}\, dt$$

$$\dot{\theta}(t) = \theta_o + \dot{\theta}$$

For the case of constant torque or moment

$$\dot{\theta} = \frac{T}{I} t$$

where

T = torque or moment

I = moment of inertia

Substituting the above relationship in the general attitude and attitude rate relationships and then integrating

$$\theta(t) = \theta_o + \dot{\theta}_o t + \frac{1}{2} \frac{T}{I} t^2$$

$$\dot{\theta}(t) = \dot{\theta}_o + \frac{T}{I} t$$

Applying the above relationships to the region of positive moment (time region A)

$$\theta(t) = \theta_o + \dot{\theta}_o t + \frac{1}{2} \frac{T_A}{I} t^2$$

$$\dot{\theta}(t) = \dot{\theta}_o + \frac{T_A}{I} t$$

The initial values can be obtained in the following manner

at $t = t_A$

$$\dot{\theta}(t) = \theta = \dot{\theta}_o + \frac{T_A}{I} t_A$$

$$\dot{\theta}_o = - \frac{T_A}{I} t_A$$

and

$$\theta(t) = A = \theta_o - \frac{T_A}{I} t_A^2 + \frac{1}{2} \frac{T_A}{I} t_A^2$$

$$\therefore \theta_o = A + \frac{1}{2} \frac{T_A}{I} t_A^2$$

Thus for time region A

$$\theta_A(t) = A + \frac{T_A}{I} \left(\frac{1}{2} t_A^2 - t_A t + \frac{1}{2} t^2\right)$$

$$\dot{\theta}_A(t) = - \frac{T_A}{I} t_A + \frac{T_A}{I} t$$

Similarly for time region B

$$\theta_B(t) = B + \frac{T_B}{I} \left(\frac{1}{2} t_B^2 - t_B t + \frac{1}{2} t^2\right)$$

$$\dot{\theta}_B(t) = - \frac{T_B}{I} t_B + \frac{T_B}{I} t$$

Noting that

$$\dot{\theta}_A(t = 2t_A) = \dot{\theta}_B(t = 0)$$

and

$$\theta_A(t = 2t_A) = \theta_B(t = 0)$$

by simple substitution one obtains

$$\frac{T_A}{I} t_A = - \frac{T_B}{I} t_B$$

and

$$A + \frac{1}{2} \frac{TA}{I} t_A^2 = B + \frac{1}{2} \frac{TB}{I} t_B^2$$

These equations can be readily solved for t_A and t_B, first by equating t_A in terms of t_B

$$t_A = - \frac{T_B}{T_A} t_B$$

and then substituting the above relationship and solving for t_B

$$t_B = \sqrt{\frac{T_A \ (B - A) \ 2I}{T_B \ (T_B - T_A)}}$$

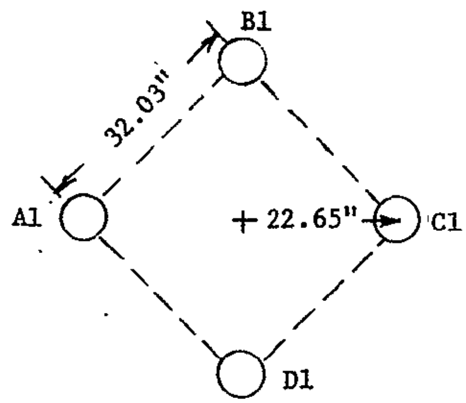
hence

$$t_A = -\frac{T_B}{T_A} \sqrt{\frac{T_A(B - A)\ 2I}{T_B(T_B - T_A)}}$$

The positive moment action time would be $2t_A$ while the opposing moment would be applied for $2t_B$. The complete cycle would last

$$t\ \text{cycle} = 2t_A + 2t_B$$

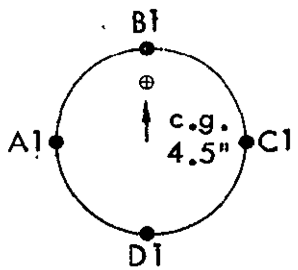
Representative vehicle characteristics for the ascent/descent mission are shown in Table 3. Using these average moments of inertia sample pitch and yaw control cycles are shown in Figure 8 for the ascent leg of the mission. The values are shown with the low level RCS thrusters always off and also for the option with them operating continuously (3 thrusters providing 40 in-lb moment per axis). In both instances it is possible to control the vehicle. The condition for which control authority by pulsing the orbit transfer engines only would not be feasible is when the lateral c.g. location falls on or outside the square formed by the engines (sketch below)

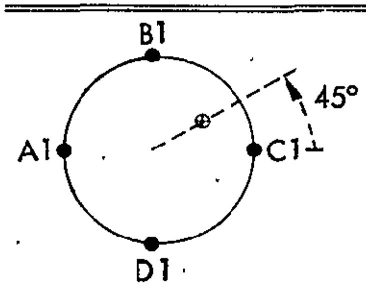


The control logic for the pulsating main motors will have to be quite complex since the cycle times for pitch and yaw control will probably be different.

Table 3. Average Spacecraft Characteristics (Ascent/Descent Mission)

		Ascent	On-Orbit	Descent
Weight			4280	
Ixx	slug-ft^2	1700	1700	1700
Iyy	slug-ft^2	4660	4300	3840
Izz	slug-ft^2	4320	4000	3465
Ixy	slug-ft^2	213	195	175





		POSITIVE MOMENT (ALL THRUSTERS ON)	(NEGATIVE MOMENT)		
		$2t_A$ (SEC)	THRUSTERS OFF	$2t_B$ (SEC)	t_{CYCLE} (SEC)
WITHOUT RCS	IN PITCH	16.2	A1, C1 & D1	16.0	32.2
WITH RCS	Iy = 4660	26.0	A1, C1 & D1	10.0	36.0
WITHOUT RCS	IN PITCH	20.4	A1 & D1	15.8	36.2
WITH RCS	Iy = 4660	47.2		9.2	56.4
WITHOUT RCS	IN YAW	19.6		15.2	34.8
WITH RCS	Iz = 4320	45.4		8.8	54.2

Figure 8. Control Authority, Pulsing Main Thrusters

The burn time would be extended for both modes of control. The additional propellant required to keep the RCS low level thrusters on during the entire ascent phase would be approximately 26 pounds. If non-nominal thrust effects were included in these calculations, the RCS propellant requirement would be increased by approximately two percent to 26.5 lbs. In either case the RCS propellant requirement thus calculated would be extremely conservative since control authority (pitch and yaw) could be maintained without the use of the RCS thrusters.

Similar analysis can be performed for roll control where the low level RCS thrusters must be used. An analysis showed that if all the orbit transfer thrusters are deflected 0.5 degrees so that all of them contribute to an unfavorable roll moment, the low level thrusters would be on for approximately 36% of the time. Two thrusters employed to maintain control would consume approximately 3.5 lbs of propellant.

As before, the above assumtion yields a conservative propellant estimate since it would be unlikely that all of the orbit transfer thrusters would be deflected in a direction that would result in additive roll moments.

2.5 TRANSPORTATION COSTS

The approach selected by the shuttle project for the allocation of transportation costs to the payloads sharing a mission and its variation with altitude of delivery is a significant driver on the mission approach selected for the Landsat follow-on mission and the resulting configuration of the propulsion subsystem. The current formulae for computing the pro-rated share involve an assessment of the fraction of the shuttle performance capability (to the selected altitude and inclination) represented by the payload weight, and the fraction of the cargo bay length utilized by the payload (including the OMS kit length if required). The larger of these two fractions is converted to a cost factor parameter by the relationship shown in Figure 9. This cost factor is the fraction of the flight costs to be assessed against that payload.

The final configuration of the Flight Support System (FSS) which supports and deploys the MMS on shuttle supported missions has not been selected at this time, but the following calculations (Table 4) using even approximate numbers clearly show the advantage of choosing a parking

$$\text{LOAD FACTOR} = \frac{\text{PAYLOAD WEIGHT}}{\text{SHUTTLE CAPACITY}} \quad \text{OR} \quad \frac{\text{PAYLOAD LENGTH}}{.60}$$

WHICHEVER IS LARGER

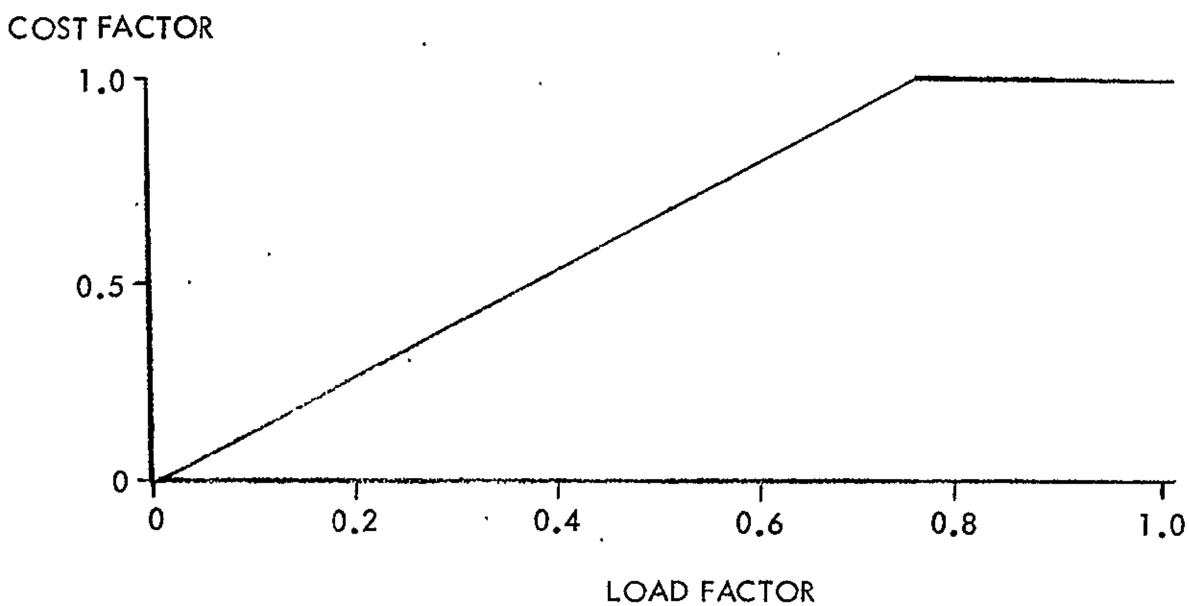


Figure 9. Shared Flight Charge

orbit lower than the desired operational altitude and providing the capability in the MMS to transfer to (and return from) the higher orbit.

Table 4. Transportation Costs

Configuration	Total Length[1] (ft)	Total Weight[2] (lb)	Delivery Altitude (nm)	Shuttle Capability (lb)	Cost Factors	
					Length	Weight
SPS-I (18.5")	28.5	5500	380.6	7000	0.63	1.00
SPS-II (60")	24.0	6700	150	30000	0.53	.30

Notes: 1 Includes 10' P/L, 10.5' or 14' MMS/FSS, and 8' OMS kit as applicable.

2 Includes 1900-lb P/L, 3600-lb or 4800-lb MMS/FSS/PM as appropriate.

For this approximate calculation the Landsat follow-on project would be assessed the whole cost of the shuttle flight if delivered to the operational altitude based on the weight cost factor and only 53% of the cost based on the length cost factor if delivered to 150 nm and the on-board propulsion used from there. It should be noted that the same fraction would apply up to the point (approximately 200 nm) where the first OMS kit is required. After that point, the fraction would increase to approximately 71%.

The use of the Viking Orbiter tank (discussed in a later section) for the shuttle missions appears to be the most cost effective choice of the available tanks because of the cost formulae. It may be desirable in the long run however, to procure and qualify a MMS-unique tank with a better shape factor to minimize the length. For example, if it is assumed that the shuttle flight cost to be prorated among the payloads were $18 million, then the reduction in transportation cost by shortening the propulsion module by one foot would be $400 thousand. Even a fraction of this cost avoidance could help amortize the added development and fabrication costs of a specially designed tank. This option has not been examined in this study.

2.6 SUMMARY OF PROPELLANT REQUIREMENTS

A circular 150 n.mi. parking orbit is considered to be representative of the class of stable low altitude Shuttle orbits. The propellant requirement summaries in this section represent this choice. To obtain ascent and descent propellant required for other parking orbits, Figure 5 (Section 2.2) should be used. Control propellant requirements can be adjusted by the same percentage as the transfer propellant requirements would change.

Table 5 summarizes the propellant budget for the ascent only mission. Included in this budget are 26 pounds for pitch and yaw attitude control. This assumes continuous use of six low level thrusters during the ascent phase. This amount could be eliminated if the center of mass is held within the envelope discussed in Section 2.4. The decision to include the 26 pounds in the propellant budget was predicated on possible center of mass excursion outside the specified limit.

The propellant required for orbit maintenance (injection error correction, orbit keeping, safe hold, etc.) was assumed to be the same as for the conventional launch vehicle mission (Section 1.0).

The preliminary assessment of possible injection errors indicated that for a 1 degree continuous attitude error, the velocity requirement would be an order of magnitude lower than for the Delta mission. This would be enough to account for increased propellant requirement for the ascent/descent mission's heavier vehicle (Table 3) for orbit maintenance (+73 lb) and safe hold operations (+ 0.4 lb).

Table 6 summarizes the propellant budget for the ascent and descent mission. The same 150 n.mi parking orbit was assumed for Shuttle-supported phases of the mission. This propellant budget, although conservative, does not include any allocation for contingency or propellant reserves.

Table 5. Propellant Requirement Summary - Ascent Only
Parking Orbit Altitude = 150 n.mi.

Ascent Propellant	428. lb
Pitch & Yaw Control During Ascent	26 lb
Roll Control During Ascent	3.2 lb
Orbit Maintenance	61.1 lb
Total	518.3

Table 6. Propellant Requirement Summary - Ascent & Descent
Parking Orbit Altitude = 150 n.mi.

Ascent Propellant	481 lb
Pitch & Yaw During Ascent	29 lb
Roll Control During Ascent	3.5 lb
Orbit Maintenance	61.1 lb
Descent Propellant	425 lb
Pitch & Yaw Control During Descent	25 lb
Roll Control During Descent	3 lb
Total	1027.6 lb

3.0 PROPULSION SYSTEM ANALYSIS

3.1 REQUIREMENTS

A propulsion system analysis of the Landsat Follow-on spacecraft was conducted to determine the optimum configuration capable of supporting both Thor-Delta and Shuttle launch operations. The scope of the analysis included the identification of candidate thrusters, tankage and related equipment/ components, propulsion system schematics and related trade-off, including Rough Order of Magnitude cost data.* In accordance with the GSFC work statement, Spacecraft Propulsion Subsystem I (SPS-I) refers to a module designed for use with the Thor-Delta 3910 launch vehicle; SPS-II designates a system intended for use with the Shuttle and SPS-IA describes essentially an SPS-I module but with additional tankage installed in the spacecraft structural tunnel. SPS-I configurations utilizing one, two, three, and four tankage elements are also to be considered.

A flight mechanics analysis was conducted to determine the propellant required to implement the missions defined in the GSFC work statement. The rationale and methods used to conduct the analysis are presented in Sections 1 and 2 of this report. The resulting propellant quantities determined are summarized in Table 7.

Table 7. Propellant Requirement Summary

Thor-Delta Launch Case	
Orbit injection error correction	12.9 lbs.
Coarse sun acquisition maneuver	1.6 lbs.
Orbit maintenance (3 years)	35.9 lbs.
Three additional coarse sun acquisition maneuvers	4.8 lbs.
Safe hold mode	5.9 lbs.
Total	61.1 lbs.

*Costing data and analysis is presented in a separate appendix.

Table 7. Propellant Requirement Summary (Cont)

Shuttle Launch Case - 150 N.Mi. Parking Orbit	
Ascent Propellant	481 lbs.
Pitch and Yaw During Ascent	29 lbs.
Roll Control During Ascent	3.5 lbs.
Orbit Maintenance	61.1 lbs.
Descent Propellant	425 lbs.
Pitch and Yaw Control During Descent	25 lbs.
Roll Control During Descent	3 lbs.
Total	1027.6 lbs.

A basic spacecraft weight of 3564 lbs. and an axial thruster specific impulse of 230 sec were used to compute the propellant weights presented in Table 7. "Basic Spacecraft" refers to the weight of the all up Landsat Follow-on spacecraft but does not include any allowance for either the SPS-I or SPS-II. Some additional study design criteria, based primarily on Ref. 4 are presented below:

Study Design Criteria

1. The primary stabilization and control forces for pitch, yaw and roll maneuvers will be provided by momentum wheels. Momentum wheel dumping will normally be accomplished magnetically. The 0.2 lbs. thrust hydrazine thrusters are back-up for the pitch, yaw and roll functions in the event of a wheel system failure and also for momentum wheel dumping.

2. Translation/orbit adjust thrust will be provided by four 5 lb. thrust engines for SPS-I and/or two 150 lb. thrusters for SPS-II. Stabilization forces required during operation of the translation thrusters will be generated by the 0.2 lb. thrusters. A gimbal system shall also be studied for use with the 150 lb. thrusters. Landsat orbital altitude will be 380 n.mi. (705 km); inclination 98.2°.

3. The propulsion modules will be required to satisfy the ground rule that, "no single failure shall prevent Shuttle retrieval".

4. Hydrazine propulsion system components under development by the NASA Low Cost Systems Office (LCSO) shall be examined for application to the MMS.

5. A blowdown mode propellant expulsion system shall be used. The desired blowdown ratio (initial tank pressure/final tank pressure) is 3 to 1.

6. The temperature of the Space Propulsion Subsystems will be controlled by heaters and thermostats to 68 $\pm$ 18°F (20 + 10°C).

7. The propellant to be used shall comply to MIL Spec MIL-P-26536C, Amendment 1, monopropellant grade hydrazine.

8. Lifetime between reservicing flights: 3 years minimum.

9. The propulsion module shall be designed to be replaced on orbit by a Flight Support System (FSS) mounted in the Shuttle cargo bay.

3.2 STUDY PLAN

The propulsion systems analysis was conducted according to the following generalized task statements:

1. Identify all subsystems and components, i.e., thrusters, tankage, latch values, filters, etc., required to synthesize the SPS I, II, and IA propulsion modules.

2. Formally contact the suppliers of the Item 1 propulsion system elements and request supporting technical data and Rough Order of Magnitude costing information.

3. Formulate representative propulsion systems capable of meeting the propellant requirements indicated in Table 4. Consider a sufficient number of options to assure that an optimum design will result.

4. Select, with GSFC concurrence, the optimum configuration for SPS-I, II, and IA. These configurations will form the basis for the design study and drawings to be prepared during the design phase of the study.

4.0 THRUSTER OPERATIONS AND ISSUES

4.1 GENERAL DISCUSSION

The primary purpose of the thruster options and issues task was to assemble performance, configuration and ROM cost data on candidate hydrazine thrusters. Beginning-of-life thrust level requirements have been specified as 0.2 lb; 5.0 lbs, and 150 lbs., with each thruster designed to operate over a 3:1 blowdown range. Qualified engines in all required thrust levels are available from several sources. Representative examples are presented in Table 8.

Table 8. Potential Thruster Capability

Company	0.20-lb Thruster	Qualified	5.0-lb Thruster	Qualified	150-lb Thruster	Qualified
Bell	Yes	No *	Yes	No	No	No
Hamilton Standard	Yes	Yes	Yes	Yes	Yes	Yes
Marquardt	Yes	No	Yes	No	Yes	Yes
Rocket Research	Yes	Yes	Yes	Yes	Yes	Yes
TRW	Yes	Yes	Yes	Yes	No	No
Hughes	Yes	No	Yes	Yes	NOT REQUESTED	

*Bell has indicated that the thruster is currently undergoing qualification testing.

The companies indicated above have been contacted and have provided performance, configuration and cost data. Each company was requested to present information on the use of single and dual seat propellant valves. Dual seat/dual coil valves, such as used on the GPS 0.2 lb. thrusters, have an effect in the overall propulsion system in that the number of latch valves, for example, will vary as a function of the thruster propellant inlet valve selected. The reliability and cost is also directly impacted by the type of propellant inlet valve used.

The baseline SPS designs employ four identical Rocket Engine Modules (REM) each consisting of three 0.20 lb. thrust engines and one 5.0 lb. thruster. The companies indicated in Table 8 were asked to provide data on both complete

REM's and individual thrusters. In order to provide a basis for a uniform response, the Space Division provided each company with the configuration information presented in Figure 10. Figure 10 should not be construed as being the selected design but as a representative configuration. In addition to the thruster, the REM includes all the electrical leads associated with valve and catalyst heaters, command/control wiring and instrumentation leads (pressure transducer not required). All electrical leads will terminate in a standard aerospace electrical connector mounted on the inside of the REM. All propulsion components, when in a non-operating mode, will be maintained at 68°F ± 18°. The REM includes the line heaters and thermal coatings necessary to meet this requirement. Fill/drain valves, lines, filters, tankage, etc., are not part of the REM. All REM's are identical and are interchangeable. If a REM component malfunctions prior to flight, the entire module will be replaced. The 150 lb. thrusters are not part of the REM assembly.

In order to provide a systematic basis for the identification of the respective REM's and thrusters, the method indicated in Figure 11 is recommended. One key feature of the suggested nomenclature is that all thrusters providing the same function have identical descriptors. For example, all thrusters capable of providing nose-up pitch forces are designated 2; all translation/orbit adjust thrusters are identified as 1.

4.2 THRUSTER CONFIGURATION AND PERFORMANCE

Configuration data on several representative thrusters is presented in Figures 12, 13, 14, 15, 16, and 17. It should be emphasized that the thrusters shown have not been selected for MMS but are presented to indicate the sizes and shapes to be used for preliminary layouts. The Rocket Research thrusters shown in Figures 13 and 14 are of special interest in that they depict the GPS thruster with a dual seat/dual coil valve and the LCSO Standardized Thruster, respectively. Additional thruster information is provided in Table 9. Thruster operating duration is a function of the SPS module selected. For the Shuttle-supported mission options wherein the 5 lb. thrusters are used for orbital transfer, a total of 481 lbs. of propellant will be passed through the four thrusters on the ascent phase and 425 on the descent mode. For the case in which the 150 lb. thrusters will provide the orbital transfer ΔV, all the

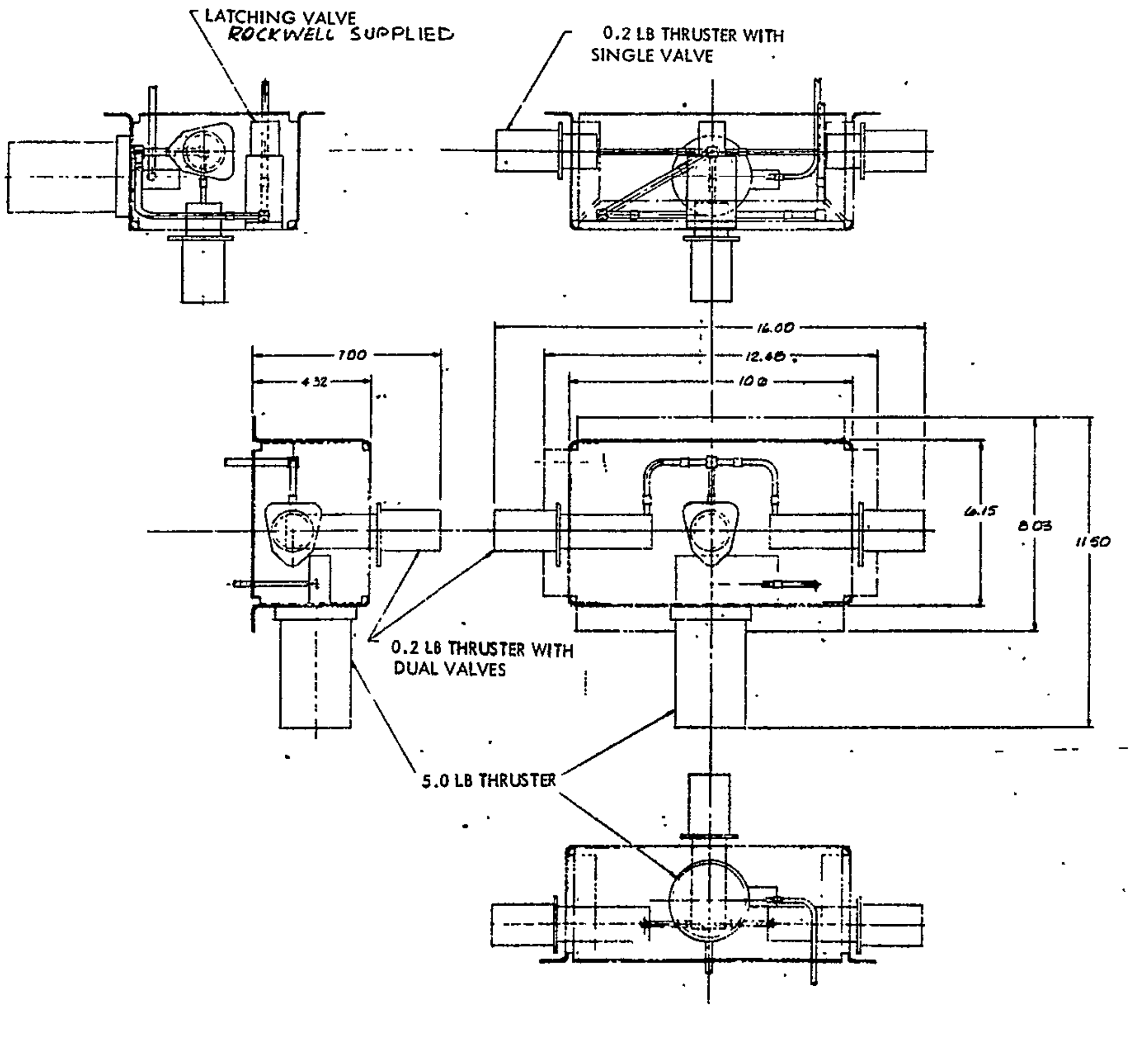


Figure 10. Concept--Module Assembly Rocket Engine, SPS-I

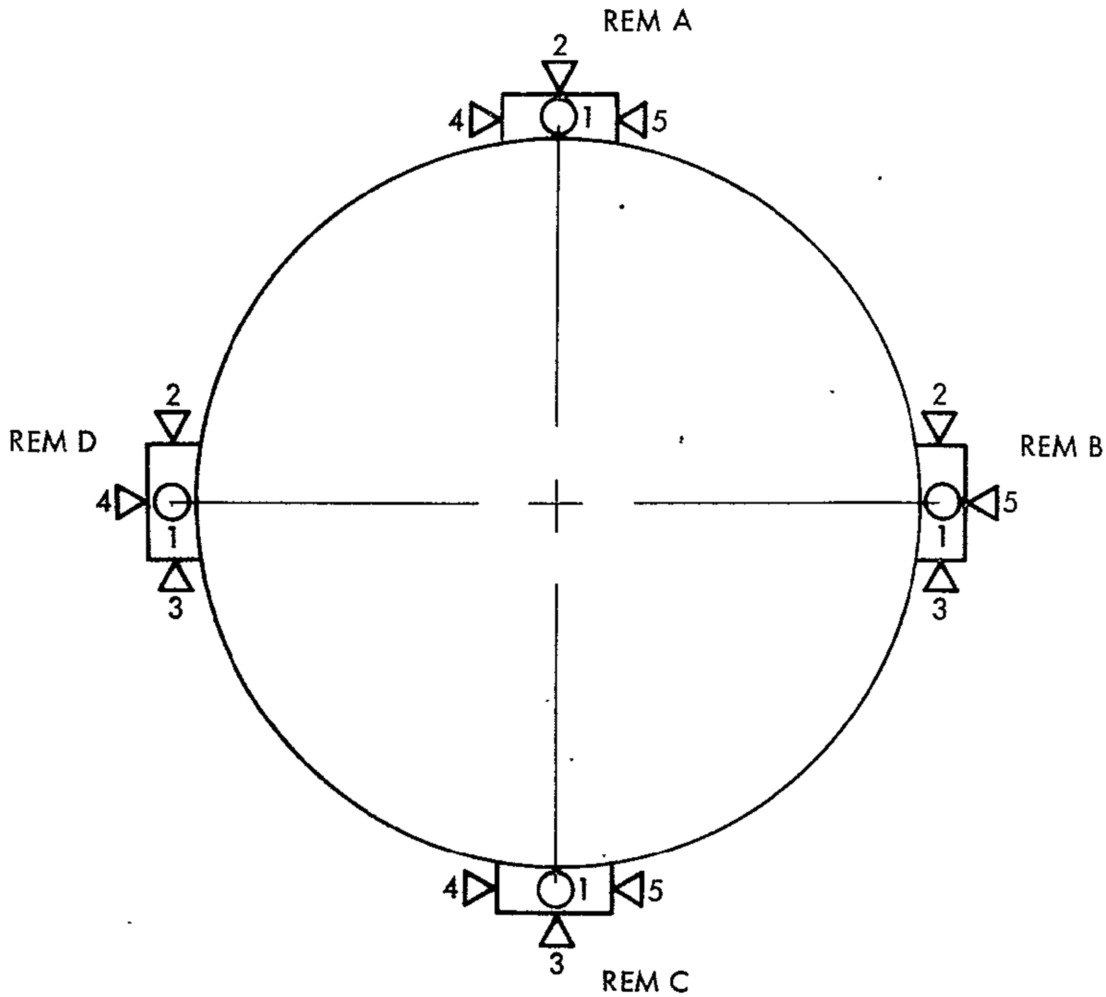
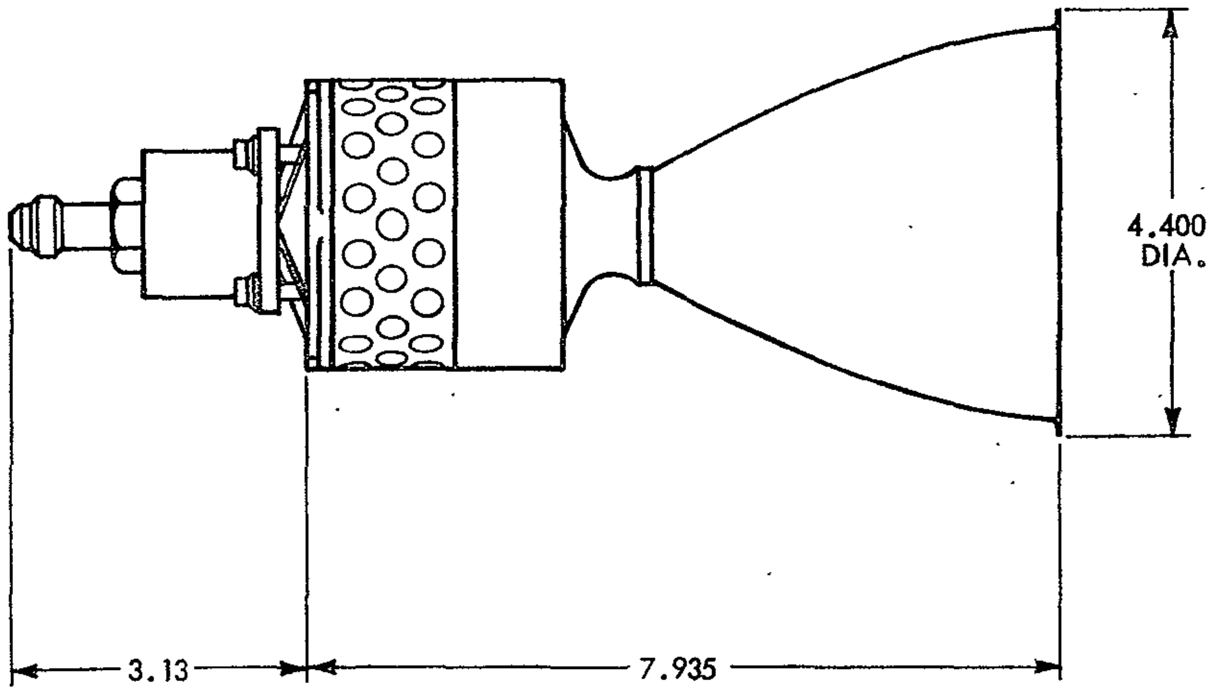


Figure 11. REM/Thruster Identification

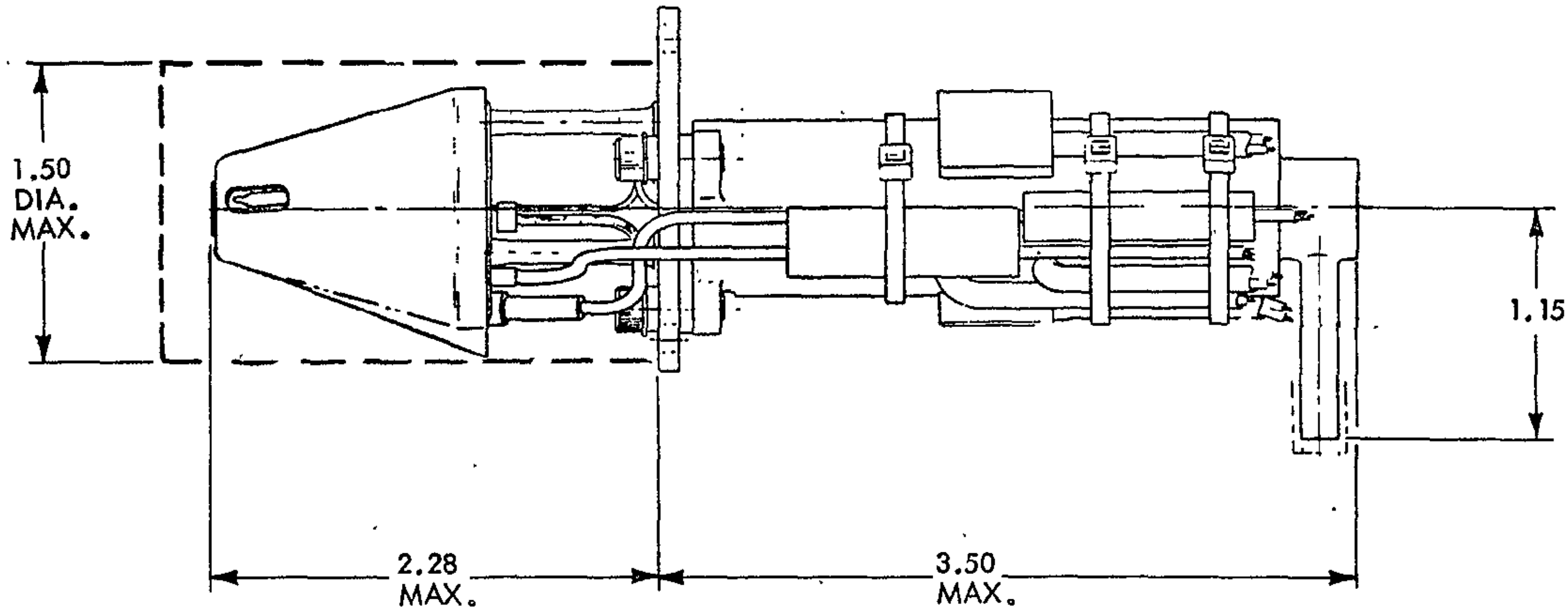
propellant will pass through them. Because the shuttle launched configuration is required to be capable of being reused, all orbit adjust thrusters should be designed to accommodate at least twice the propellant quantities identified above. Approximately 60% of the remaining 125 lbs. of propellant carried for the Shuttle launch will also be passed through the axial thrusters. The remainder is assumed equally divided among the 0.20 lbs. Thrusters meeting the shuttle launch case duration criteria will also satisfy the Thor-Delta launch.

None of the 150 lb. thruster configurations show gimbal actuator equipment. Gimballing the 150 lb. thrusters has received consideration for



WEIGHT = 3.0 LB_m MAX.

Figure 12. Marquardt 155 Lb_f Thruster



WEIGHT = 1.1 LB_m MAX.

Figure 13. Rocket Research Corp. 0.2 Lb_f GPS Thruster

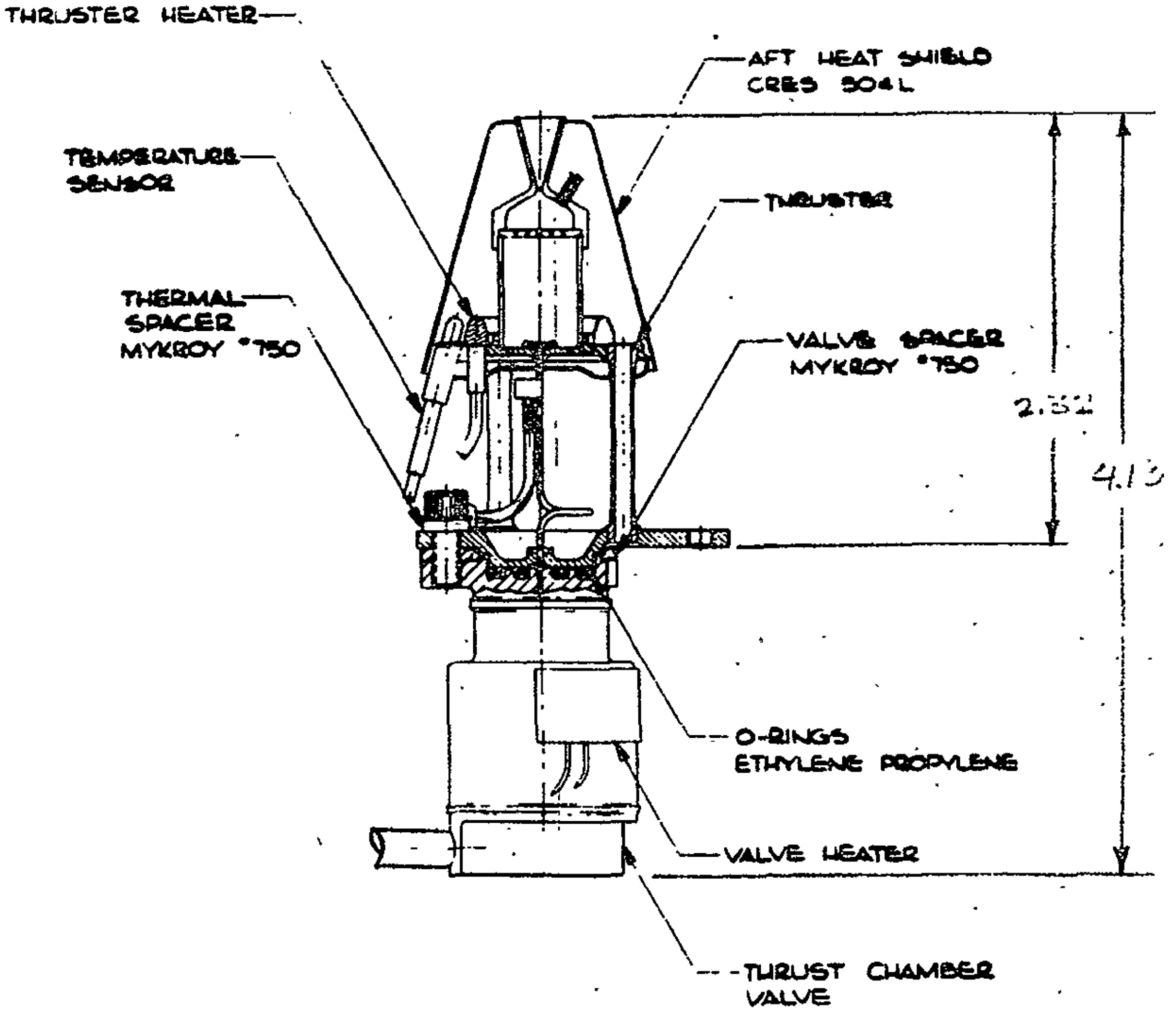


Figure 14. Rocket Research LCSO 0.20 Lb_f Thruster

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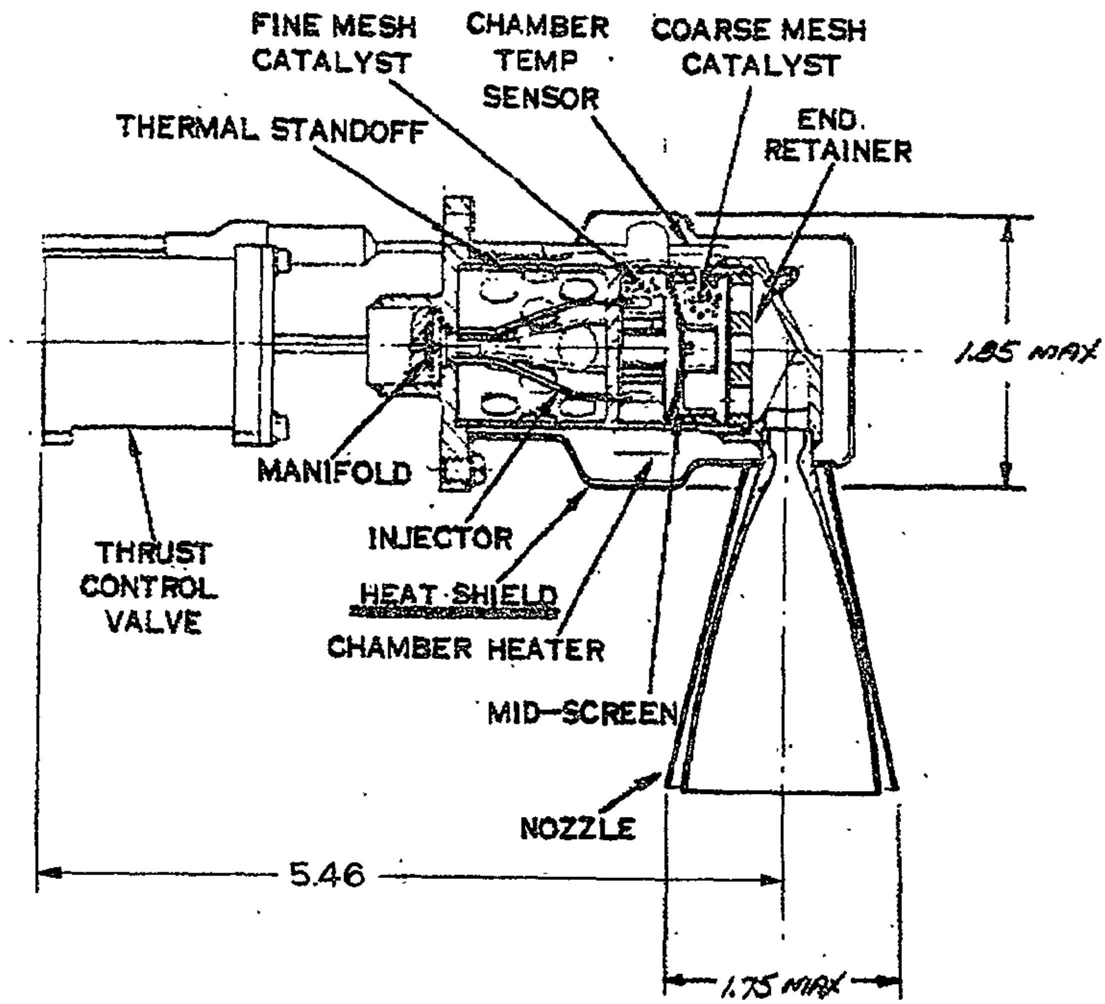
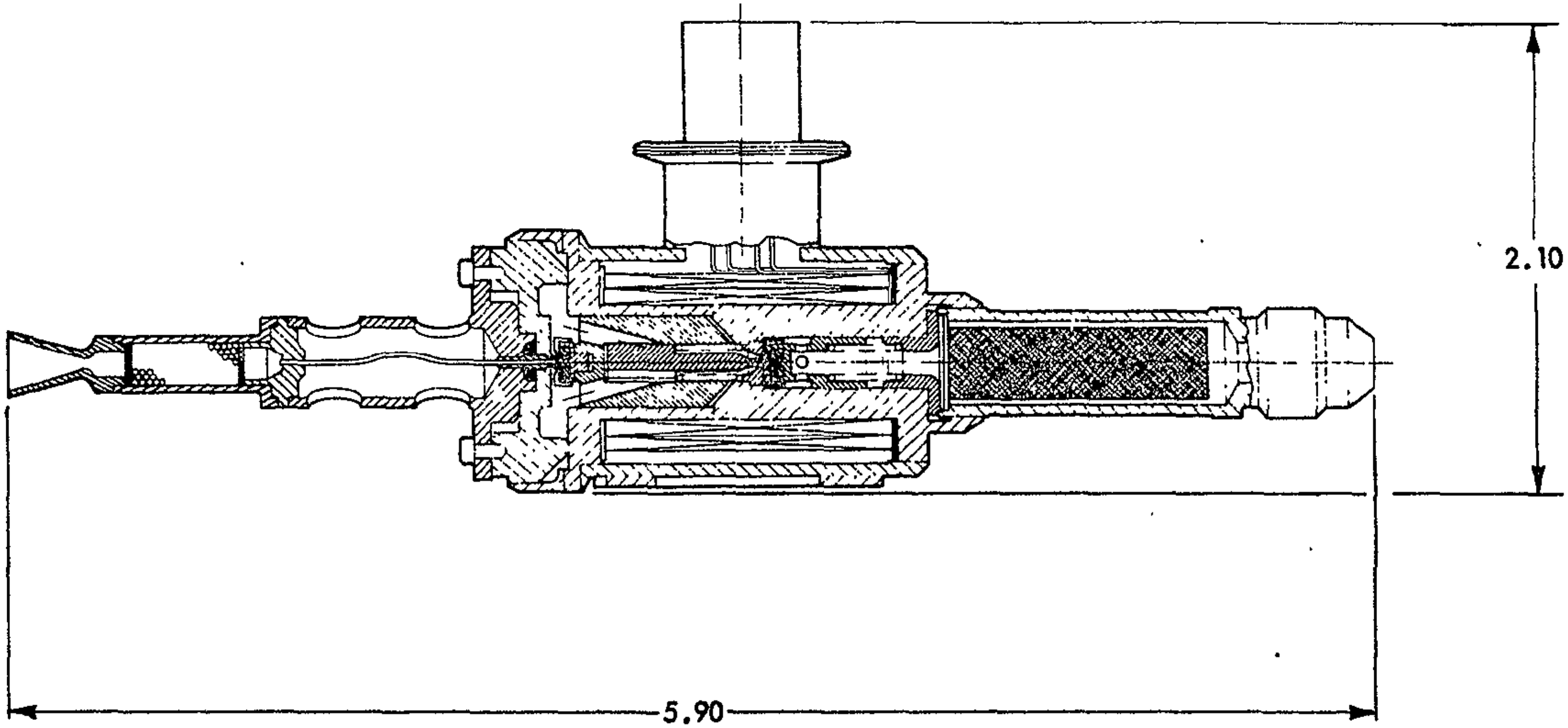


Figure 15. Hamilton Standard IUE 5.0 Lb_f Thruster



NOTE: HEAT SHIELD NOT SHOWN

WEIGHT = 0.7 LB_m MAX.

Figure 16. TRW 0.1 Lb_f FLTSATCOM Thruster

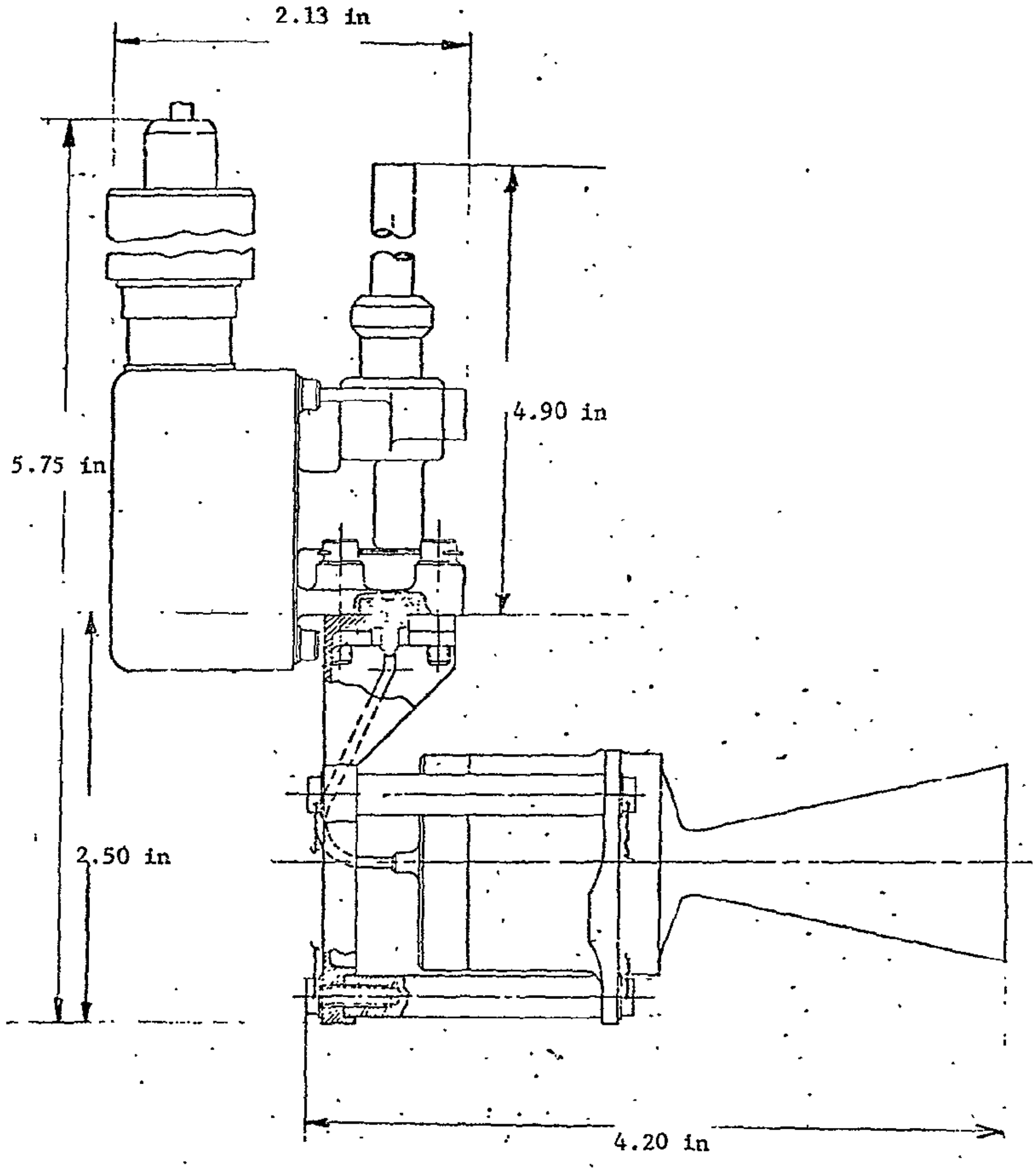


Figure 17. Hughes 5.0 Lb Thrust Engine Assembly

Table 9. Thruster Requirements

Parameter	0.2 LBF Thruster	5.0 LBF Thruster	150 LBF Thruster	Remarks
Operating Pressure	80 to 450	80 to 450	80 to 450	Thrusters to be capable of operating in blowdown mode.
Internal Leakage	1 scc GN_2/Hr @ P_{max}	5 scc GN_2/Hr	3 scc GN_2/Hr @ P_{max}	Mission life up to 3 years
External Leakage	$< 10^{-6}$ scc/sec He @ P_{max}	10^{-5} scc He/sec	10^{-5} scc He/sec	
Power	5 watts @ 32 vdc	30 watts @ 32 vdc	59 watts @ 32 vdc	
Thrust Control Valve	Single or redundant seat	Single seat or redundant	Single seat	
Operating Voltage	24 to 32 vdc	24 to 32 vdc	26 to 32 vdc	
Catalyst Bed Temp.	400 F	TBDV	TBDV	
On-Time/Duty Cycle	8 ms/ 1%, 5% & 15%	125 ms/ 12.5% & 14%	Steady State	

TBDV - To be determined by vendor.

NOTE

Shuttle missions will require a propellant through-put of 481 lbs on the ascent phase and 425 lbs during descent. In the event of the loss of a 5 lber, the opposite thruster will also be shut down for control purposes thereby requiring that all the remaining ΔV propellant to be passed by the remaining 5 lbers. A total of 125 lbs of propellant is assumed to be equally divided among the 0.20 lbers.

application to the SPS-II design. A brief investigation has indicated that space-rated actuators for use with this size engines are relatively uncommon items. The actuator used on the Mariner '71 and VO '75 300 lb. thrust bi-propellant engine, however, is a viable option. The actuator for Mariner '71 was designed and built by JPL in-house; the VO '75 equipment was fabricated by General Electric. JPL has provided drawings and performance data on the actuator and GE has supplied ROM costing information.

Jet vanes were examined briefly as an alternate to a gimbal system. The early JPL Mariner Spacecraft utilized a 50 lb. thrust hydrazine engine equipped with jet vanes. JPL indicated that heat soak back through the vanes was a significant problem. It was also indicated that the Mariner jet vane immersion in the exhaust stream was very short when compared with the MMS mission. The long burn time of the MMS would impose an extremely severe operating environment on the vanes. A potential solution may be found in a technique which immerses the vanes into the exhaust stream only when thrust vector control is required. The gimbal system, however, appears to be a less complex system and is therefore potentially more cost effective.

As previously indicated, a blowdown ratio of 3:1 has been selected. The performance variation of representative 0.20 lb. and 5.0 lb. thrust engines is shown in Figures 18 and 19, respectively. The Space Division has tested 0.20 lb. thrusters obtained from three different engine suppliers and has substantiated the trend shown in Figure 19. The performance of the 5.0 lb. thruster has also been duplicated by the Space Division but only with engines provided by a single supplier. It should be observed that a 3:1 variation in inlet pressure does not in all cases produce a 3:1 change in thrust. This phenomena is caused by the fact that the pressure drop across the rest of the system is not always linear with inlet pressure. Corresponding data for specific impulse was also obtained and a representative example is shown in Figure 7.

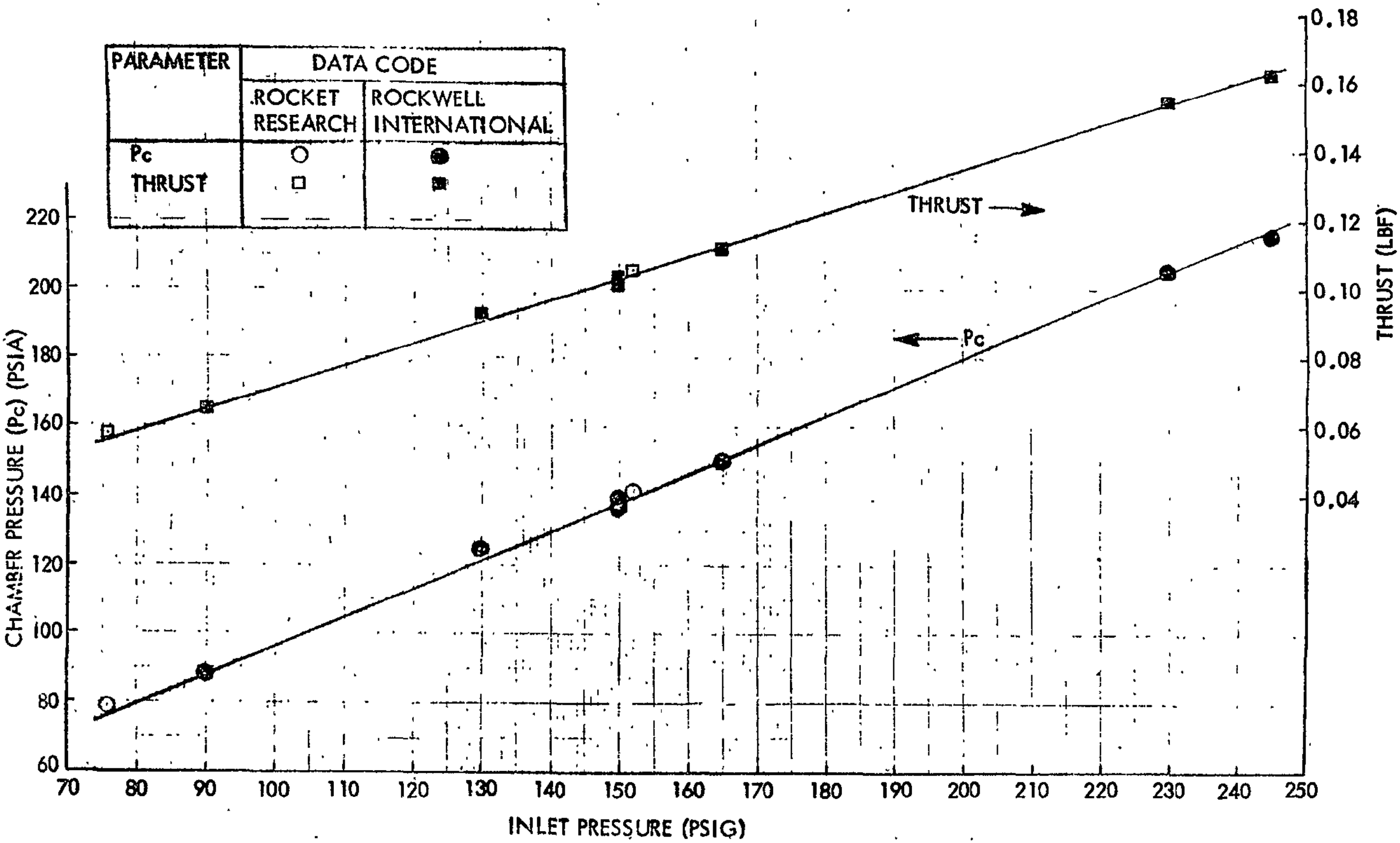


Figure 18. Rocket Research Corporation 0.2 Lb_f Thruster Performance Data

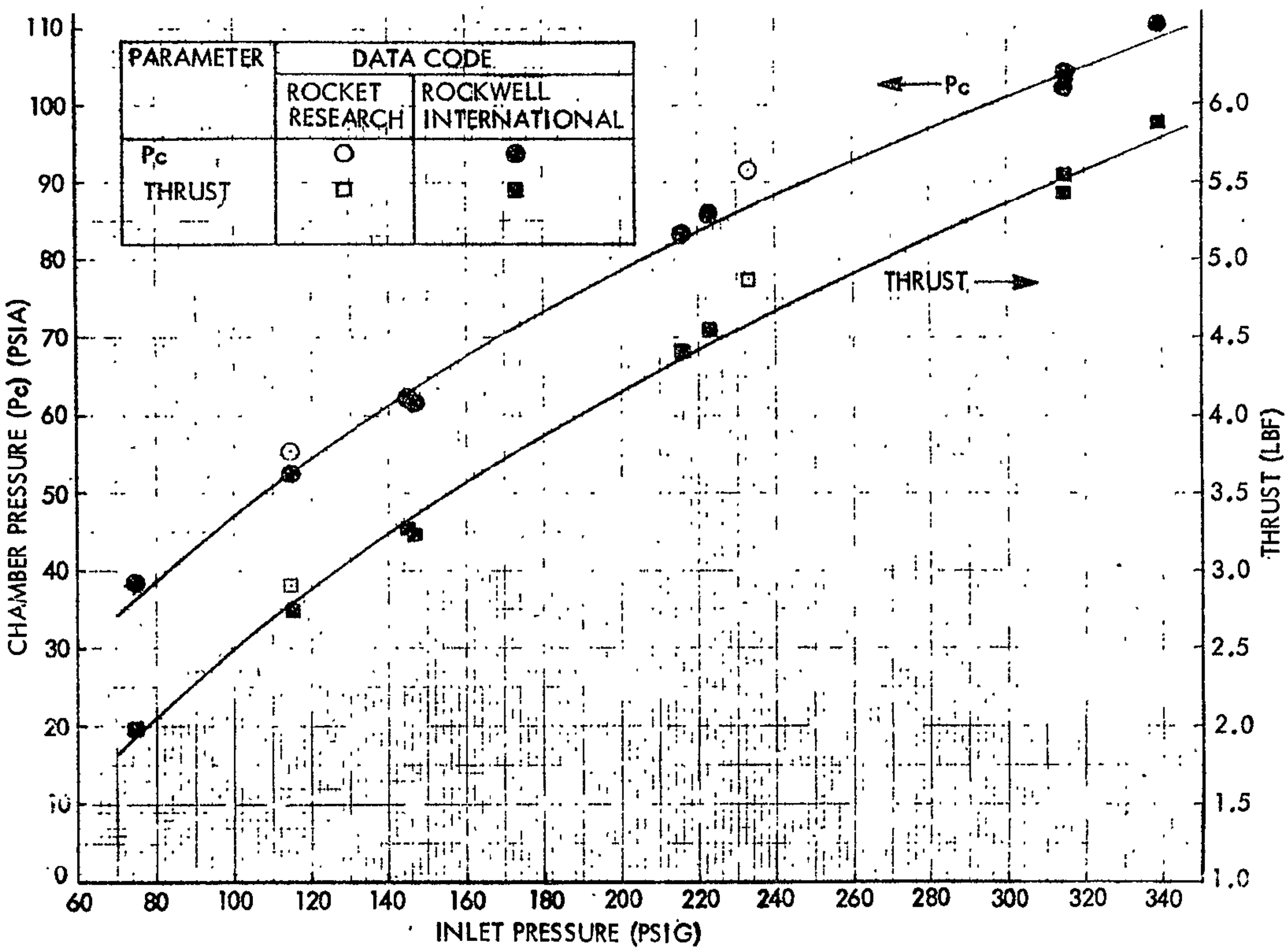


Figure 19. Rocket Research Corporation 5.0 Lb_f Thruster Performance Data

5.0 TANKAGE OPTIONS AND ISSUES

5.1 INTRODUCTION

The primary activity conducted during this phase of the study consisted of the acquisition of geometric, performance and cost data of hydrazine tankage systems capable of meeting the propellant requirements indicated in Table 7. The three axis stabilization mode requirement of the Landsat Follow-on spacecraft requires the use of tankage systems capable of maintaining a continuous flow of propellant to the thrusters in a sustained zero g environment. Both positive expulsion and capillary propellant management techniques meet this requirement. Of the two methods, the positive expulsion system is used in essentially all three axis stabilized spacecraft currently operational and there are a significant number of tankage sets available with hydrazine propellant capacities ranging from 10 to 275 lbs. For that reason, the bulk of activity expended on tankage selection was concentrated on positive expulsion systems employing an elastomeric diaphragm. A limited discussion of other candidate systems is also included in Section 5.3 of the report.

5.2 PRESSURE SYSTEMS, INC., TANKAGE SYSTEMS

Pressure Systems, Inc. (PSI) is the principal source of the candidate tankage sets. A summary of flight qualified systems produced by PSI is presented in Table 10 and some representative configurations are shown in Figure 20. The program names identified in Column 1, however, should be regarded as generic descriptors. PSI, for example, lists eight variations of the 16.5 inch diameter tank. The differences are centered mainly on mounting techniques and tank wall thickness. Similarly, there are three qualified versions of the 22.14" tankage. Because of the large number of options, it was not practical to request cost data on all potential candidates. PSI has indicated that the difference in cost between the ATS and BSE tankage attributable to the mounting technique used is less than 5% with the BSE design having the lower cost. This figure was indicated as being representative of the costs incurred to modify the mounting configuration of tankage in the ATS/HEAO family. This general statement may not necessarily apply to tank sets such as Marots/GPS which use

Table 10. Pressure Systems, Inc., Candidate Tank Data

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Name	P/N	ID (IN)	Opr. Press (PSIG)	Tank Wt. (LB)	Burst To Oper. Press	Int. Volume (Cu. In.)	Qual. N_2H_4 Volume (Cu. In.)	Qual. Vol To Int Vol %	BD Ratio	Fuel Wt. (LB) **	Fuel Wt. @ 3:1 BD Ratio	Supplier Max. Vol. (Cu. In.)	% QV To IV	Max Fuel Wt. (LB)	Max BD Ratio
IUE	80222-1	9.41	400	2.9	2.0	415	290	69.9	3.3	10.54	10.1	332	80	12.1	5.0
GPS	80216-1	12.88	396	5.2	2.0	1080	918	85.0	6.7*	33.4	26.2	918	85	33.4	6.7
Marots	80225-1	15.38	319	8.2	2.1	1820	1385	76.1	4.2	50.3	44.1	1547	85	56.2	6.7
ATS	80177-1	16.5	400	10.2	2.5	2300	1662	72.3	3.6	60.0	55.7	1955	85	71.0	6.7
HEAO	80226-1	22.14	350	15.0	2.0	5555	3705	66.7	3.0	134.6	134.6	4722	85	171.6	6.7
Shuttle	80228-1	28.0	355	43.0	3.0	11350	8017	71.0	3.4	291.4	275.0	10215	90	371.2	10.0
*** Viking '75	80183-1	36ID X55.56	330	95.0	2.6	43811	39440	90.0	10.0	1433.4	1061.5	39440	90	1433.4	10.0

NOTE: Shuttle RCS Burst/Relief = 1.50.

* Not including supplemental ullage gas tankage.

** Based on a hydrazine density of 0.0363 lbs/cu. in. @ 68° F.

*** Data based on tank without propellant management device.

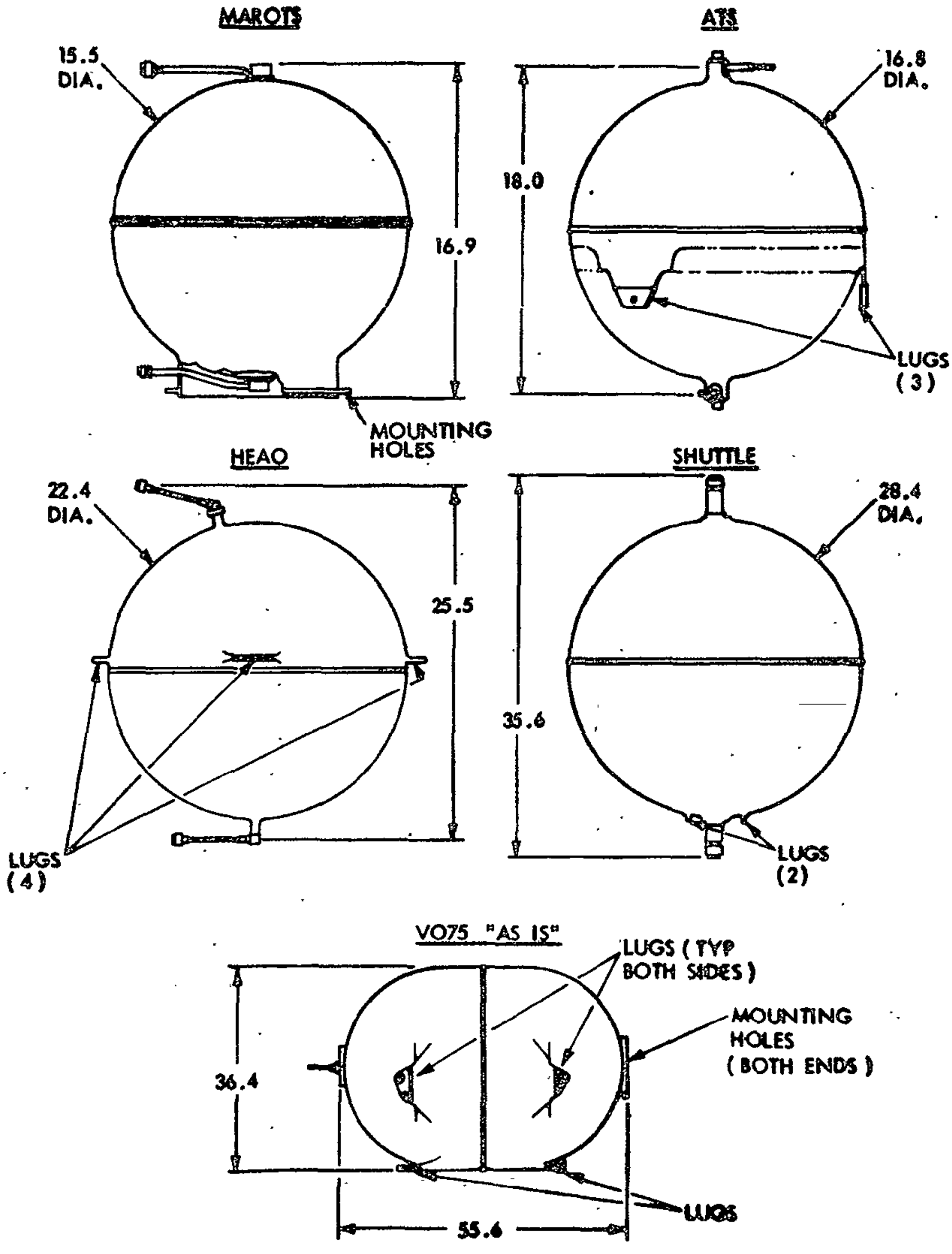


Figure 20. PSI Tankage Configurations

a unique crown mounting design. These tanks, however, also offer a degree of flexibility in that a tank set could be assembled from two "tops" with mounting attachments provided at the equator.

Several other items of importance found in Table 10 require discussion. PSI has indicated the data presented in Column 7, internal volume, contains an allowance for the diaphragm, i.e., the volume indicated is totally available for pressurant/propellant storage. The supplier maximum volume information, Column 13, applies to a condition wherein the diaphragm is "snug but not stretched." Note that the blowdown ratio associated with this condition, however, is significantly greater than the value of 3:1 indicated as desired. It should also be observed that, with one exception, all PSI positive expulsion tankage is only qualified for installation in the spacecraft with the diaphragm perpendicular to the launch vehicle thrust axis. The exception is the BSE tankage which is installed with the diaphragm parallel to the launcher centerline. Conversations with PSI personnel have been held to assess the impact of other mounting configurations. Some design studies have indicated, for example, the desirability of mounting the Marots tank upside down, i.e., with the propellant over the ullage gas during launch. PSI is actively considering such an arrangement for a different application and can foresee no reasons which such a mounting orientation would not be feasible. Using the Marots tank in this mode would require a certain amount of engineering effort, fixture modification, and requalification. It should also be observed that all diaphragms provided by PSI will be fabricated from AF-E-332.

The V075 tankage appears well suited for application to the SPS-II configuration. The tank does not incorporate a positive expulsion device but uses a capillary system to provide propellant management. It should be pointed out that PSI is responsible only for the manufacture of the tank shell; the propellant management device (PMD) is fabricated by Fansteel. The Jet Propulsion Laboratory (JPL) was responsible for the installation by the PMD and subsequent qualification of the completed system. Use of the complete V075 tankage system would require that another organization acquire the tank/PMD as components, complete the assembly and qualify the unit. JPL has indicated that the necessary tooling and fixtures would be made available and technical consultation provided but that JPL could not be considered as a supplier of the complete, qualified system.

PSI has indicated that if a simplified mounting system could be used in MMS, a 20% savings in unit cost could be realized. Non-recurring costs incurred to implement the required changes have been evaluated and are included in the 20% savings factor.

Space Division personnel have contacted Fansteel to obtain ROM cost data on the V075 PMD. The information received indicates that the PMD cost is 28% of the figure provided by PSI on 24 June for the "as is" V075 tank. PSI was asked to provide cost data covering the installation of an SD supplied PMD in the V075 tank shell. The response received indicated the cost incurred would be 0.7% of the unit cost of an "as is" V075 tank. Additional discussion of the cost factors is contained in a separate appendix. It should be reiterated that the above cost data are based on the availability of jigs, tooling, and technical documents from JPL at no cost to this program.

A significant factor evolves from Table 10. Utilizing developed tanks, the only other viable tankage system applicable to SPS-II is a cluster of four Shuttle tanks. The recurring costs for such an arrangement is approximately twice those for the modified V075 tankage. As discussed in Section 2.5, a new tank development may be cost effective when launch costs are considered.

Based on the cost and weight factors, the modified V075 appears to be an attractive selection. An analysis will be required to determine whether the capillary system is capable of supplying propellant to the axial and attitude control thruster during all phases of the mission. In addition, some concern has been expressed over the differences in the fluid dynamics properties, particularly contact angle, between MMH and hydrazine. For propellant orientation, the V075 tank uses a surface tension device consisting of an open channel 12 element central vane assembly as shown on Figure 21. The propellant management device was designed for use with MMH and N_2O_4 while the MMS will use hydrazine.

Table 11 presents a comparison of the low and high values found for the applicable properties of both MMH and hydrazine. The difference between the low and high values is due primarily to the test data scatter.

Table 12 presents a comparison of the capillary performance characteristics of the two different propellants assuming the same retention system design.

Table 11. Comparative Fluid Properties

PROPERTY	MMH CH_3NHNH_2 (MIL-P-27404A)		HYDRAZINE N_2H_4 (MIL-P-26536C)	
	LOW	HIGH	LOW	HIGH
SURFACE TENSION, σ, DYNES/CM	33.8	35.2	38	66.6
DENSITY, ρ, LB/GAL	7.334		8.415	
CONTACT ANGLE, θ, DEGREES	1	7	4	55
MEASURED BUBBLE PRESS, P_B, PSI (325 x 2300 MESH SCREEN AT 70°F)	1.4	1.5	.9	2.5
DYNAMIC VISCOSITY, μ, CENTIPOISE (AT 67°F)	.85		.97	

Table 12. Comparison of MMH & Hydrazine Capillary Performance Characteristics

	LOW	HIGH
1) RATIO (HYDRAZINE/MMH) OF MEASURED BUBBLE PRESSURES		
$P_{B_H}/P_{B_M}\Big\rvert_{\text{MEASURED}} =$	.60	1.79
2) RATIO (H/M) OF COMPUTED BUBBLE PRESSURES		
$P_B : \sigma \cos\theta ; \quad P_{B_H}/P_{B_M}\Big\rvert_{\text{COMPUTED}} =$	.62	1.98
3) CAPILLARY RETENTION STABILITY		
RATIO (H/M) OF MAXIMUM ALLOWABLE ADVERSE ACCLERATION, g		
BOND NUMBER $= \rho g D^2/\sigma ; \quad g_H/g_M = \rho_M \sigma_H / \sigma_M \rho_H =$	.94	1.72
4) CAPILLARY FLOW STABILITY		
RATIO (H/M) OF MAXIMUM CAPILLARY FLOW RATE, $\dot{\omega}_{MAX}$		
$\Delta P_{FLOW} = \frac{fL}{D}\,\frac{\dot{\omega}^2_{MAX}}{2\rho g} \leq P_B ; \quad \frac{\dot{\omega}_H}{\dot{\omega}_M}\Big\rvert_{MAX} = \sqrt{\frac{\rho_H P_{B_H}}{\rho_M P_{B_M}}} =$	.83	1.43

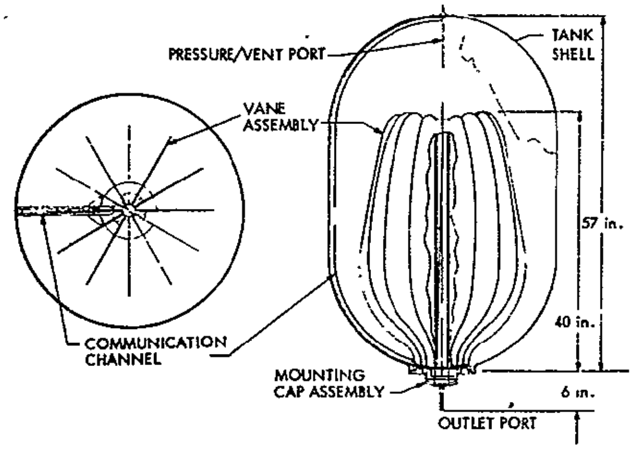


Figure 21. V075 Propellant Management Device

Using the extreme worst case data, the bubble pressure for hydrazine is still 60% of that for MMH showing that the larger contact angle for hydrazine is substantially off-set by hydrazine's higher surface tension.

A comparison of Bond numbers as shown in item 3 of Table 12 reveals that under the worst case computed conditions, the adverse acceleration required to lose control of the propellant location is only reduced 6% by the substitution of hydrazine for MMH in a given capillary system. Balancing the bubble pressure with the fluid flow pressure drop as shown in item 4 of Table 12 indicates that the capillary flow rate capability for hydrazine in a given surface tension may be as low as 83% of that for MMH. This is not considered a severe penalty since capillary flow channels are usually designed for flow safety factors of 1.5 to 4.0.

The major problem regarding the substitution of hydrazine in place of MMH is the high contact angle of hydrazine. Although stability analysis shows that the high surface tension of hydrazine adequately compensates for its higher contact angle, there is some concern that this high contact angle may retard, or in some cases prevent, initial liquid wetting of a dry capillary surface. Scale model testing with hydrazine would be required to resolve this issue.

Table 13 presents a comparison of available data for the acceleration and flow rate environments affecting capillary system performance. Additional evaluation of the V075 system's capability and the MMS vehicle's imposed environment will be required to verify acceptable use of this tank. These

Table 13. Comparison of MMS and Viking 75 Capillary System Environments.

ENVIRONMENT \ VEHICLE	VIKING	MMS
ADVERSE ACCELERATION, g/g_o		
ATTITUDE MANEUVERING, MAX LATERAL THRUST		13×10^{-5}
ATTITUDE MANEUVERING, ROTATIONAL	20×10^{-5}	
DURING VENTING	1×10^{-5}	N.A.
SHUTTLE DEPLOYMENT		326×10^{-5}
CAPILLARY FLOW, LBM/HR		
FORWARD BULKHEAD CONDENSATION	1×10^{-2}	
FEED OUT DURING ADVERSE ACCELERATION	N.A.	4.0

analyses have been discussed with JPL personnel and additional performance data will be transmitted to support a final decision.

5.3 OTHER TANKAGE CONCEPTS

Two other tankage concepts were briefly considered for the MMS propulsion module. The first was the TRW Block 5D tankage. This equipment uses an AF-E-332 bladder to achieve positive expulsion of the hydrazine. The spherical task has an internal diameter of 9.9" and is therefore in the same size category as the IUE tankage. The propellant capacity, however, is approximately 70% greater than the IUE. The increased capacity is obtained by significantly reducing the ullage gas volume which, in turn, results in an unacceptedly high blowdown ratio. The low ullage volume results from the fact that the Block 5D is a regulated rather than a blowdown system. As was the case with the IUE, the limited propellant capacity indicated the Block 5D tankage was not an acceptable candidate for the MMS.

The second concept is the tankage system developed for the RCA SATCOM spacecraft. The internal diameter of the tank is identical to the ATS but a capillary device rather than a diaphragm is used to provide propellant management. The system is both flight qualified and operational. Two spacecraft have been orbited; the first launch occurred in December 1975. The Thor Delta 3914 launch vehicle was used on both SATCOM flights. It is reasonable to expect, therefore, that the capillary propellant management device used to supply hydrazine to the SATCOM 0.2 lb. thrusters would be acceptable for the MMS. It should be observed, however, that the SATCOM does not use 5.0 lb thrust engines which raised the question of whether the flow rates required by the MMS thrusters could be satisfied by the SATCOM tankage. The feasibility of simultaneously supplying four 5.0 lb thrusters has been discussed with RCA personnel. The results indicate that the SATCOM tankage is capable of meeting the SPS-1 requirements. Pertinent details of the SATCOM tank system are presented in Table 14.

Both the SATCOM tank shell and PMD are manufactured by Fansteel. RCA, however, designed and developed the PMD and Fansteel is currently making arrangements with RCA which would allow Fansteel to manufacture and market the complete tankage system.

It appears that significant cost savings could result from the use of the SATCOM tankage. The current geometry of the SATCOM tank (Figure 22) would require modification to be compatible with the SPS-1 design constraints.

Table 14. RCA System Propellant/Pressurant Tank

Program Source	RCA SATCOM
Expulsion Device	Propellant Management Device
Volume (Internal)	2350 in^3
Dimensions (Sphere)	16.5 in. dia.
Pressure	
Operating	450
Proof	675
Burst	900
Material	6 A14V Ti
Tank Weight (Includes PMD)	5.2 lbs

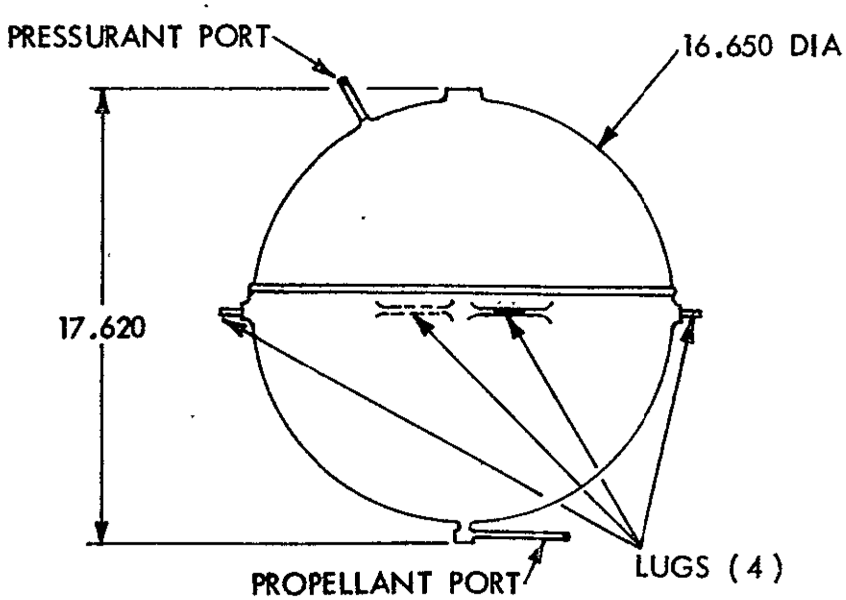


Figure 22. RCA $SATCOM_2$ Tank

6.0 SCHEMATICS AND TANKAGE ARRANGEMENTS

6.1 GENERAL DISCUSSION

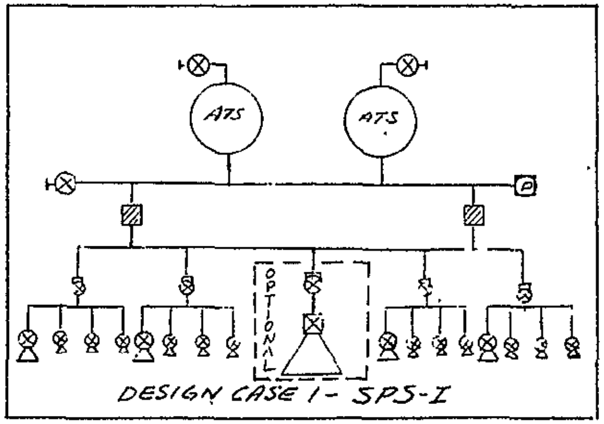
Schematics and tankage arrangements have been generated for a large number of potential configurations. The more viable options are summarized in Table 15 and schematics, component identification and recommended tankage are presented in Figures 23 through 31. The baseline configurations of the SPS-I, SPS-I-A, and SPS-II modules were considered to utilize only 0.2 and 5.0 lb thrusters in accordance with Reference 4. However, because of early concerns over the long burn times required of the SPS-II and uncertainty about the control authority required, a decision was made to examine the use of 150-lb thrusters. Accordingly, each of the following concepts shows 150-lb thrusters in a dashed-line box as an option. The issues and final selection are discussed in Section 6.3 below.

6.2 SCHEMATIC DISCUSSION

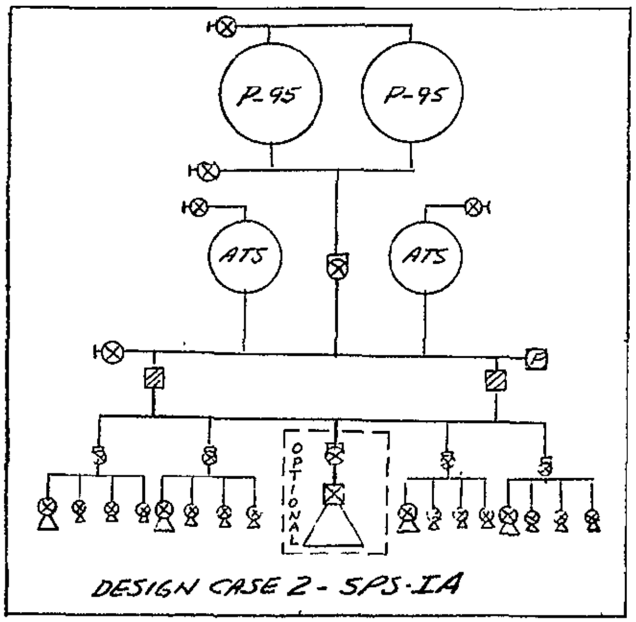
Design Case 1, Figure 23, was designated the baseline and is based on information found in Reference 4. Study of the figure indicates that the loss of any thruster in a REM in the failed open mode would require the shutdown of that entire REM and would impact the operation of the one located directly opposite. The principal reason for this result is that the spacecraft will not remain stabilized with a 5.0 lb. thrust engine firing on one side of the spacecraft, i.e., thruster REM B-1 cannot operate in the steady state mode unless thruster REM D-1 is operating in a similar mode. Should the MMS lose REM's B and D, however, full control can be provided by REM's A and C. Additional mission flexibility would result if the 5.0 lb. thrust engines were latched independently of the 0.20 thrusters. Such a schematic is presented in Figure 24. Loss of a 5.0 lb. thrust engine would still require pulsed operation of the opposite thruster but full attitude control capability would remain. Similarly, loss of a 0.20 lb. thruster group would permit full retention of the translation/orbit adjust capability. Separate latching of the 5.0 lb. thrusters is inherent if the dual seat/dual coil GPS thrusters are used. This arrangement, shown in Figure 25, eliminates the need for latch valves for the 0.20 lb. thrusters.

Table 15. Propulsion System Configuration Definition

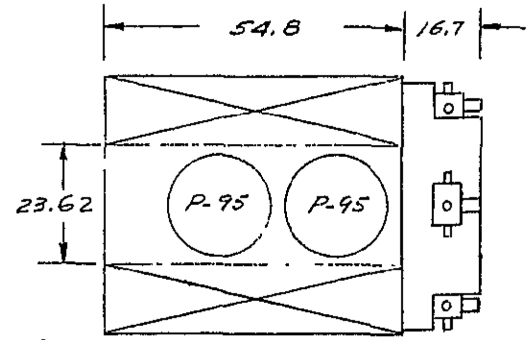
DESIGN CASE	CONFIGURATION	EXPULSION DEVICE		THRUSTER SIZE			REMARKS
		DIAPHRAGM	PMD	150 LBF	5 LBF	0.2 LBF	
1	SPS-I (Baseline)	Yes	No	Optional	Yes	Yes	Use of ATS tanks presents clearance problem relative to tank attachment
2	SPS-IA (with 2 ATS)	Yes	No	Optional	Yes	Yes	SPS-IA is made up of SPS-I plus two HEAO tanks in tunnel
3	SPS-IA W/4 ATS	Yes	No	Optional	Yes	Yes	Propulsion module configuration change required.
4	SPS-IA W/4 GPS or Marots	Yes	No	Optional	Yes	Yes	
5	SPS-II with 4 Shuttle tanks	Yes	No	Optional	Yes	Yes	
6	SPS-II with VO '75 + PMD	No	Yes	Optional	Yes	Yes	Existing PMD may not be suited for hydrazine service
7	SPS-II with VO '75 + IUE Tanks	No Yes	No	Optional	Yes	Yes	VO '75 starts with higher pressure. IUE Tank to be refilled by VO '75 when empty
8	VO '75 + SPS-I	No Yes	No	Optional	Yes	Yes	Uses SPS-I to settle propellants.

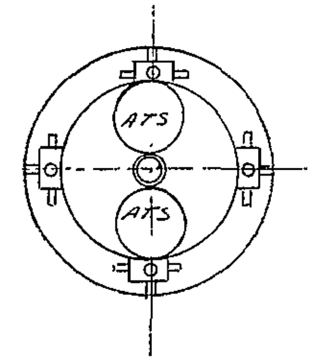


Total Propellant
Capacity at 3:1
Blowdown Ratio = 111 LB_m



Total Propellant
Capacity at 3:1
Blowdown Ratio = 381 LB_m





COMPONENT	DES. CASE 1 QUANTITY	DES. CASE 2 QUANTITY
ATS TANK	2	2
P-95 TANK	0	2
REM		
LATCH VALVE	4	5
FILTER	2	2
FILL/DRAIN, GN_2	2	3
FILL/DRAIN, PROP.	1	2
PRESS XDUCER	1	1
0.2 LBF THRUSTER	12	12
5.0 LBF THRUSTER	4	4
150 LBF THRUSTER	OPTIONAL	OPTIONAL
	—	—

Figure 23. Design Cases - 1 and 2

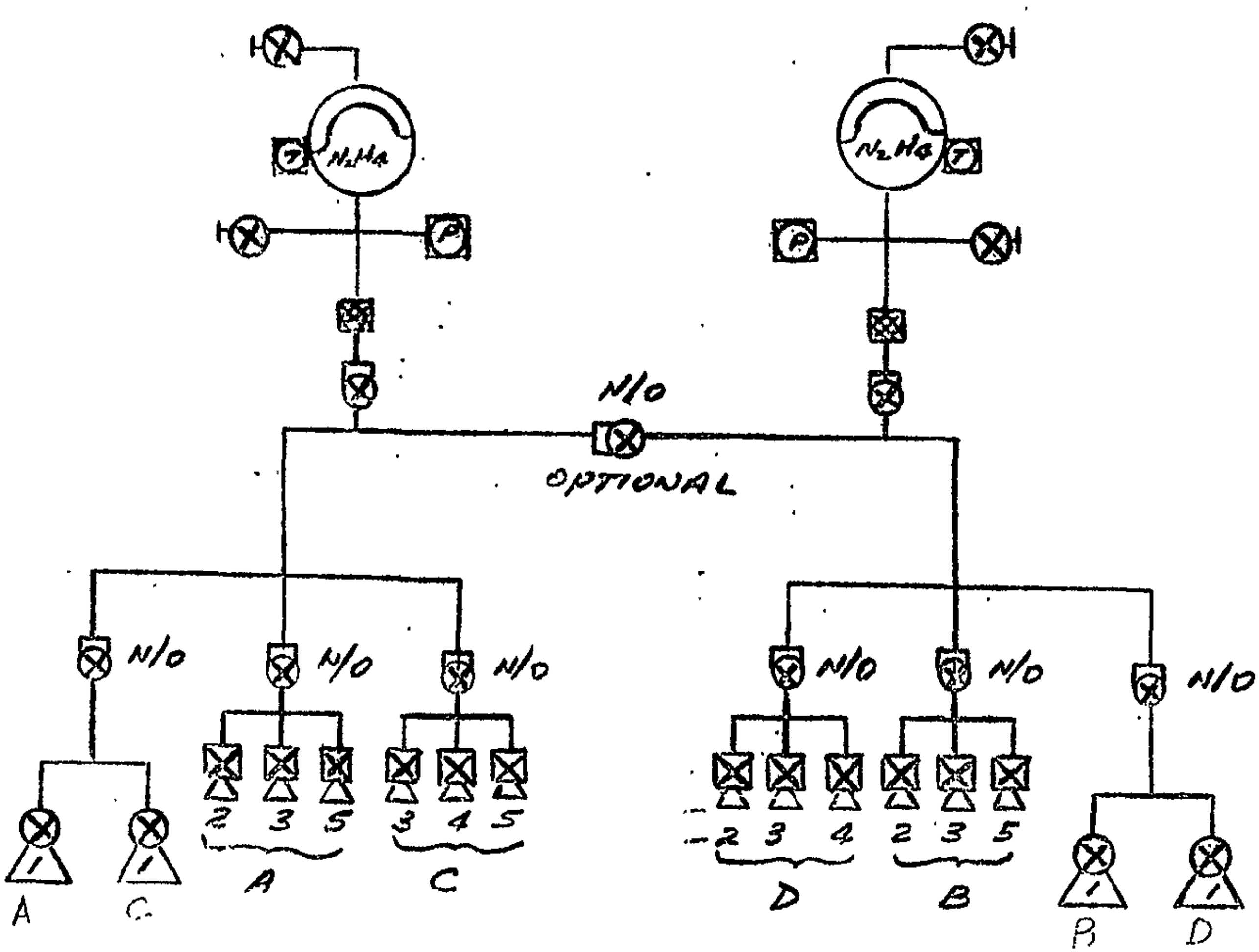


Figure 24. SPS-1 Propulsion System With Single TCV's

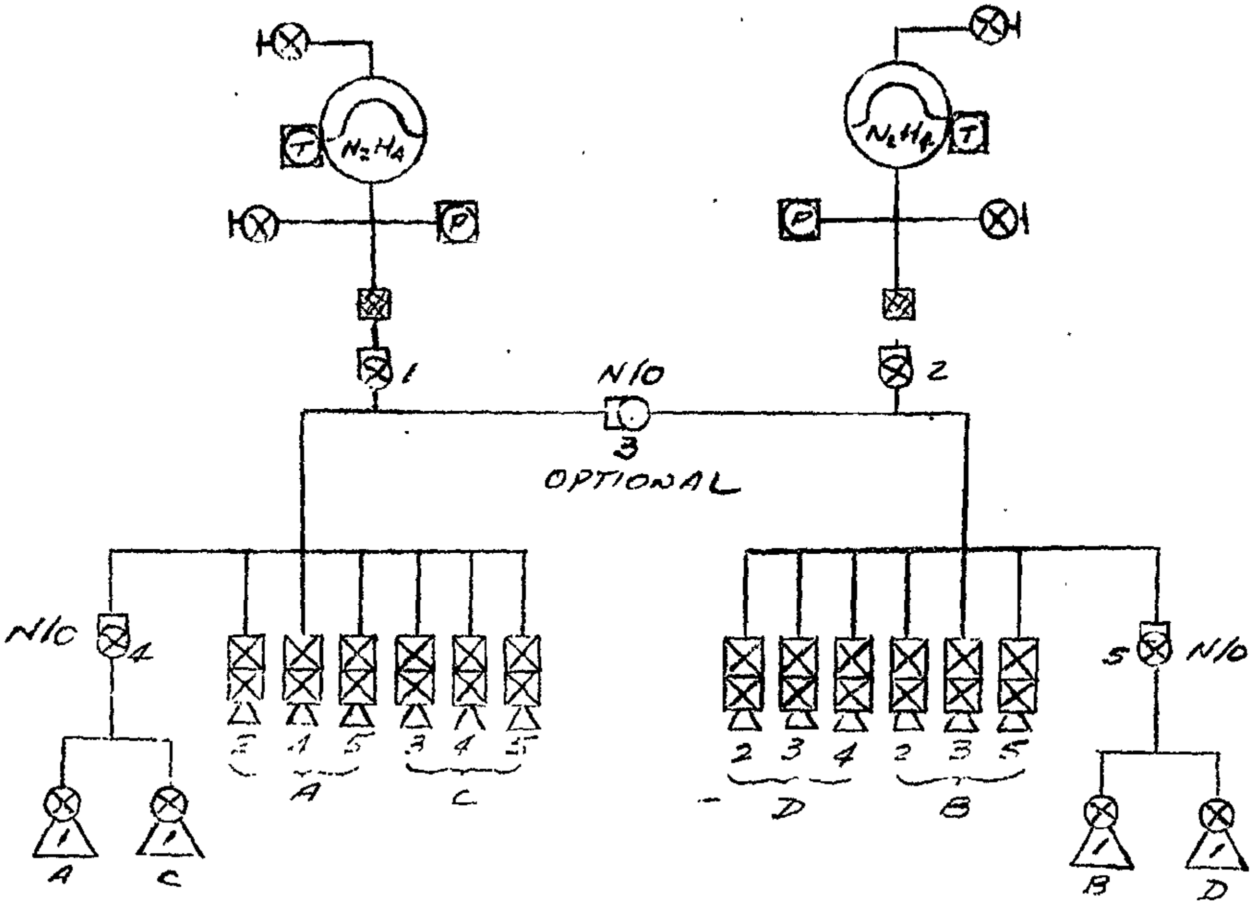
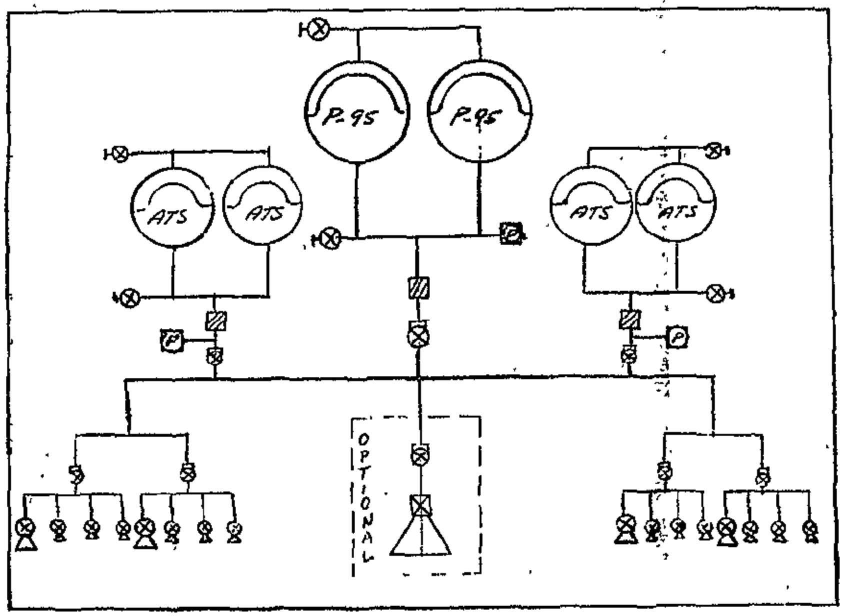
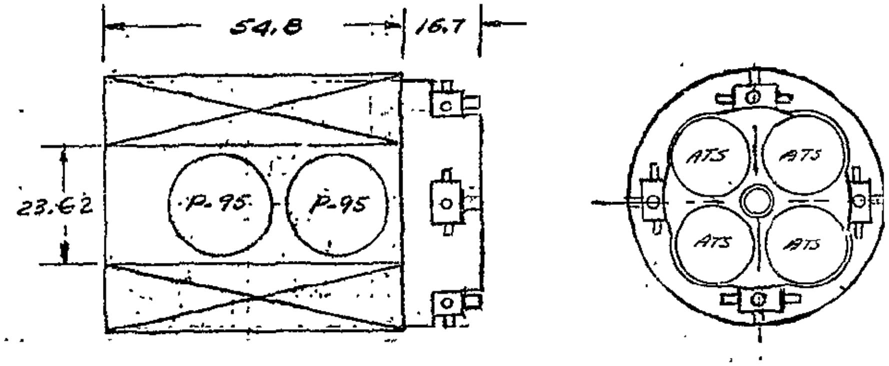


Figure 25. SPS-1 Propulsion System With Redundant TCV's

Total Propellant
Capacity at 3:1
Blowdown Ratio = 492 LB_m

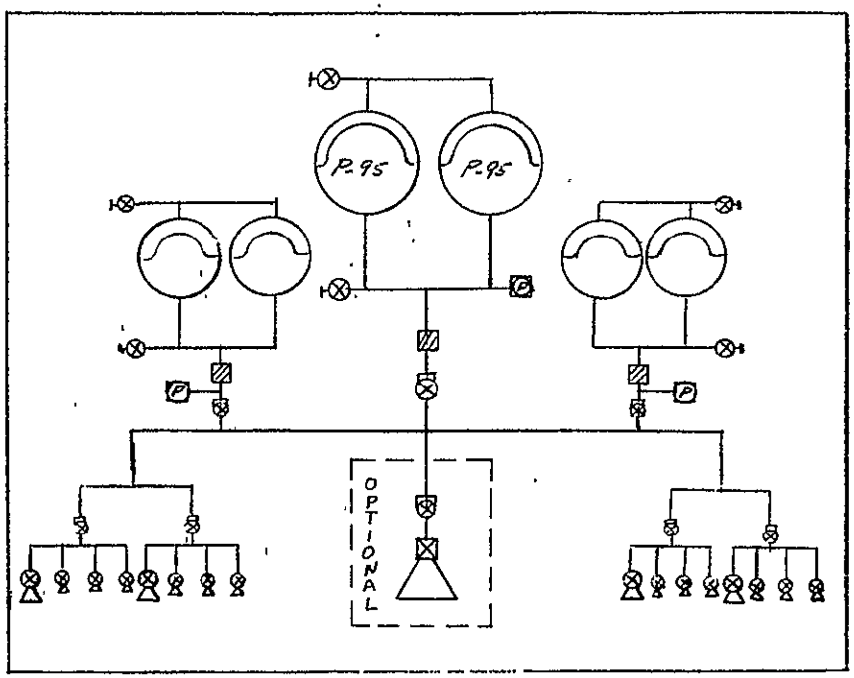


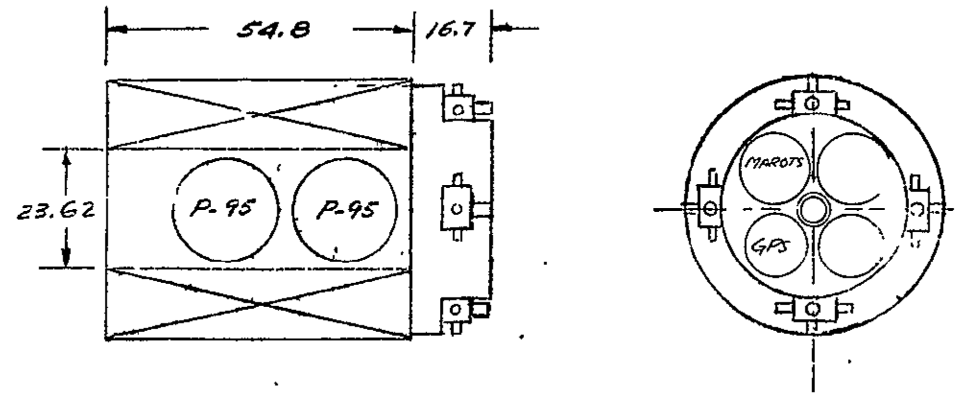


COMPONENT	QUANTITY
P-95 TANK	2
ATS TANK	4
REM	
LATCH VALVE	7
FILTER	3
FILL/DRAIN, GN_2	3
FILL/DRAIN, PROP.	3
PRESS XDUCER	3
0.2 LBF THRUSTER	12
5.0 LBF THRUSTER	4
150 LBF THRUSTER	OPTIONAL
TOTAL	—

Figure 26. Design Case 3

Total Propellant
Capacity at 3:1
Blowdown Ratio
= 446 LB_m with 4 MAROTS Tanks
= 374 LB_m with 4 GPS Tanks





COMPONENT	DES. CASE 4A QUANTITY	DES. CASE 4B QUANTITY
MAROTS TANK	4	0
GPS TANK	0	4
P-95 TANK	2	2
REM		
LATCH VALVE	7	7
FILTER	3	3
FILL/DRAIN, GN_2	3	3
FILL/DRAIN, PROP.	3	3
PRESS XDUCER	3	3
0.2 LBF THRUSTER	12	12
5.0 LBF THRUSTER	4	4
150 LBF THRUSTER	OPTIONAL	OPTIONAL
TOTAL	—	—

Figure 27. Design Case 4

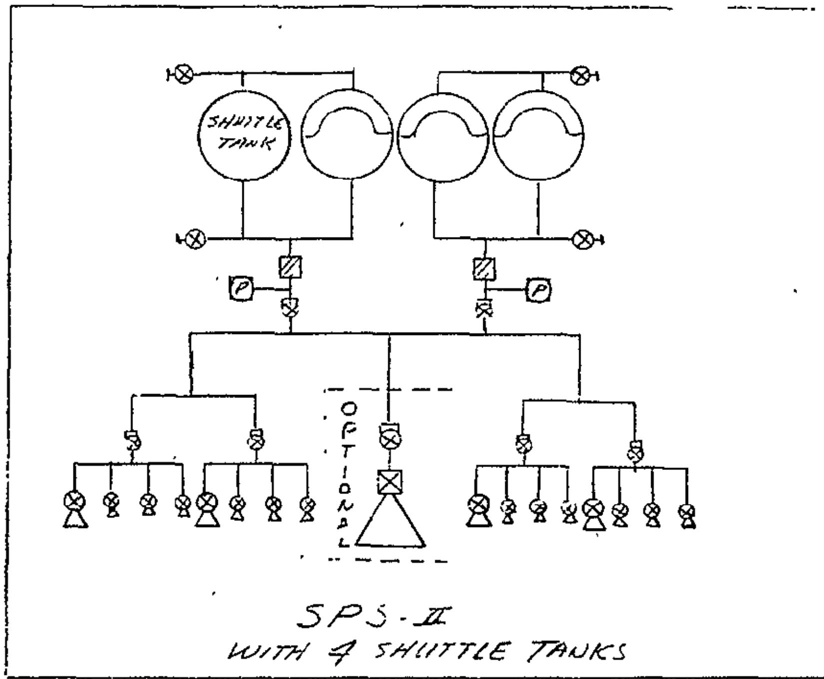
The factors discussed above relative to thruster/latching valve arrangment also apply to design cases 2, 3, and 4. The basic differences are found in the tankage options selected. The capability of each option is identified in the applicable figure.

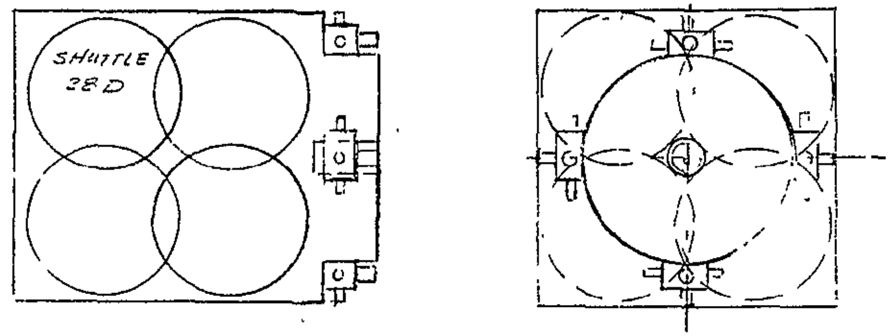
Design case 5, Figure 28, addresses the Shuttle-launched SPS-II module. The comments above relative to the thruster/latch valve arrangement also apply to SPS-II. The tankage selected is the same as is being used in the Shuttle Orbiter Auxiliary Power Unit hydrazine system. This unit is very similar to the tankage being produced by PSI for the JPL MJS spacecraft. The Shuttle design uses a simpler mounting technique and does not require the MJS diaphragm restraining device. Design case 5 also incorporates the 150 lb. thrust orbit adjust engines. A further analysis is required to determine if the 5.0 lb. thrust engines are necessary when the 150 lb. thrust engines are used. The potential replacement in the REM of the 5.0 lb. thruster with a 0.20 lb. thrust engine is also an option. It has been established that the axial force requirements of SPS-I can be satisfied with a 0.2 lb. thruster but obviously a longer burn time is required.

Design Case 6, Figure 29, employs the VO '75 MMH tank, including the propellant management device, PMD. Details on this tankage are presented in Section 5.0, Tankage Options and Issues. It is apparent that, with respect to component costs, Case 6 is considerally more cost effective than Case 5. Subsequent to fueling and prior to orbit placement by the Shuttle, the tankage will undergo a number of acceleration orientations and a time line study should be conducted to assure that propellant will be available for the initial stabilization and ΔV burn operations. Although it appears that hydrazine may be directly substituted for the MMH (see Section 5) some concern still exists which may require a test program to resolve.

Design Case 7, Figure 30, was generated as a possible solution to the situation wherein it was found inadvisable for economic or technical reasons to use the VO '75 tank with PMD. The Case 7 design incorporates two positive expulsion IUE tanks to provide enough propellant to achieve a propellant settling burn. When operational altitude has been reached and the orbit transfer thruster shut down, propellant for orbital operations will not be available from the VO '75 tank and the IUE capacity is inadequate. Periodic

Total Propellant
Capacity at 3:1
Blowdown Ratio = 1100 LB_m

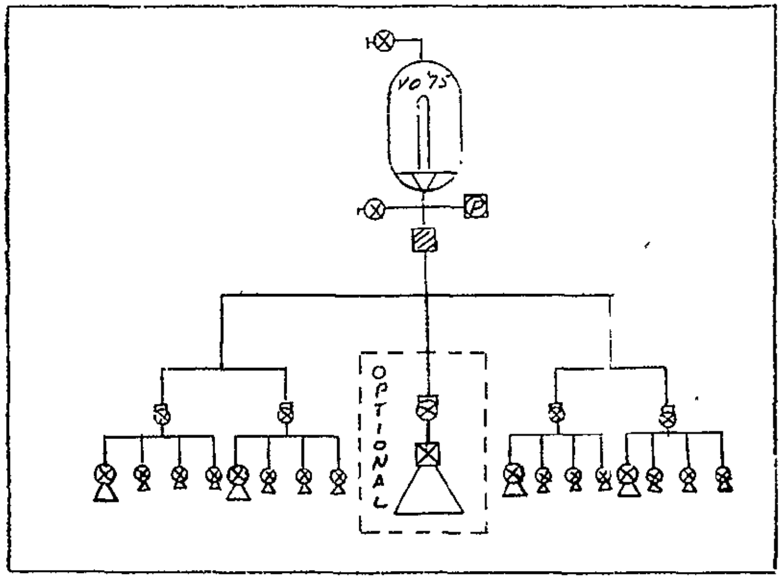


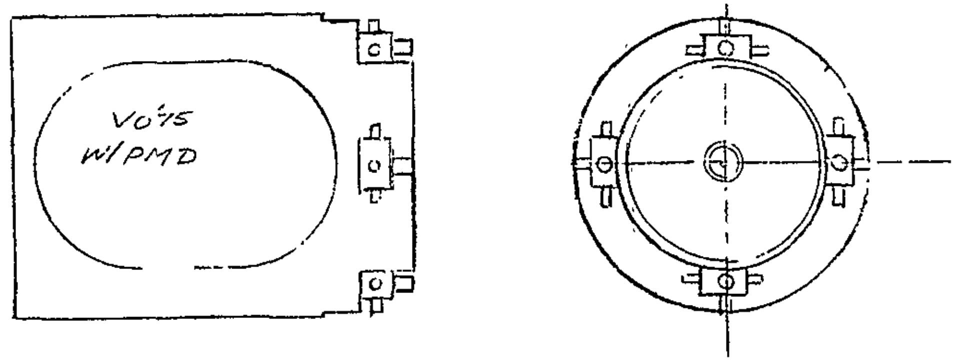


COMPONENT	QUANTITY
SHUTTLE TANK	4
REM	
LATCH VALVE	7
FILTER	2
FILL/DRAIN, GN_2	2
FILL/DRAIN PROP	2
PRESS XDUCER	2
0.2 LBF THRUSTER	12
5.0 LBF THRUSTER	4
150 LBF THRUSTER	OPTIONAL
TOTAL	—

Figure 28. Design Case 5

Total Propellant
Capacity at 3:1
Blowdown Ratio = 1062 LB_m

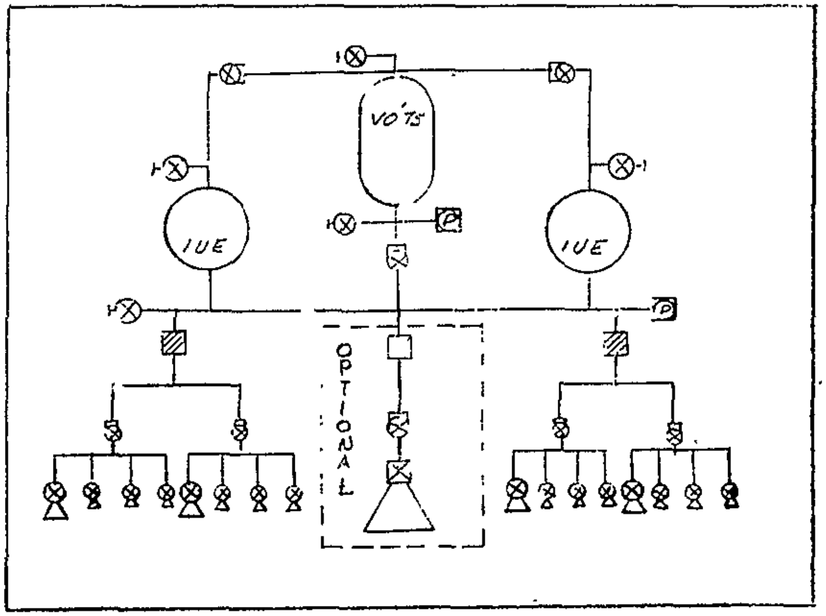


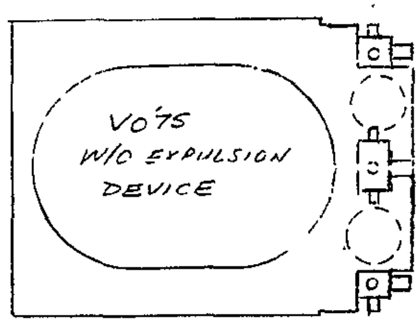


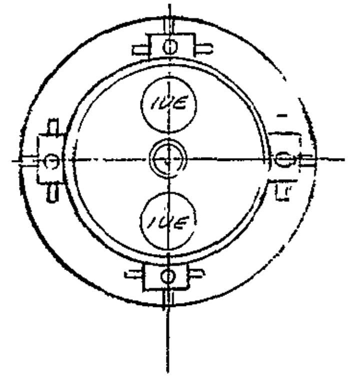
COMPONENT	QUANTITY
VO'75 TANK (PMD)	1
REM	
LATCH VALVE	5
FILTER	1
FILL/DRAIN, GN_2	1
FILL/DRAIN, PROP.	1
PRESS XDUCER	1
0.2 LBF THRUSTER	12
5.0 LBF THRUSTER	4
150 LBF THRUSTER	OPTIONAL
TOTAL	—

Figure 29. Design Case 6

Total Propellant
Capacity at 3:1
Blowdown Ratio = 1082 LB_m







COMPONENT	QUANTITY
IUE TANK	2
VO'75 TANK	1
REM	
LATCH VALVE	8
FILTER	3
FILL/DRAIN, GN_2	3
FILL/DRAIN, PROP.	2
PRESS XDUCER	2
0.2 LBF THRUSTER	12
5.0 LBF THRUSTER	4
150 LBF THRUSTER	OPTIONAL
TOTAL	

Figure 30. Design Case 7

recharging of the IUE tankage was considered but does not appear attractive due to the fact that a propellant settling burn would be required and the pressure schedule of the VO '75 tank must be such that it is always greater than the IUE tank.

Design Case 8, Figure 31, evolved as a solution to the IUE recharging problem. This case incorporates a complete SPS-I module with SPS-II. SPS-I can provide propellant settling capability in addition to performing the normal on-orbit functions. While Case 8 may not result in a lower components cost relative to Case 6, the manufacturing/assembly/test operations may offer compensating cost savings. Further, substantial savings appear achievable over Case 5.

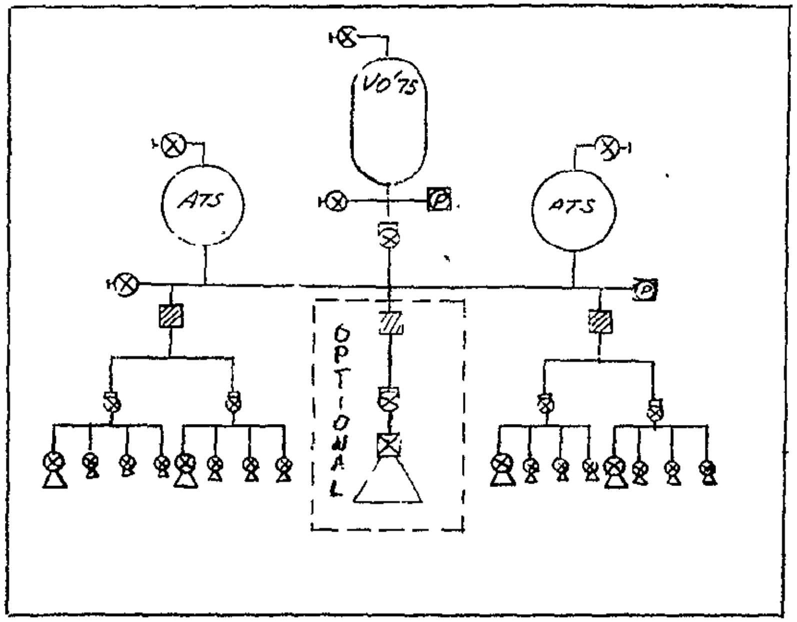
6.3 150-LB THRUSTER ISSUES

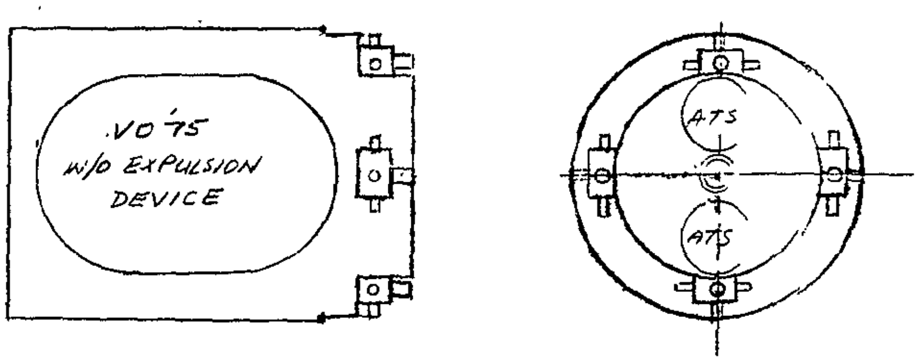
As discussed in Section 2.2, the thrust provided by the baseline MMS Propulsion Subsystem with 5.0-lb thrusters results in mission thrust-to-weight ratios in the range of 0.006 to 0.002. For the Shuttle launched missions this results in very long thrusting times on the order of one orbit. In order to improve this situation, a brief examination was conducted of alternative concepts utilizing 150-lb thrusters either as supplements to the baseline configuration or as replacements for the 5.0-lb thrusters.

For either SPS-I or SPS-II, it was considered that dual 150-lb thrusters would be required to meet the reliability goals. Preliminary estimates of the travel of the center of gravity for various potential mission configurations led to mounting the thrusters on a gimballed platform in order to assure adequate margins for control. This assembly produced an overall length requirement which was not compatible with the volumetric restrictions on SPS-I. The concept could be utilized for SPS-II but, as indicated in Section 2.5, the transportation cost formulae are strong drivers for decreasing the overall length.

The final selection was to return to the baseline configuration. The analyses described in Sections 2.2 and 2.4 have shown that the 5.0-lb thrusters can meet the mission requirements and there appears to be no significant advantage to the use of the larger thrusters to offset the increased complexity, cost, and length penalties involved.

Total Propellant
Capacity at 3:1
Blowdown Ratio = 1173 LB_m





COMPONENT	QUANTITY
ATS TANK	2
VO'75 TANK	1
REM	
LATCH VALVE	6
FILTER	3
FILL/DRAIN, GN_2	3
FILL/DRAIN, PROP.	2
PRESS XDUCER	2
0.2 LBF THRUSTER	12
5.0 LBF THRUSTER	4
150 LBF THRUSTER	OPTIONAL
TOTAL	

Figure 31. Design Case 8

7.0 EXAMINATION OF LOW COST SYSTEMS OFFICE (LCSO) COMPONENTS

This section presents the evaluation results of the LCSO equipment as they apply to the MMS Propulsion Module. The LCSO equipment examined include the Martin-Marietta Propellant Control Assembly (PCA) (Figure 32) the individual components contained in the PCA (Figure 33) and the standardized 0.2 lbf thruster (Figure 14). All of the LCSO equipment were developed for the MJS program under the technical direction of JPL. The manufacturers of the LCSO equipment and their qualification status are shown on Table 16.

7.1 PROPELLANT CONTROL ASSEMBLY (PCA)

The Propellant Control Assembly is used to distribute pressurized hydrazine from the storage tank to the thrusters. The PCA as shown in Figure 32 consists of a bistable (latching) solenoid-actuated value, a filter and a pressure transducer with associated manifolding and mounting brackets. When the latching valve is opened, filtered hydrazine is distributed throughout the propulsion system up to the propellant inlet control valve.

The LCSO Propellant Control Assembly is fabricated by the Martin Marietta Corporation. Individual components making up the PCA were subcontracted and procured by Martin. A list of the components and their respective manufacturers are presented on Table 16. The overall dimensional envelope of the PCA is 3.50" x 6.03" x 10.50".

The factors used to determine the suitability of the standardized PCA for use with the MMS include performance, packaging and cost. To collect the data necessary to conduct the evaluation, JPL, Martin, and the individual subcontractors were contacted. A cost estimate for the PCA was obtained from the Martin Marietta Corporation and the results are presented in a separate appendix. A detailed discussion of performance of each LCSO component is presented in subsequent paragraphs. In general, the following was found:

1. The qualified flow rate of the bistable latch valve is nearly 12 percent below that required by the two-5.0 lbs thrusters.
2. The allowable pressure drop across the LCSO filter is beyond the acceptable level of the SPS-I.

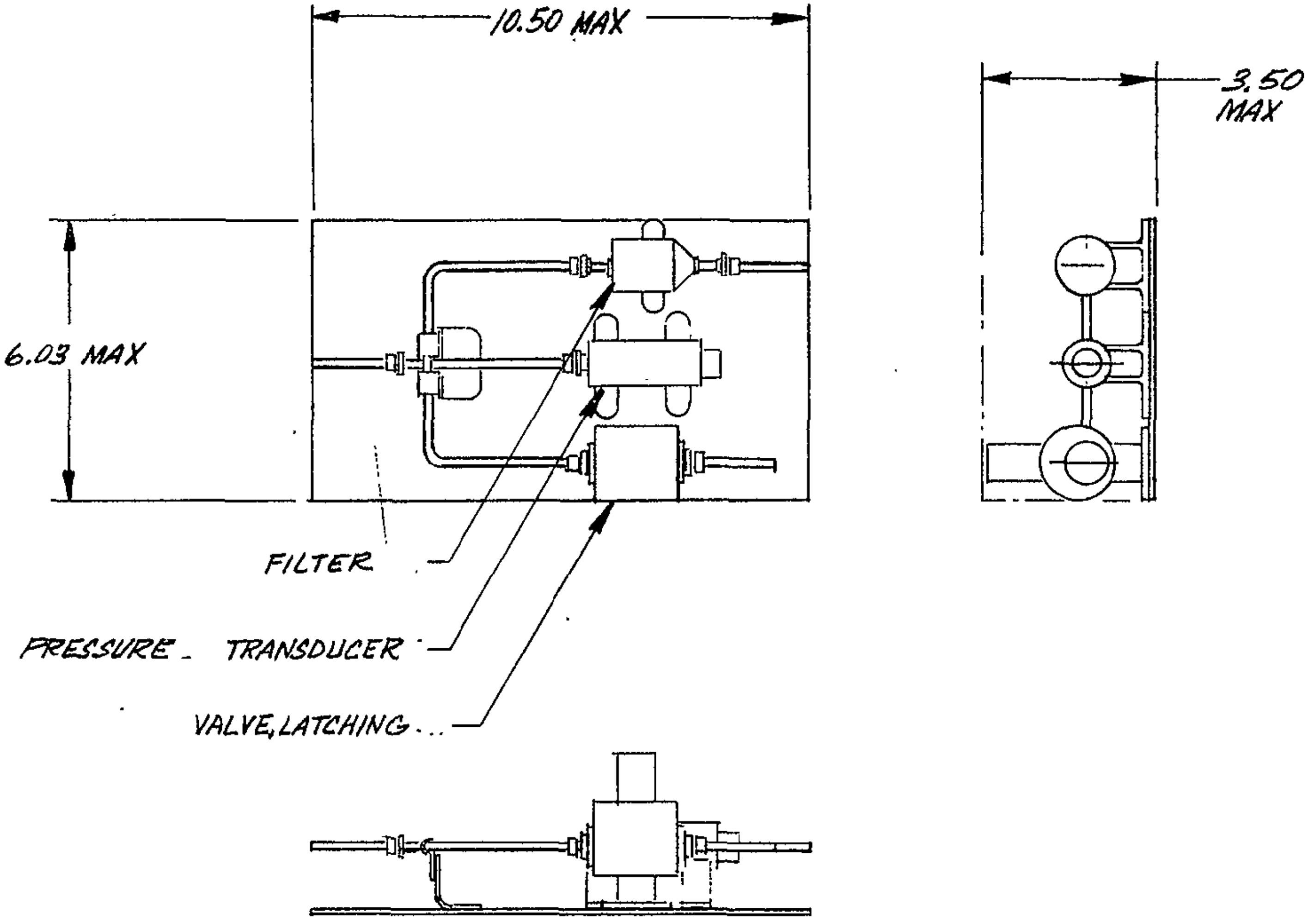


Figure 32. PCA Interfaces and Envelope

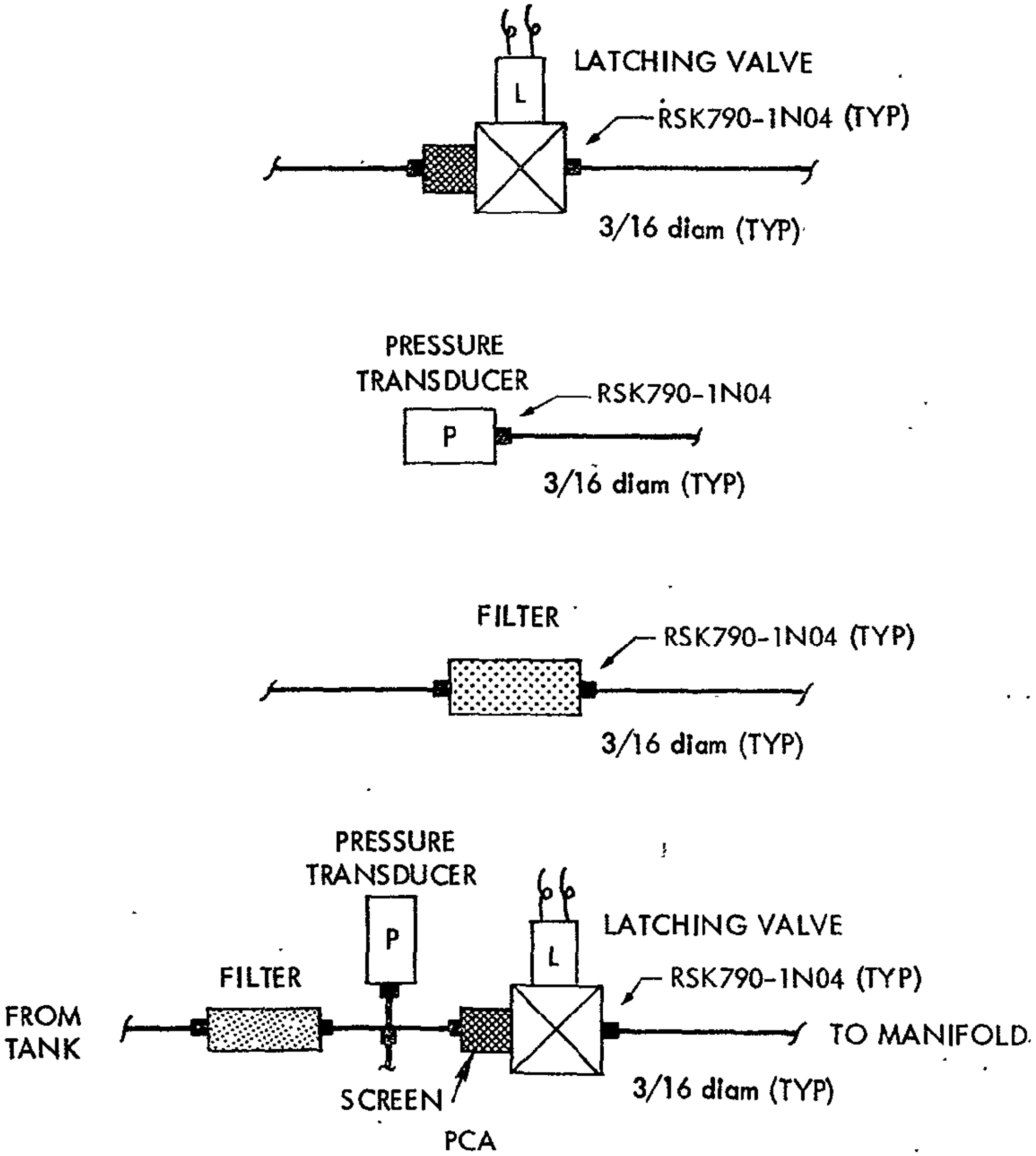


Figure 33. Standardized PCA and Individual Components

Table 16. LCSO Component List

COMPONENT NAME	MFR	UNIT Wt., LB_m	QUAL. STATUS	PREVIOUS USAGE
0.2 LBF Thruster	Rocket Research	0.70	In Progress	MJS
Bistable Latch Valve	Marquardt	0.61	Complete	MJS
Fill/Drain Valve	Pyronetics	0.25	Complete	ISEE, BLK 5-D, HEAO GPS, MJS
Pressure Xducer	Std. Controls	0.5	Complete	Pershing, Trident, Lance MJS
Filter	Wintec	0.30	Complete	MJS
Propellant Control Assembly	Martin	1.5 max.	Complete	MJS

3. Because of the fixed component arrangement and envelope of the PCA, Figure 32, the PCA is not well suited for installation in SPS-I.

7.1.1 Bistable Latch Valve

The LCSO bistable latch valve is a magnetic latching coaxial flow solenoid operated valve designed for long term hydrazine exposure flow control. The valve incorporates position indicator switches for remote monitoring of poppet position. The valve is of all welded construction with absolute hermetic seals. Materials of construction in contact with the hydrazine are stainless steel and an elastomer poppet/seat interface seal of ethylene propylene terpolymer (AF-E-102).

Qualification testing has been conducted by Marquardt Company on two valves from production lots. Both valves passed qualification testing after having been subjected to sine and random vibration, pyro-shock, functional, cycle life and contamination sensitivity tests. Table 17 summarizes the acceptance test performance characteristics of the two test valves. The requirements specified for the test units are those required by the MJS per Martin-Marrieta Corporation Specification PD4700191L.

Examination of the baseline SPS-I configuration indicates that the maximum flow rates, which occur at the beginning of mission life, required to sustain the firing of each set of 2-5 lbf thrusters is 0.043 pps. While this flow rate is somewhat higher than the demonstrated rates (up to .038 pps) of the standardized latch valve, it is the opinion of a JPL contact that the existing valve design should have no problem meeting the higher flow rate (0.043 pps) of the 5 lbf thrusters provided that this is the absolute maximum. However, it should be recognized that flow rates can vary depending on the inlet pressure and the corresponding thrust and specific impulse characteristics of the thruster. If the thrust level is higher and the specific impulse is lower than the predicted values, the resultant demand flow rate will be higher. For this reason, the use of the LCSO bistable latch valve with the 5 lbf thruster is marginal in the sense that it may limit the beginning of life performance. Calculations show that the initial thrust of the 5 lbf thruster will decrease about 12 percent at the qualified flow rate of 0.038 pps. No problem is expected with the 0.2 lbf thruster as the demonstrated capability of the bistable latch valve is well within its requirements.

Table 17. Acceptance Test Performance Characteristics Summary

	Test Unit No.	1	2
	Serial No.	0010	0008
Parameter	Requirement	Measured Value	
Armature Stroke	.017 - .018 in.	.0170 in.	.0170 in.
Latch Force - Closed - Open	2.2 lb. min. 2.2 lb. min.	4.40 lb. 3.60 lb.	3.30 lb. 3.20 lb.
Weight	*0.75 lb. max.	0.61	0.61
Insul. Resistance between isolated points	*>100 megohms	>170,000 megohms	>280,000 megohms
Dielectric Strength between isolated points	*<0.10 milliamps	<.060 milliamps	<.060 milliamps
Power - Open Coil - Close Coil	*15 watts max. @ 32 vdc, $40^{o}F$	12.21 watts 12.21 watts	12.34 watts 12.18 watts
Open Threshold Voltage	*17 vdc max.	11.68 vdc	12.69 vdc
Close Threshold Voltage	*15 vdc max.	12.93 vdc	11.38 vdc
Open Response	*20 ms max.	10.0 ms	9.9 ms
Close Response	*15 ms max.	8.0 ms	6.0 ms
Reverse Relief Pressure	*-150 psid max.	-135 psi	-98 psi
Flow Rate @ 10 psid	*>.03 pps H_2O	.038 pps	.035 pps
Internal GN_2 Leakage	*<1.0 scch	0.0 scch	0.0 scch

*Denotes PD4700191L Spec Requirement

The LCSO bistable valve features a reverse pressure relief which allows the downstream pressure to relieve itself whenever it reaches a pressure greater than the upstream pressure by a value between 98 and 135 psi. This feature is typically provided on many latching valves used for isolation purposes. The latching valves used on the MMS provide thruster isolation and the pressure relief feature prevents over pressurization of the lines between a closed latch valve and the thrusters controlled by that valve.

7.1.2 Pressure Transducer

The LCSO pressure transducer is one of the 213-75 series of transducers developed and qualified by Standard Control Inc. for a number of programs including the MJS, Trident, Pershing and Lance. Figure 34 shows the envelope of the 213-75-340 pressure transducer designed to satisfy the requirements of JPL specification C5511302. Two other candidate pressure transducers of the same series as the MJS transducer were also recommended by Standard Control. Performance of the 3 pressure transducers is compared on Table 18. The LCSO pressure transducer is made of 15-5 PH CRES and the others are constructed of 17-4 PH and 304L. All three materials have been proven to be compatible with hydrazine.

7.1.3 System Filter

The LCSO filter contains a metallic element which provides 18-micron, absolute filtration. The filter inlet and outlet ports are 1.5 inches long of 3/16 inch diameter 404L CRES tubing. Allowable pressure drop across the filter is 5 psid at a flow rate of 0.03 pps of water.

Examination of the LCSO filter indicates that the pressure drop is excessively high. A comparable filter, made by the same manufacturer, Wintex, for the GPS is better suited for the MMS. The characteristics of the GPS filter are given below:

Part number	MC286-0064
Operating pressure	400 psig
Proof pressure	600 psig
Burst pressure	1600 psig
External leakage	1×10^{-6} secs, helium

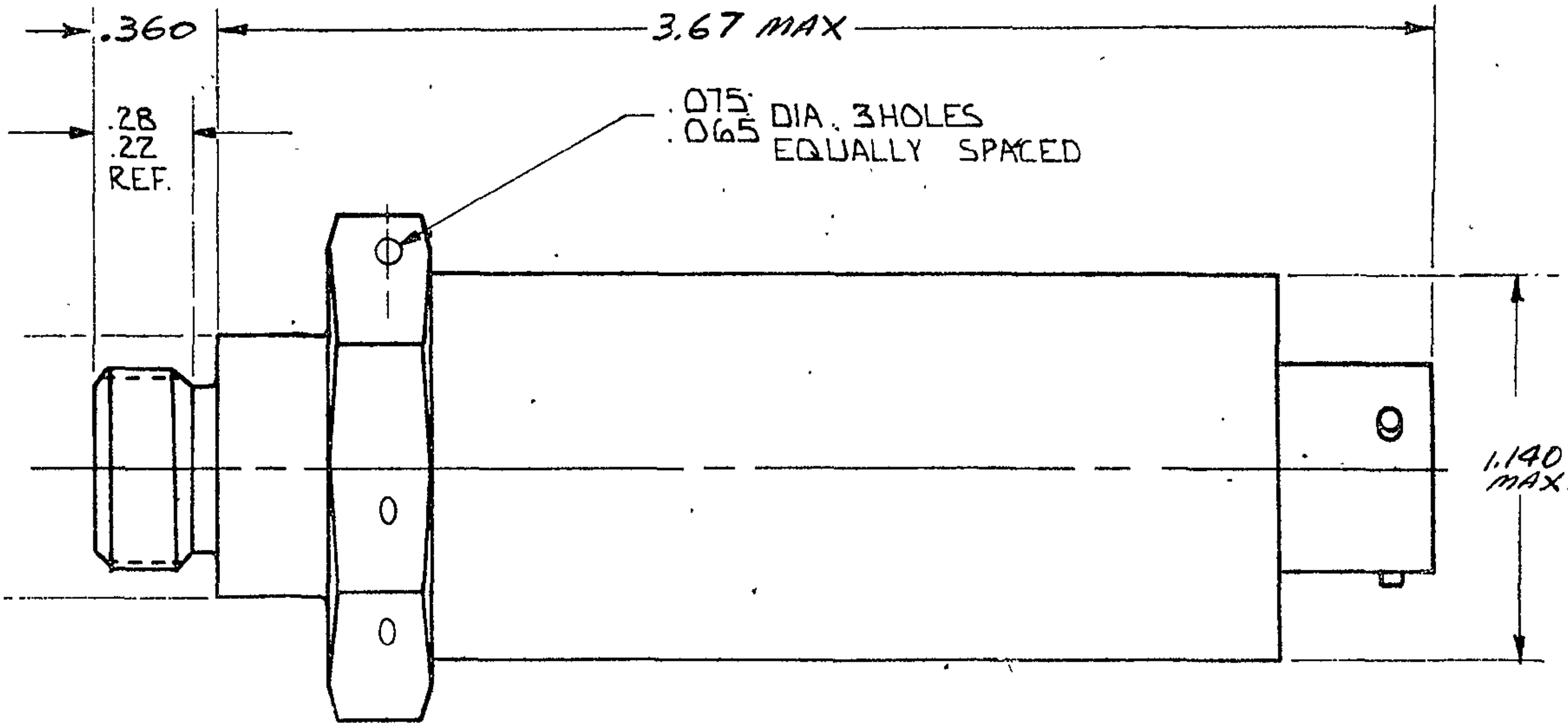


Figure 34. LCSO Pressure Transducer P/N 213-75-340

Table 18. Pressure Transducer Characteristics
(Standard Controls)

	P/N 213-75-340 (JPL)	P/N 213-75-330-04	P/N 213-75-280 (Martin-Marietta)
Pressure range, psia	0 - 1200	0 - 500	0 - 500
Proof pressure, psig	2400	750	1000
Burst pressure, psig	4800	2000	2000
Input voltage, vdc*	24 ± 2	28 ± 2.8	22 - 32
Input power, watts	0.25	1.4	0.45
Output load current, MA	0.01 max	0.005	
Output @ 0 pressure, vdc	0.100 ± .050	0 ± 0.05	.050 +0.100 -0.000
Output @ rated pressure, vdc	2.950 ± .050	5 ± 0.05	3.000 +.000 -0.100
Temp. range, °F	-20 to +160	-30 to +160	+10 to +150
Linearity	± .50% FS		± 0.5% FS max
Hysteresis	± .20% FS		± 0.2% FS max
Repeatability	± .10% FS		± 0.1% FS max
Total error	± 1% FS band @ 77F	± .15% FS	
Weight, lbm	0.5 max	0.5 max	0.6 max

*Reverse polarity protected.

Flow rate and pressure drop	0.06 pps, hydrazine, at 3 psid
Filtration rating, ABS	15 microns
Weight	0.30 lb (actual)

7.2 PROPELLANT/GAS FILL AND DRAIN VALVE

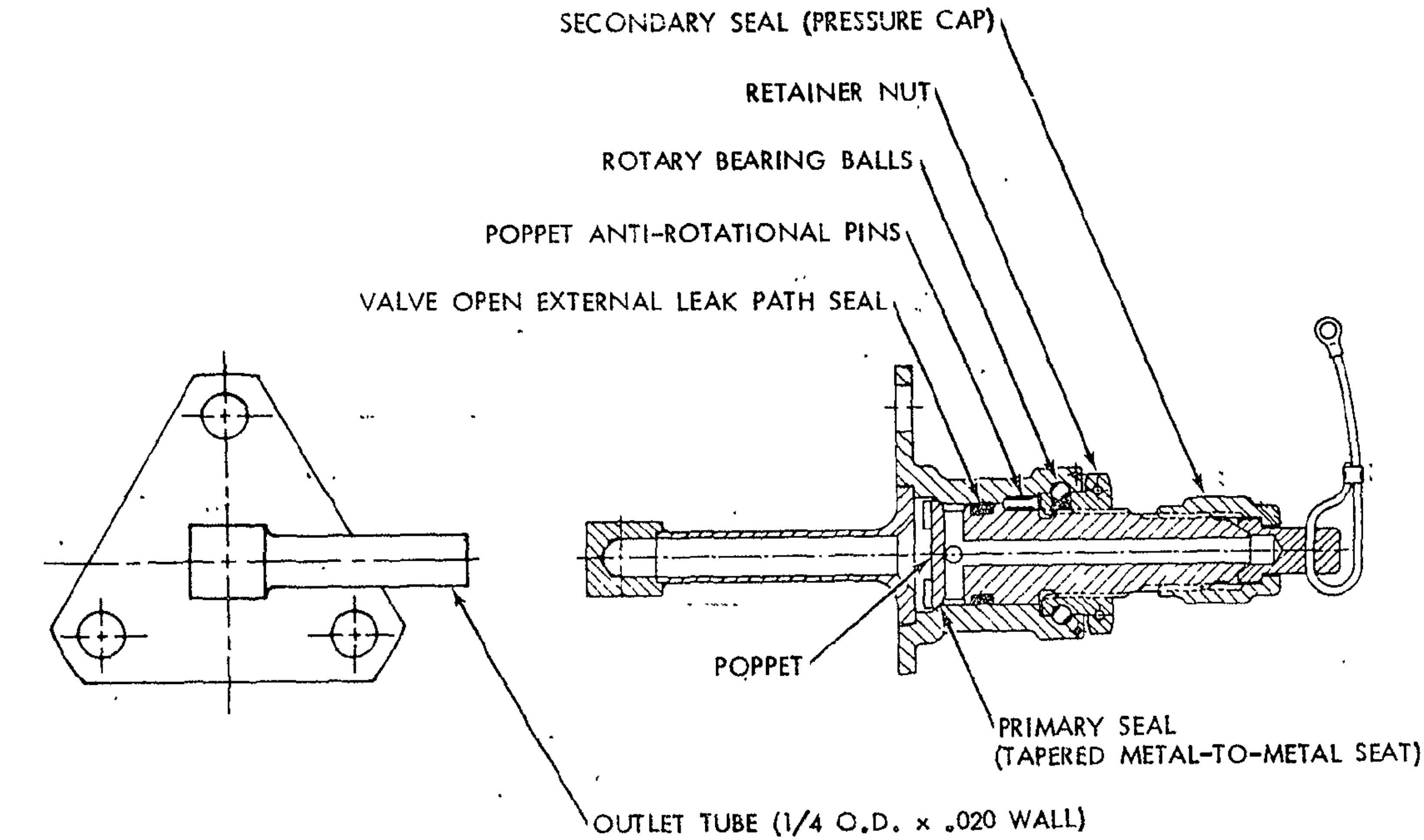
The LCSO fill and drain valve (Figure 35) is a stainless steel, in-line, flange mounted, manually operated valve. Materials of construction include 304L CRES body, 17-7 PH CRES poppet and retainer, 440C CRES pins, EPR O-ring with teflon backup ring, teflon pin and 440C stainless steel balls. The primary sealing function is accomplished by the poppet engaging a tapered seat in the valve body which forms a metal-to-metal seal. The secondary (redundant) sealing function is accomplished by means of a cap and a conical aluminum seal on the flared tube inlet post. During loading of propellant or pressurant, the cap on the flared tube inlet post is removed and the servicing line attached. The valve is opened by turning the outer nut approximately 3/4 of a turn counter-clockwise. When servicing is complete, the valve is first closed by turning the outer nut clockwise and torqueing it to a specific valve. The flared tube protective cap is then installed and torqued.

The performance of the LCSO fill and drain valve is shown on Table 19. To preclude human errors during servicing, the propellant and gas fill and drain valves should be configured with different size tube diameters such that one cannot be mistaken for the other.

7.3 THRUSTER (0.2 LBF)

The LCSO 0.2 lbf thruster is manufactured by Rocket Research. This thruster is basically the same as the 0.1 lbf thruster used on GPS except for the propellant inlet valve. The LCSO thruster employs a single seat Moog valve (Figure 36), whereas the GPS uses a series redundant Wright Components valve.

The LCSO 0.2 lbf thruster is pictorially shown in Figure 14. It consists of two major subassemblies: a thrust chamber assembly and a Moog Model 51-109 solenoid valve. The thrust chamber assembly includes the injector, nozzle, a decomposition chamber formed by the thruster chamber body, bed plate, and catalyst, catalyst heater, temperature sensor and thermal shield. The injector assembly consists of a 0.010 inch I.D. capillary tube which carries the propellant to the catalyst bed, a downstream flange which adopts the injector to the



PICTORIAL CROSS SECTION

Figure 35. Fill and Drain Valve (MC284-0408-0001 & -002)

Table 19. Fill and Drain Valve (MC284-0408-0001 & -0002, Pyronetics)

● DESIGN PARAMETERS

OPERATING PRESSURE	400 PSIG
PROOF PRESSURE	600 PSIG
BURST PRESSURE	1600 PSIG
LEAKAGE	
INTERNAL	1×10^{-7} SCCS, HELIUM
EXTERNAL	1×10^{-5} SCCS, HELIUM
INLET PORT	
-0001 (NITROGEN)	3/16 INCH FLARED TUBE
-0002 (HYDRAZINE)	1/4 INCH FLARED TUBE
ENDURANCE	100 OPEN/CLOSE CYCLES
OPERATING TEMPERATURE RANGE	45°F TO 100° F
FLOW RATE	0.06 LB/SEC HYDRAZINE AT 20 PSI DELTA
WEIGHT:	0.25 LB MAX.
	0.19 LB (ACTUAL)

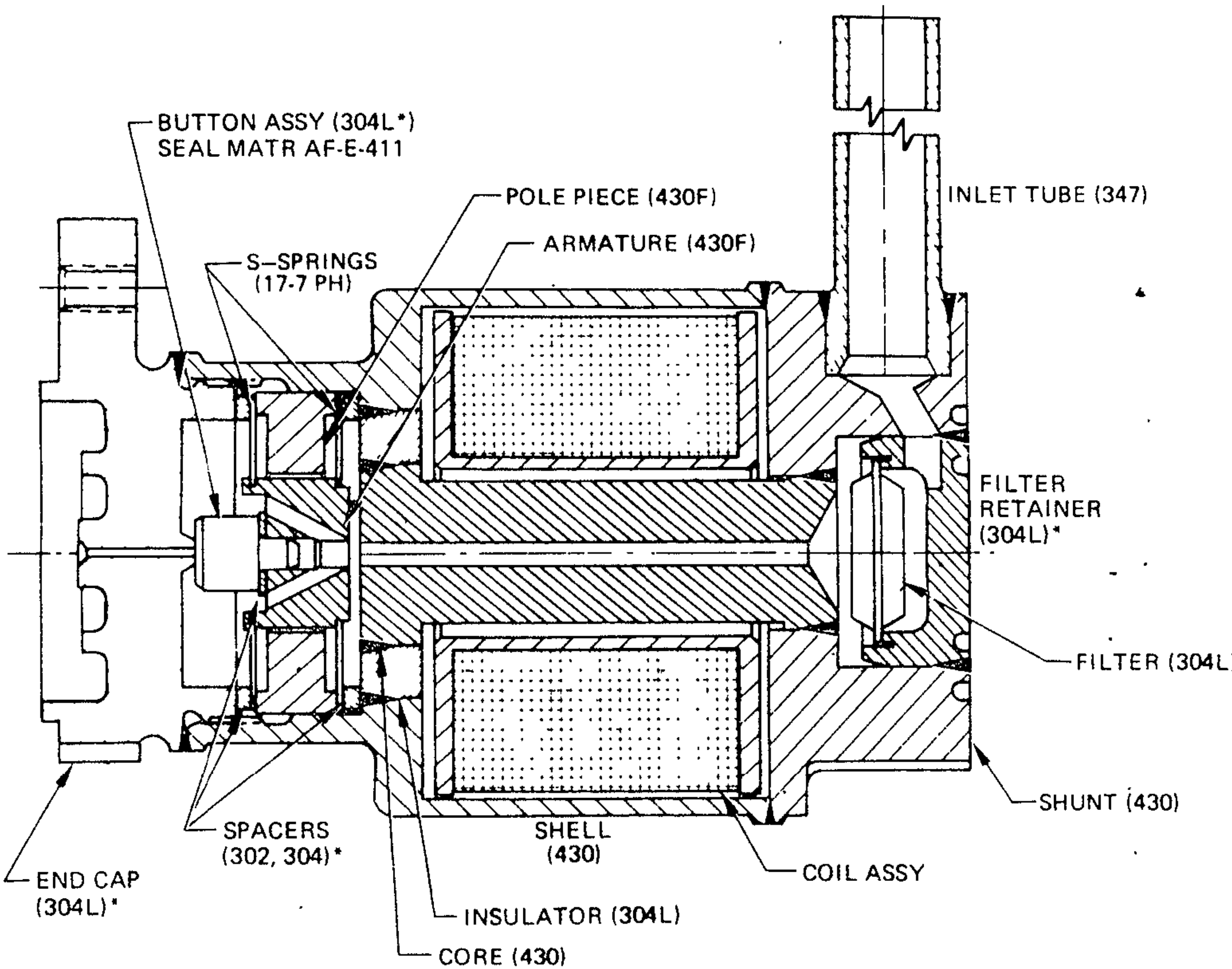


Figure 36. PV-MOOG Model 51-109

decomposition chamber and thermal standoffs which limit heat transfer from the hot chamber to the valve flange. A thermal shunt, with one end attached to the capillary tube and the other end to the upper injector flange, limits the heat buildup in the capillary tube. Two circular 100-mesh screen discs, oriented at 45 degrees relative to one another, are located at the downstream end of the capillary tube. The screen discs serve to distribute the propellant across the surface of the catalyst and also prevent catalyst fine migration into the capillary tube.

The nozzle assembly is welded to the decomposition chamber and contains the chamber pressure top. The nozzle has an expansion ratio of 100 : 1 with a thrust diameter of 0.023 inch and a 15 degree half angle. The decomposition chamber is surrounded by a gold-plated thermal shield to provide low emittance. The rocket engine assembly (REA) has two catalyst bed heaters connected in parallel for redundancy. A platimum resistance-type temperature sensor is used to monitor the catalyst bed temperature.

The compliance of the LCSO 0.2 lbf thruster with the JPL specification requirements is shown on Table 20. This thruster is still under-going qualification testing and when completed should have no problem meeting the requirements of the MMS.

7.4 SUMMARY

Review of the LCSO components, namely, the bistable latch valve, the fill and drain valve, the pressure transducer, the filter, and the 0.2 lbf thruster, indicates that all except the bistable latch valve and filter are well within the requirements of the MMS. The standardized bistable latch valve appears marginal in that it may not be able to handle a flow rate greater than 0.043 pps without a potentially unacceptable pressure drop across the valve. A demand flow rate greater than 0.043 is likely if the thrust level of the nominally rated 5.0 lbf thruster is higher or the corresponding specific impulse of the thruster is lower than it is now expected. The LCSO filter is unacceptable because of its high pressure drop characteristic. A more suitable filter is the qualified GPS design. To assist in the final component selection, a cost tradeoff of the LCSO components along with other candidate components has been conducted and the results are presented in a separate appendix.

Table 20. 0.2-lbf T/VA Specification ES509778 Functional Compliance Status

Item	Requirement	Design Capability	Remarks
T/VA			
JPL Dwg. 10071189	Provide pulse mode and steady-state thrust over feed pressure ranges of 70 to 420 psia and propellant temperatures of 40 to 140°F	Complies	
Propellant	MIL-P-26536C Amendment 1 or STM-N020	Partial compliance complies	250 to 350°F limit cycle pulse shape degradation No pulse shape degradation with STM-N020
Steady State Performance			
Thrust	0.18- to 0.22-lbf at 350 psia, 28 vdc, 70°F, and vacuum	Complies	Nominal breadboard and development thrust = 0.212 lbf
Thrust reproducibility	$3\sigma = \pm 5\%$ at 350 psia and 150 psia, 28 vdc, 70°F, and vacuum	Partial compliance	Measured ±5% at 350 psia, ±6 4% at 150 psia
Specific impulse	220-lbf-sec/lbm min. @ 350 210-lbf-sec/lbm min. @ 150	Complies	Measured minimum = 221 lbf-sec/lbm Measured minimum = 212 lbf-sec/lbm
Total impulse predictability	±5% for total impulse and specific impulse in excess of 2 seconds	Complies	
Roughness	$3\sigma = \pm 30\%$ from 150 psia to 350 psia, period = 5 sec	Noncompliance	Measured maximum = 51%. Recommend increasing requirement
Response	30 msec to 10% P_c @ 500 msec 80 msec to 90% P_c @ 500 msec 120 msec to 10% P_c (tailoff)	Noncompliance	Measured maximums: 43 msec to 10% (rise), 121 msec to 90% (rise), and 259 msec to 10% (decay) Recommend increasing requirements.

Table 20. 0.2-lbf T/VA Specification ES509778 Functional Compliance Status (Cont)

Item	Requirement	Design Capability	Remarks
Pulse Mode Performance			
Minimum pulse width	T/VA operational for 0 008 sec on-times	Complies	Verified during proposal testing; to be verified during development and TA
Minimum off time	T/VA capable of operating with 0 012 sec off times	Complies	Verified during proposal tests
Minimum impulse bit	Minimum impulse bit = 0 003 lbf-sec at 350 psia pulse width = 0 008	Complies	Verified during proposal tests, to be verified during development and TA
Impulse bit repeatability	±15% from 150 to 350 psia TBD from 70 to 150 psia and 350 to 400 psia	Complies, $3\sigma = \pm 6.7\%$	±6.7% 3σ measured during proposal tests ±2.8% maximum measured for GPS duty cycles
	±25% for variable temperature environment	Predicted compliance of ±15.7%	Verified during development
Centroid repeatability Pulse width [illegible]40 msec pulse off time [illegible]400 msec	Table II		Verified during development. Breadboard and development ATP data indicate compliance
Response	30 msec to 10% P_c 80 msec to 90% P_c TBD msec to 10% P_c (tailoff) Pressure ranges of 150 to 380 psia	Complies	ATP measurements 22 ms (27 max) to 10% P_c 48 9 ms (77 max) to 90% P_c 167 ms (401 max) to 10% P_c tailoff
Minimum specific impulse	100-lbf-sec/lbm Pulse widths ≥10 msec	Complies	Measured minimum = 105 lbf-sec/lbm during development and breadborad ATP
Vacuum duty cycle	Meet the requirements of specification when performing any combination of duty cycles as typified by the two mission sequences of Table III	Predicted compliance	Verified during extensive proposal testing (22 hrs steady state and 379,329 pulses) To be verified during development and TA testing
Hot restarts	T/VA operational under worse case heat soak back	Complies	Verified during proposal testing Mount temp - 170°F, Prop temp Initial = 180°F

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