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FINAL REPORT – PHASE II

PROPELLANT SELECTION FOR UNMANNED SPACECRAFT PROPULSION SYSTEMS

CONTRACT NASw-1644

VOLUME III

STUDY OF PROPULSION STAGE COMMONALITY
AND ATTITUDE CONTROL SYSTEMS REQUIREMENTS

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
WASHINGTON, D.C.

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Lockheed

MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

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FOREWORD

This report was prepared by the Lockheed Missiles and Space Company, Sunnyvale, California, and contains the results of a study performed for the National Aeronautics and Space Administration Office of Advanced Research and Technology, under Phase II of Contract NASw-1644, Propellant Selection for Unmanned Spacecraft Propulsion Systems. The report is printed in three volumes:

Volume I	Results, Conclusions, and Recommendations
Volume II	Analysis of Propellant Sensitivity, Secondary Propulsion, and Ground Operations
Volume III	Study of Propulsion Stage Commonality and Attitude Control Systems Requirements

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INTRODUCTION

The Phase II study was divided into five major tasks plus reporting (Task V). Task I covered analysis of the sensitivity of propellants to various system perturbations. Task II entailed the comparison of using a secondary versus a primary propulsion system for minor ΔV requirements. In Task III the ground operational requirements and problems of the candidate propellants were studied. Task IV was the investigation of the feasibility of using a common stage, with minimum modification, for alternate space missions. Task VI was the identification of attitude control system requirements for the missions and configurations considered in Task IV. This volume presents the results of Tasks IV and VI. Tasks I, II, and III are discussed in Volume II.

Section 1
PROPULSION STAGE COMMONALITY

1.1 INTRODUCTION

The purpose of the propulsion stage commonality task was to: (1) investigate the feasibility of utilizing a common stage, with minimum modification, for alternate space missions and (2) to assess the sensitivity of commonality objectives to propellant combination selected. In the analysis the basic propulsion stage structure, propellant tanks, and engine systems were designed for a Mars orbiter mission. For other missions the basic stage remained essentially fixed, with insulation, surface coatings, pressurization system and meteoroid protection varied to suit the mission requirements. Propellant tank sizes were fixed by the Mars orbiter mission and propellant loading was varied for other missions if this improved performance.

The baseline propulsion stage was designed for orbit injection at Mars of a Mars orbiter sized for interplanetary injection by a Titan IIID/Centaur launch vehicle. The mission description and propulsive maneuvers required are as follows:

- 1973 Mars Orbiter/Lander
- 205-day duration with 195-day interplanetary trip, and orbit trim after 10 days in orbit about Mars
- 6,950 ft/sec. total velocity required of the stage
- Parking orbit ascent mode to 100 nm with up to 90 min. in earth orbit

Four propulsive steps are:

- First midcourse = 50 ft/sec @ T = 3 days
- Second midcourse = 17 ft/sec @ T = 165 days
- Orbit insertion = 6555 ft/sec @ T = 195 days
- Orbit trim = 328 ft/sec @ T = 205 days

Three propellant combinations were considered in detail, including one cryogen (F_2/H_2, pump fed), one space-storable ($FLOX/CH_4$, pump fed), and one Earth-storable (N_2O_4/A-50, pressure fed). In addition one pressure fed space storable (OF_2/B_2H_6) stage was examined to a lesser extent. Three engine companies – Aerojet-General, Pratt and Whitney Aircraft, and Rocketdyne provided technical data for the propellants and engine systems used in this task.

The basic study approach for the task is shown in Fig. 1. This approach was taken in order to develop and evaluate a commonality stage in an iterative manner. A propulsion module was first defined and its propulsion requirements determined for the Mars Orbiter mission. This was accomplished by utilizing the Mars Orbiter spacecraft and optimized pump-fed propulsion stage configurations issuing from contract NASw-1644 Phase I and scaling them to match the capability of a Titan IIID/Centaur launch vehicle to perform the nominal Mars Orbiter mission. This included the configurations utilizing F_2/H_2, $FLOX/CH_4$, OF_2/B_2H_6 and N_2O_4/A-50 propellants. The stage was also resized to adapt to the 10-foot diameter Centaur. Payload weight and dimensions were scaled down accordingly, but no attempt was made to reconfigure the spacecraft or to analyze spacecraft details such as capsule payload requirements vs aerodynamic drag parameter.

An analysis was then made to determine the feasibility of using the commonality stage to perform the following missions when launched from Earth by the Titan IIID/Centaur:

(a) Mars Orbiter-Orbit Injection Stage (Baseline)
(b) Venus Orbiter-Orbit Injection Stage
(c) Lunar Cargo Delivery-Orbiter/Lander Stage
(d) Jupiter Flyby-Earth Escape Stage
(e) Solar Probe to 0.20 AU-Earth Escape Stage
(f) Mars Orbiter – stage sized to Titan IIID/Centaur but used on Titan IIID with ascent burn
(g) Mars Orbiter – ascent burn and orbit inject stage optimized for Titan IIID/Centaur
(h) Jupiter Orbiter – orbit injection stage used first at Earth in ascent burn mode

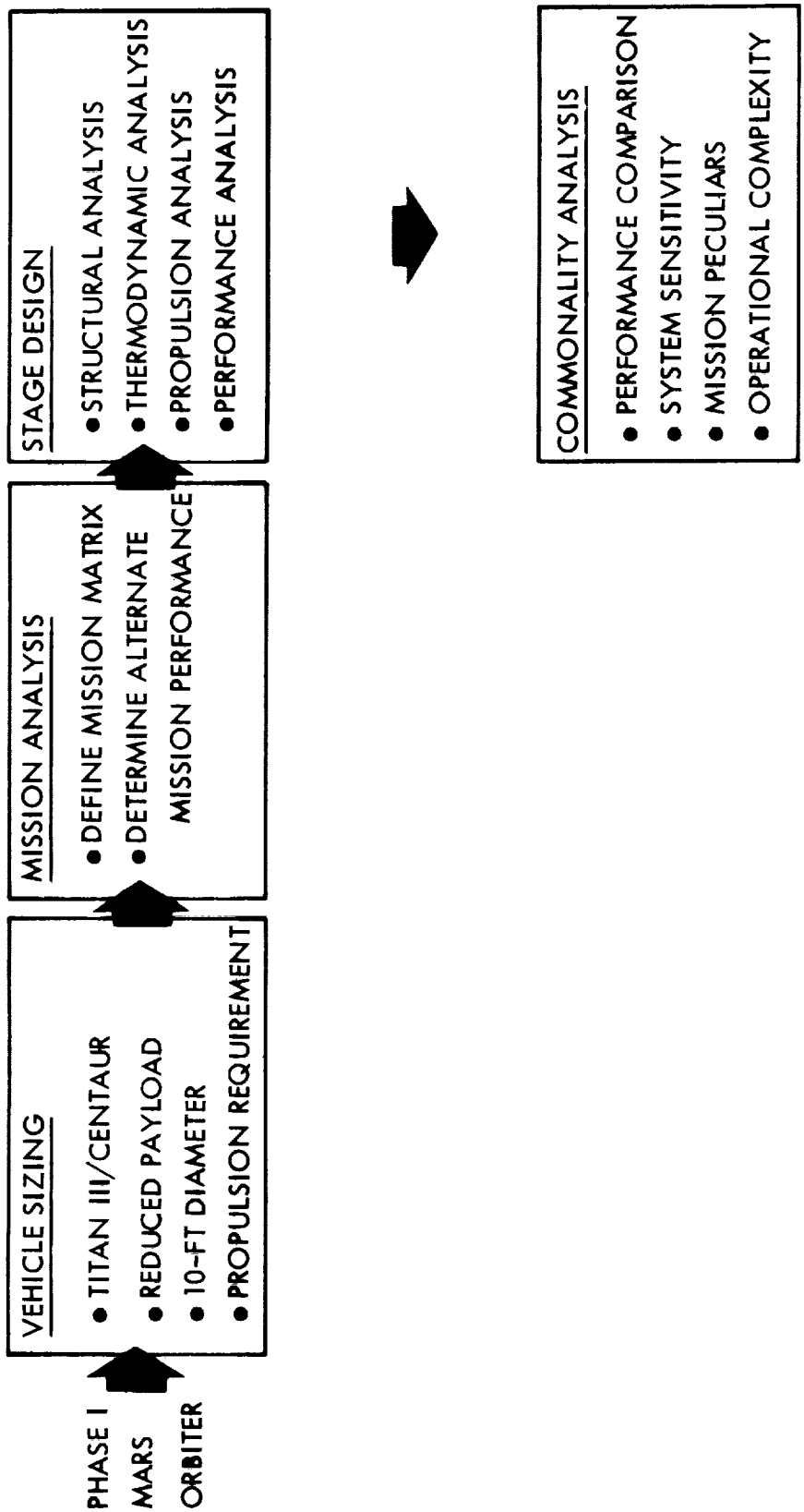


Fig. 1 Study Approach for Commonality Stage

The characteristics of the selected missions were defined and payload capabilities of the commonality stage in performing these missions was estimated.

Having established that the commonality stage payloads were potentially very attractive, the commonality stage design was initiated. The basic structural, propulsion, and thermodynamic characteristics were defined in detail and an optimization procedure was carried out to completely describe the design parameters and performance characteristics of the Mars orbiter vehicle. This module was then flown on the other selected missions and a new optimization computed allowing the insulation thickness, pressurization requirements, and ullage volume to vary. Preferential spacecraft orientation and shadow shielding were used where advantageous. Modifications required to the propulsion stage structure, insulation, propellant loadings, etc., were identified and mission performance recalculated to determine a refined payload capability for each stage and mission combination. Concurrently, optimum sizing for stages used in the ascent-burn mode at Earth departure prior to use for midcourse corrections and for orbit insertion at Mars were determined. Payload capability in Mars orbit for a stage sized to Titan IIID/Centaur, but launched by Titan IIID and used in the ascent burn mode, was also determined.

Finally, the commonality characteristics were evaluated and the commonality concept analyzed. This was accomplished by making a detailed comparison of the relative performance, operational, and complexity/simplicity advantages of each propellant combination, in comparison with the remaining propellants, for each mission.

1.2 INITIAL STAGE SIZING

The first commonality subtask was to size the Mars orbiter spacecraft and propulsion stage to the capability of the Titan IIID/Centaur. It was also necessary to develop scaling laws and to define requirements for use in later detailed thermal, structural, design, and performance optimization analyses. The ground rule for stage sizing was that the Mars orbiter of Phase I be scaled to fit the dimensions and performance capability of the launch vehicle of Phase II. In Phase I the propulsion module was

sized so that two Voyager spacecraft could be launched on a single Saturn V booster. In Phase II, a single spacecraft launched by the Titan IIID/Centaur was assumed, and a size and weight reduction for the orbiter and propulsion module was therefore required.

The stage sizing was accomplished by optimizing the Mars orbiter vehicle for each propellant combination and within the Titan IIID/Centaur booster capability shown in Fig. 2. This figure indicates that a payload of 9700 pounds can be injected to the Mars transfer velocity of 38,540 feet per second. This 9700 pounds encompasses the payload, spacecraft, and propulsion module. Only the propulsion module design was analyzed in detail while the spacecraft, science, capsule, etc. were lumped as useful payload.

The first parameter examined in the analysis was the engine thrust/weight (T/W) effect for each propellant and vehicle combination. By NASA ground rule the F_2/H_2 and $FLOX/CH_4$ systems utilized pump-fed engines and the OF_2/B_2H_6 and N_2O_4/A-50 systems utilized pressure-fed engine. The vehicle payload was computed for various T/W values with normalized values of inert weight fraction (γ'). Specific Impulse (I_{sp}), Mars arrival energy, and stage velocity (ΔV) were fixed. These data were then plotted and are presented in Figs. 3 and 4. The plots include several values of I_{sp} in order to cover several propellant combinations and to determine the gravity loss effect of I_{sp}.

Figure 3 shows that the maximum payload values for the pump-fed systems were fairly independent of the thrust/weight parameter for values of thrust/weight above 0.45 to 0.50. The pressure-fed systems were insensitive to thrust/weight values above 0.35 to 0.40, as seen in Fig. 4. From these plots a thrust value of 5,000 pounds were selected for the pump-fed systems and 3500 pounds for the pressure-fed systems, with throttling to 500 pounds assumed as a requirement. The pressure-fed engine thrust was selected on the low end of the range in order to minimize size.

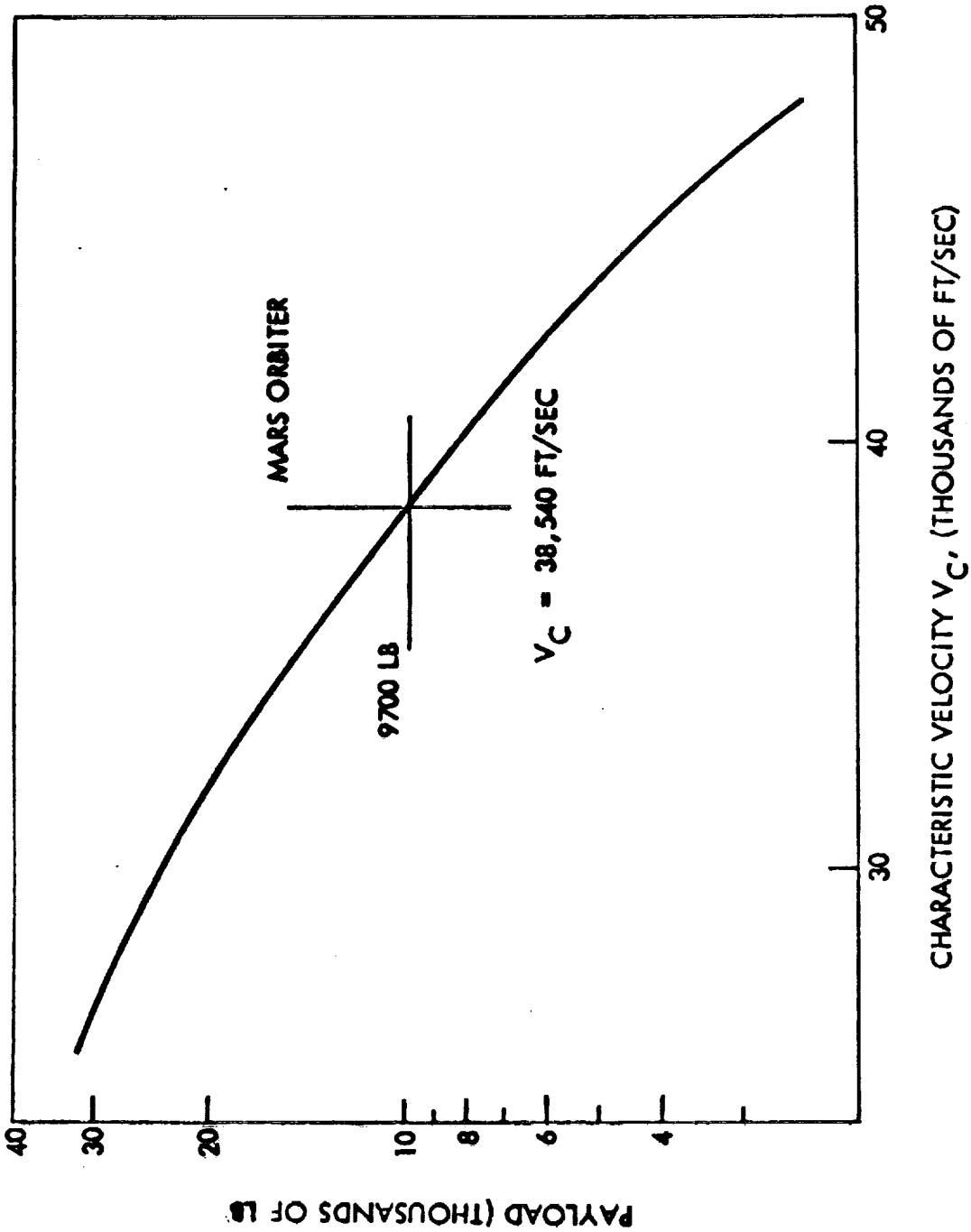


Fig. 2 Titan IIID/Centaur Performance Capability

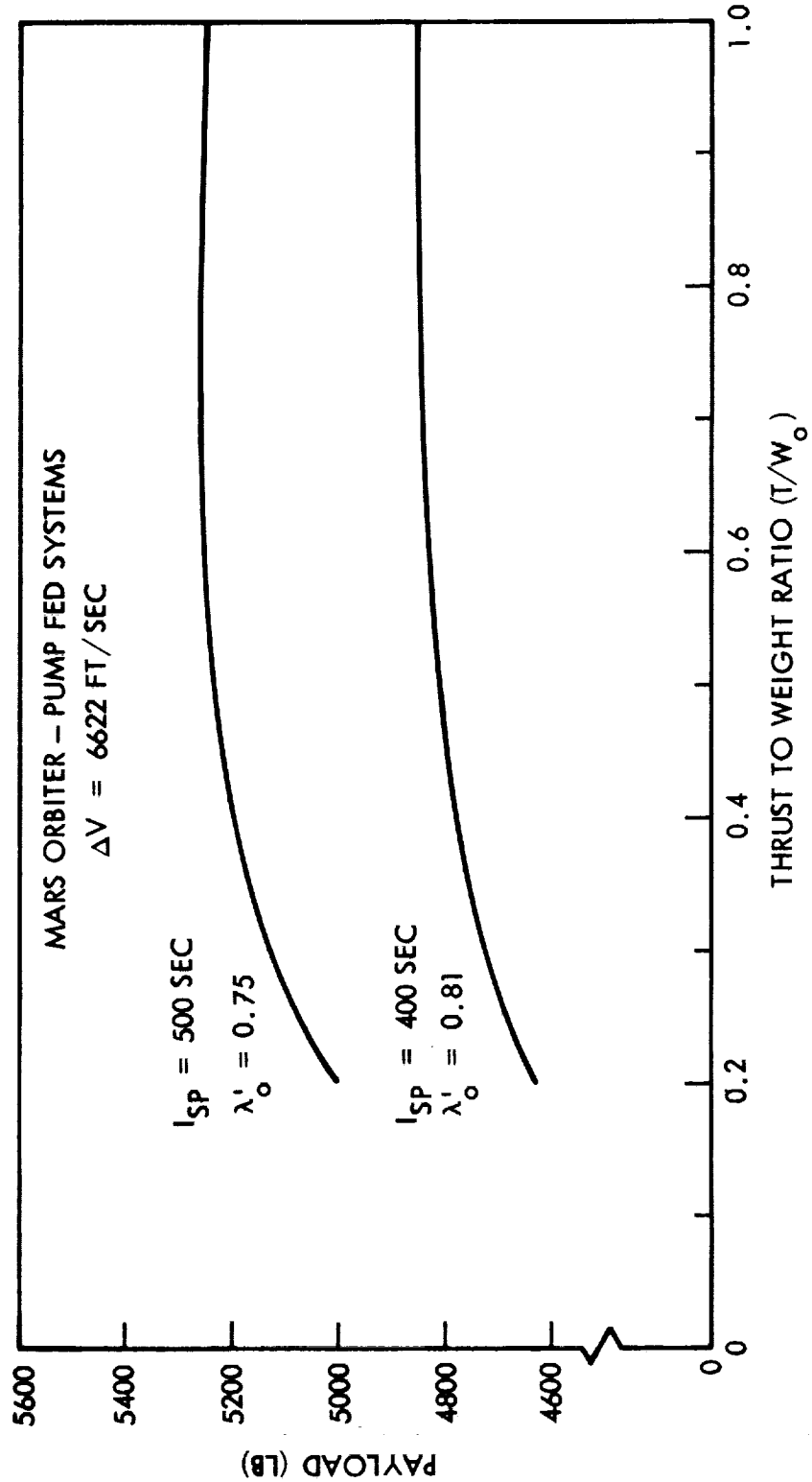


Fig. 3 Thrust-to-Weight Effects for Pump-Fed Systems

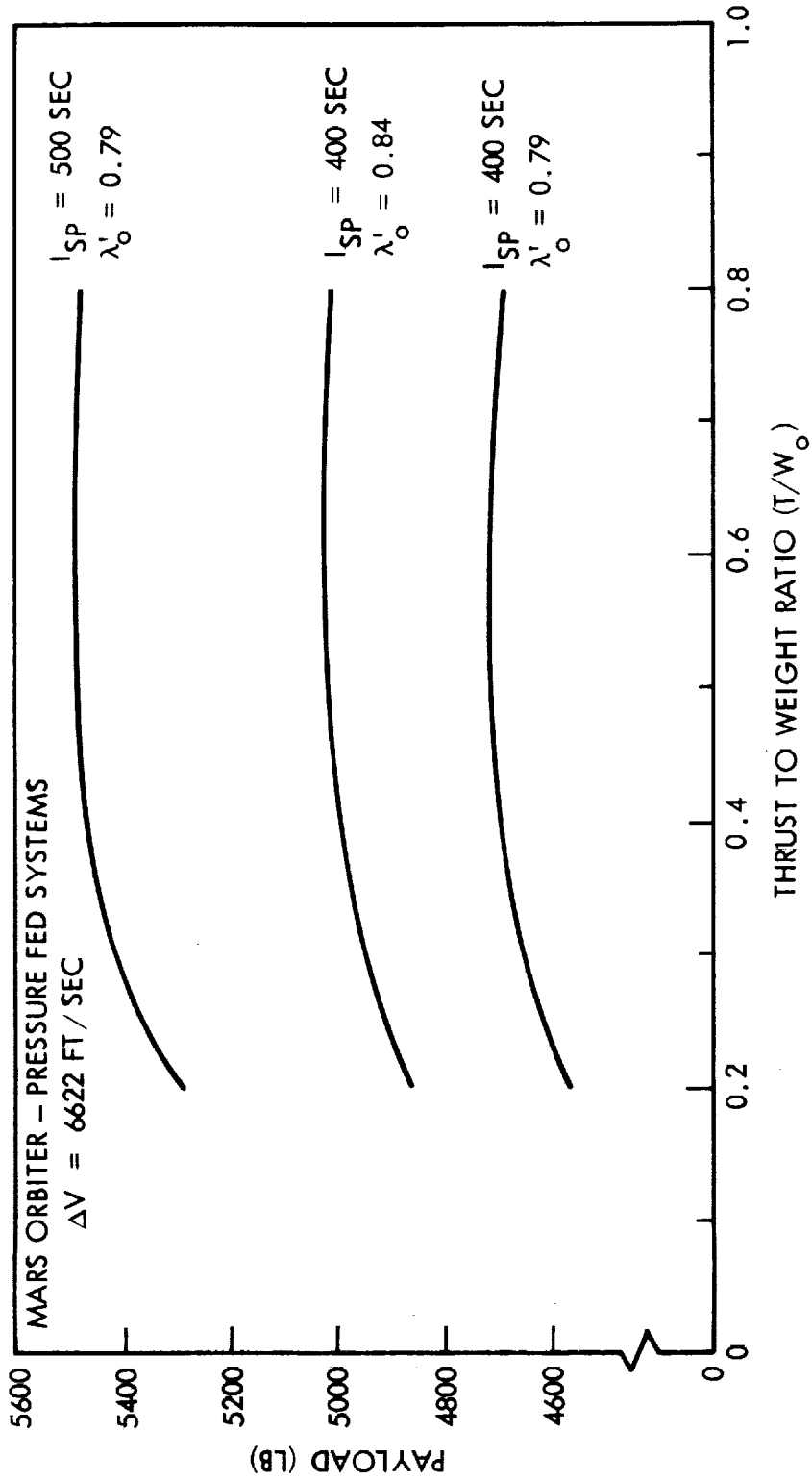


Fig. 4 Thrust-to-Weight Effects for Pressure-Fed Systems

Engine data were solicited from the engine manufacturers for the chosen engine sizes and types, while for interim use, engine data from Phase I were scaled down for use in a preliminary assessment of the general performance level of the new stage. The scaled down data are shown in Table 1 along with the favored vehicle orientation for each propellant combination.

Table 1

VEHICLE ASSUMPTIONS FOR PRELIMINARY SIZING OF A COMMONALITY STAGE

Propellant	Feed Type	Thrust (lb)	Mixture Ratio (O/F)	Isp (sec)	Orientation
F_2/H_2	Pump	5,000	13/1	465	Sun on Capsule
$FLOX/CH_4$	Pump	5,000	5.75/1	406	Sun on Capsule
OF_2/B_2H_6	Press.	3,500	3.82/1	412	Sun on Capsule
N_2O_4/A-50	Press.	3,500	2/1	316	Sun on Tank

Subsequent to the selection of preliminary propulsion characteristics, stage concepts were developed based on Phase I designs. Configurations using four spherical propellant tanks were assumed for all except F_2/H_2. For the F_2/H_2 system one arrangement with four spherical tanks, and a second arrangement with an ellipsoidal hydrogen tank and two spherical fluorine tanks were evaluated. The system with the ellipsoidal tank had the lowest stage inert weight and was consequently selected. All tanks were designed with 2021 aluminum and were supported by laminated fiberglass struts. The basic vehicle structure was also designed with an aluminum space frame. Each tank was individually insulated with multilayer double-aluminized mylar and Tissuglas spacers and protected from meteoroids by a dual-wall aluminum shield. Pressurization spheres were assumed stored inside the propellant tanks for cryogens and space storables, but could be stored externally if enclosed within the main tank insulation. No thermodynamic analysis was conducted for this initial stage sizing but design data adapted from the Phase I results were utilized. Preliminary drawings were made, but are not presented since they are essentially unrefined versions of the final layouts shown later in Section 1.5.

Preliminary weights were generated for the alternate concepts for use later in detailed performance analyses. These preliminary weights and payload capabilities for the Mars orbiter are shown in Table 2.

Table 2

PRELIMINARY WEIGHTS AND PAYLOAD CAPABILITIES FOR A MARS ORBITER

Propellant	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Injected Weight (lb)	9700	9700	9700	9700
Propellant Load (lb)	3609	4008	3963	4704
Propulsion Module Inert Weight (lb)	918	869	1002	1030
Payload to Mars Orbit (lb)	5173	4823	4735	3966

These results showed that using the new criteria and fixed initial weight yielded performance characteristics, on a propellant comparison basis, very similar to that computed in Phase I. The F_2/H_2 system had the best performance, followed closely by FLOX/CH_4 and OF_2/B_2H_6. The N_2O_4/A-50 system payload was significantly lower than the others.

The Mars orbiter propulsion module defined above for each propellant combination was then evaluated to see how well it could perform the broad spectrum of space missions of interest.

1.3 MISSION AND STAGE REQUIREMENTS DEFINITION

With the preliminary definition of a Mars Orbiter propulsion module completed, and attractive performance promised for the specified alternate missions, the next step was to define these missions in greater detail and to state the requirements each mission placed on the commonality stage. The missions, a Mars Orbiter/Lander,

Lunar Cargo, Venus Orbiter, Jupiter Orbiter, Jupiter Flyby, Solar Probe (0.2 AU) are summarized in Table 3 and described in the following paragraphs.

Table 3

CHARACTERISTICS OF ALTERNATE MISSIONS

Mission	Launch Year	Injection Velocity Departing Earth (ft/sec)	Time to Final Burn (days)	Stage ΔV After Injection at Earth (ft/sec)	Number of Burns Expected
Mars Orbiter	1973	38,540	205	6,950	4 or 5
Lunar Cargo	1975	36,027	3	9,006	4
Venus Orbiter	1976	38,339	170	7,200	3
Jupiter Orbiter	1980	47,361	900	6,726	4
Jupiter Flyby	1975	46,600	hours	0	1
Solar Probe (0.2 AU)	1977	55,000	hours	0	1
*Grand Tour of Jovian Planets	1977	47,900	hours	0	1
*Earth Synchronous Equatorial Orbiter	1975	33,660	hours	5,991	1

*These missions were added to the matrix specified, but no detailed description or analyses are provided.

1.3.1 Mars Orbiter Mission

The Mars Orbiter Mission, described earlier, is designed to obtain data from orbit about Mars and from a capsule landed on the Martian surface. The spacecraft includes an orbiter and a lander, both of which are placed in orbit about Mars by the commonality stage. The mission description and propulsive maneuvers required of a

spacecraft launched by Titan IIID/Centaur to an injection velocity of 38,540 ft/sec are as follows:

- Mission year 1973
- 205-day duration with 195-day interplanetary trip, and orbit trim after 10 days in orbit about Mars
- 6,950 ft/sec total velocity required of the stage

Four propulsion steps are:

- First midcourse = 50 ft/sec @ T = 3 days from launch
- Second midcourse = 17 ft/sec @ T = 165 days
- Orbit insertion = 6555 ft/sec @ T = 195 days
- Orbit trim = 328 ft/sec @ T = 205 days

An additional use for the commonality stage for a Mars Orbiter Mission is in assisting the launch vehicle to achieve the required 38,540 ft/sec injection velocity at Earth. This mode, called ascent burn, was assumed for a spacecraft launched by Titan IIID without Centaur. It was also used for a spacecraft launched by Titan IIID/Centaur, but with increased propellant loading over that required for post-injection burns.

1.3.2 Lunar Cargo Mission

The Lunar Cargo mission is primarily a resupply lander to support manned activities on the lunar surface. It could be initiated as soon as a high energy upper stage is developed, probably 1975 at the earliest. The basic mission profile would include a launch with the Titan IIID/Centaur to an injection velocity of 36,027 ft/sec. The propulsion module would first be utilized for midcourse maneuvers with the engine throttled to 500 pounds thrust. The first midcourse maneuver is scheduled for 38 hours with a ΔV of 40 feet per second, and the second maneuver at 68 hours with a ΔV of 28 feet per second. The primary maneuver consists of two parts, orbit acquisition and descent. The orbit acquisition occurs at 74 hours and requires a ΔV of 3195 feet per second, and the landing maneuver occurs at 81 hours and requires a ΔV of 5811 feet per second, and continuous throttling capability.

1.3.3 Venus Orbiter Mission

The objective of the Venus Orbiter mission is to obtain detailed information about the atmosphere of Venus. The spacecraft includes both an orbiter and a lander. The lander is considered as part of the payload in this study. The mission opportunity considered in this study was for the year 1976. The propulsion requirements vary from opportunity to opportunity but 1976 is typical and a likely initial flight. The booster is again the Titan IIID/Centaur, providing an injection velocity of 38,339 ft/sec. The propulsion module would be used only for midcourse maneuvers and orbit injection. The first midcourse correction occurs after 7 days and requires a ΔV of 50 feet per second, and the second midcourse correction occurs after 155 days with a ΔV of 17 feet per second. Both midcourse corrections are conducted with the engine throttled to 500 pounds thrust. The primary maneuver consists of the orbit injection maneuver requiring a ΔV of 7200 feet per second. This provides an elliptical orbit with good observational capabilities. No orbit trim maneuver is specified.

1.3.4 Jupiter Orbiter Mission

The objective of the Jupiter Orbiter mission is to obtain initial detailed data describing the atmosphere and magnetic field of Jupiter. The initial flight for such a mission is postulated for 1980 and all mission requirements are based on that opportunity. Other opportunities would have slightly different requirements. The booster is again the Titan IIID/Centaur and a combination of launch vehicle plus commonality stage burns provides an injection velocity of 47,361 ft/sec departing Earth. This use of the spacecraft stage in the launch stage is called ascent burn. The ΔV contribution of the propulsion module to the ascent burn is a variable that is affected primarily by the propellant selected in the propulsion module. There are three midcourse corrections which are accomplished with a throttled engine. They occur after 7, 100, and 850 days, and require a ΔV of 50, 17, and 17 feet per second, respectively. The orbit injection maneuver of Jupiter is the final propulsive step conducted at full thrust to provide a ΔV of 6642 ft/sec. This final burn takes place 900 days after launch from Earth.

1.3.5 Jupiter Flyby Mission

The objective of the Jupiter Flyby mission is to initiate exploration of interplanetary space beyond Mars and to obtain data in the vicinity of Jupiter. This flyby mission is typical of a broad range of direct ascent missions in which the propulsion module is the last stage in the ascent stack and the mission energy requirements dictate the mission payload. The first opportunity for this mission is likely to be 1975, and the injection velocity departing Earth is 46,600 ft/sec. The commonality stage is discarded after this one burn.

1.3.6 Solar Probe (0.2 A.U.) Mission

The objective of the Solar Probe mission is to obtain in situ measurements of the solar environment. This is also a direct ascent mission in which the propulsion module becomes the last stage in the ascent stack. It is used to achieve the required injection velocity of 55,000 ft/sec. Following this one burn the stage is discarded.

1.4 PRELIMINARY PERFORMANCE ESTIMATES FOR ALTERNATE MISSIONS

In the initial stage sizing analysis a propulsion stage for the Mars Orbiter was defined and the weights shown in Table 4 for inerts and propellant were calculated.

Table 4

PRELIMINARY STAGE WEIGHTS

Propellant	Inert Weight (lb)	Propellant Weight (lb)
F_2/H_2	918	3610
$FLOX/CH_4$	869	4008
OF_2/B_2H_6	1002	3963
N_2O_4/A-50	1030	4705

This stage, described in Section 1.2, was then flown on the missions defined in Section 1.3 and the mission payloads calculated. The stage weights together with specific impulse values previously listed in Table 1 were used to calculate the alternate mission performance capabilities without specific thermodynamic analyses. The payloads obtained are presented in Table 5. For the Jupiter Orbiter mission shown, two commonality stages were assumed used. One stage, fully tanked, was used to increase the booster injection capability, and the other stage was used to perform the post-injection maneuvers. The second stage was off-loaded, with a propellant load of 42 percent for the F_2/H_2 system decreasing to 30 percent for the N_2O_4/A-50 system. In the detailed analysis performed later it was found that the Jupiter Orbiter performance could be improved by using only one commonality stage for both ascent burn and orbit injection.

The significant conclusions from this preliminary analysis were that: (1) all of the missions considered could be performed by a common stage designed to the Mars Orbiter requirements, and (2) a detailed thermal/structural/design analysis should be conducted for this common stage to refine the design, confirm the preliminary findings, and uncover any major problems. This detailed analysis is discussed in the following sections.

1.5 DETAILED STAGE DESIGN AND ANALYSIS

A detailed design analysis of the commonality stage was conducted with F_2/H_2, FLOX/CH_4, OF_2/B_2H_6, and N_2O_4/A-50 propellants using the Mars orbiter mission as the basis for a design optimization. The preliminary configuration layouts were refined and structural and weight analyses conducted. Engine data supplied for this task by the supporting rocket engine companies was evaluated and engine performance, weight, and design conditions were selected for the analysis. A detailed thermodynamic analysis was made in which all configurations were modeled, energy balances determined, and a system optimization conducted for each mission and propellant combination. The performance of each propulsion module for each mission and propellant combination was then determined and final designs including weights of each of the elements of the propulsion module, were established.

Table 5

PRELIMINARY PAYLOADS

Propellant	Mars Orbiter	Jupiter Flyby	Solar Probe (0.2 AU)	Grand Tour of Jovian Planets	Syn. Eq. Earth Orbiter	Lunar Cargo	Venus Orbiter	Jupiter Orbiter (Dual Stage)
F_2/H_2	5173	4550	1700	3950	6411	3454	4935	1767
$FLOX/CH_4$	4823	4240	1460	3640	6018	3168	4595	1463
OF_2/B_2H_6	4735	4150	1350	3550	5935	3073	4500	1280
N_2O_4/A-50	3966	3390	820	2800	5077	2429	3740	582

The designs and analyses conducted are defined in the following paragraphs.

1.5.1 Performance of an Ascent Burn Stage

A requirement for ascent burn capability for the commonality stage was not identified until the task analysis was well under way, thus no preliminary screening analysis was conducted. It is therefore necessary to insert at this point a discussion of the impact of introducing a stage with ascent burn capability at Earth. This stage must assist the launch vehicle in achieving the required interplanetary injection velocity when departing Earth, then continue on the mission to provide midcourse corrections, orbit insertion at the planet, and orbit trim maneuvers.

There were two basic cases to be considered, a Mars orbiter and a Jupiter orbiter. It also appeared that a Mars orbiter launched by the Titan IIID booster without Centaur was a very interesting combination if the commonality stage could assist in the Earth departure phase. Figure 5 shows the payload capability for these three missions as a function of commonality stage propellant load. This figure identifies the F_2/H_2 performance which is typical of all the cases. The first point to consider is the no-ascent-burn baseline Mars orbiter indicated by a circle on Fig. 5. This system, with approximately 3600 pounds of propellant, yields a payload of over 5000 pounds. As the propellant is increased for the Mars mission the ΔV imparted by this stage during the ascent burn increases, but the payload is relatively flat and finally decreases for propellant loads above 6000 pounds. Considering that this stage could also be used for a lunar lander and that for the lunar mission a propellant load larger than for the Mars orbiter baseline is desirable, a 6000-pound propellant load system was selected as one point for analysis of the ascent mode. This analysis was restricted to pump-fed F_2/H_2 and FLOX/CH_4 systems.

Figure 5 also indicates that the baseline commonality stage has a propellant load that falls almost exactly on the maximum payload point for an ascent burn Jupiter orbiter. This basic stage was analyzed both for a Jupiter Orbiter and for the ascent burn Mars

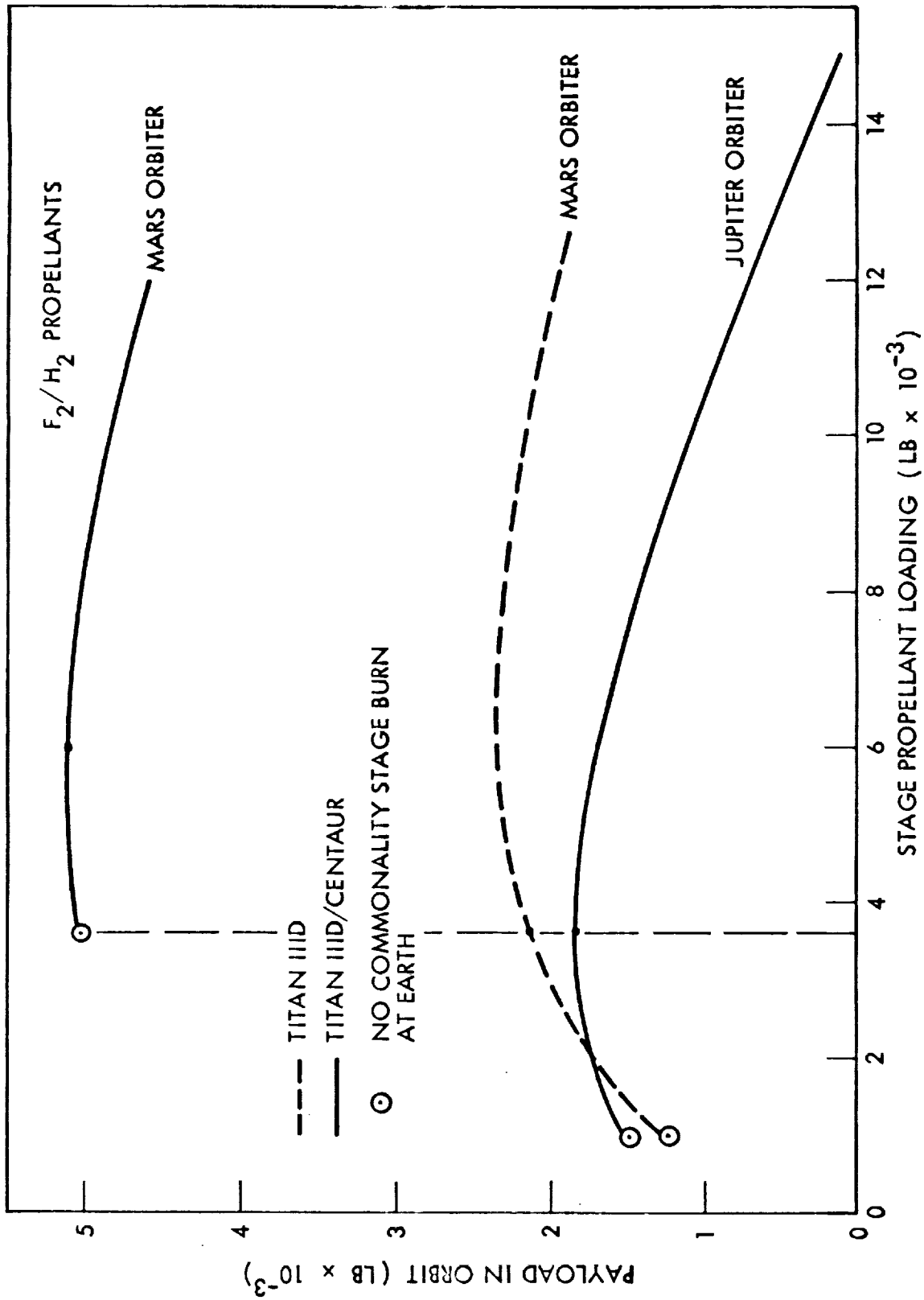


Fig. 5 Commonality Stage Payload Vs Propellant Loading

mission launched by Titan IIID without Centaur. The desirability of this mode for the later mission is that a payload of more than 2000 pounds could be delivered and the cost of the Centaur could be saved for early missions.

Design details and performance figures for ascent burn stages are discussed in the following sections in combination with the basic systems.

1.5.2 Stage Design

The detailed stage design consisted of developing concepts, conducting a structural analysis, determining parametric system weights, and defining a propulsion module. The interrelationships of the various elements of the stage design are shown in Fig. 6 where Family A represents the baseline stage and Family B the larger stage with 6000 lb of propellant. The propulsion analysis is independent of this and is not iterative since the design data were based on the earlier stage sizing analysis. The basic elements of the design analysis (structural, thermodynamic, propulsion, and performance analysis) are described in the following sections.

1.5.2.1 Basic Data. The stage design analyses were performed using the following basic data and assumptions.

Mission Data. The missions evaluated are those described in Sections 1.3. The Mars Orbiter is the baseline mission for which the commonality stage is sized. This stage is then flown on the remaining missions with minor changes to insulation, meteoroid shielding, and pressurization gas volume.

Propulsion System Data. The engine system requirements were based on the Mars Orbiter mission profile, duty cycle, vehicle size, and thrust-to-weight analysis. Although no specific emphasis was given to ascent burn for the thrust level determination, the thrust level selected in the analysis is adequate for ascent burn also. The basic engine was assumed to handle all the secondary propulsion requirements such as

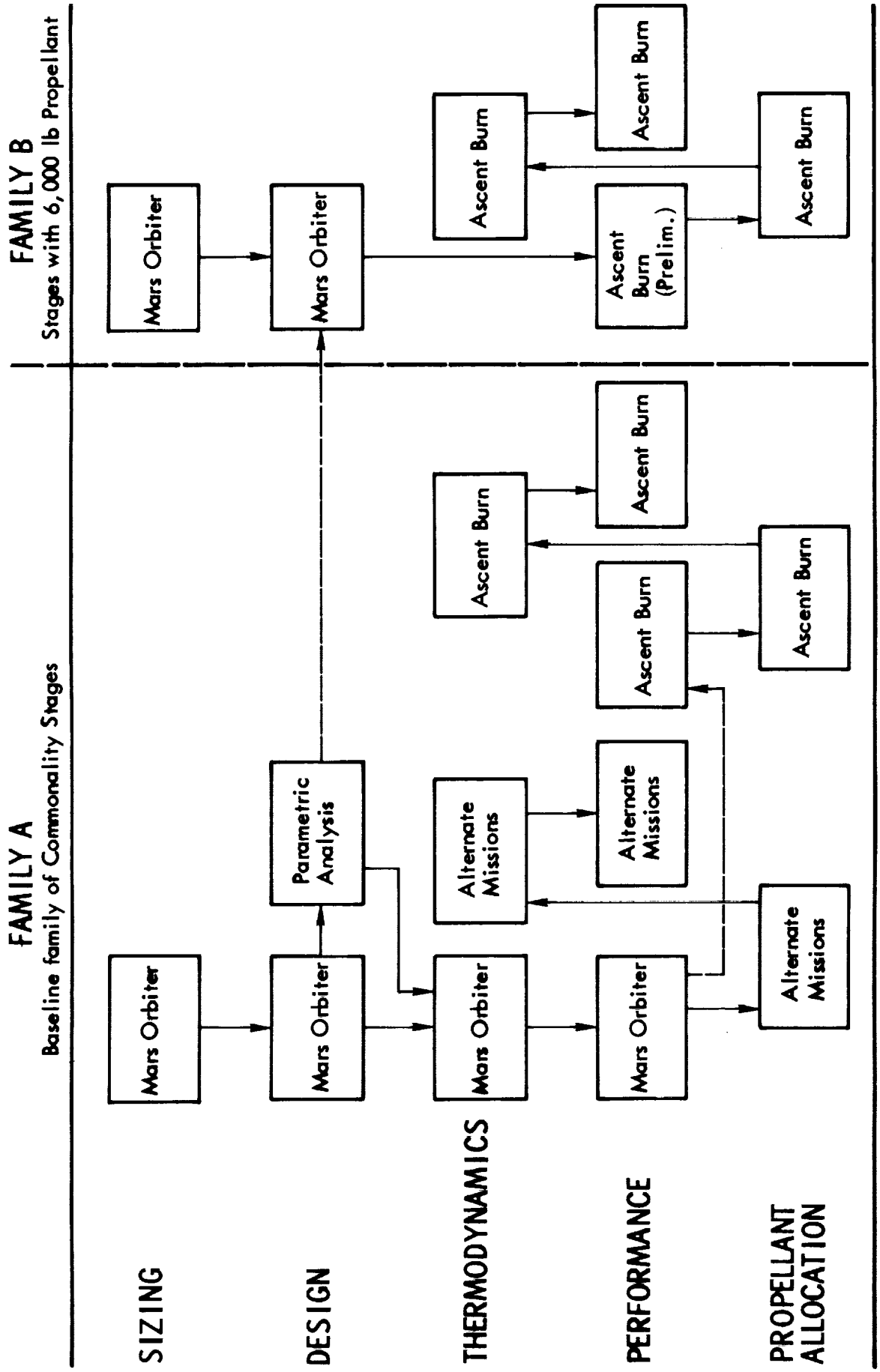


Fig. 6 Analysis Sequence for Commonality Stage

midcourse and orbit trim maneuvers by means of throttling. For F_2/H_2 and FLOX/CH_2, pump-fed regeneratively cooled engine systems were specified and for OF_2/B_2H_6 and N_2O_4/A-50, pressure-fed ablatively cooled engine systems were specified. For the two pump-fed engine systems a primary thrust level of 5000 pounds was selected and for the pressure-fed engine systems a primary thrust level of 3500 pounds was selected. The scondary thrust level of 500 pounds was chosen for all systems, to be obtained by step-throttling. Step-throttling is satisfactory for all except the Lunar Cargo mission which will require continuous throttling. For the pump-fed engine systems an expansion ratio of 100 was assumed and for the pressure-fed systems an expansion ratio of 60 was selected in order to hold the stage to a reasonable length. Still lower expansion ratios might be required for the pressure-fed system if the stage length were limited.

These engine requirements were transmitted to Aerojet-General Corporation, Pratt & Whitney Aircraft, and Rocketdyne. These engine companies provided design data, performance, weights, and dimensions based on the requirements specified by Lockheed. The weights of pressure-fed engine systems with ablative chambers are very dependent upon duty cycle and burn time. The weight data in this report is therefore based on the specific Mars Orbiter mission requirements.

The data obtained from the engine companies was analyzed and compared with the Phase I parametric data and reviewed for consistency between engine companies. The propulsion data selected for the commonality stage are presented in Table 6.

Vehicle Configuration Data. The vehicle configurations were based on the engine systems described in Table 6 and the propellant loadings and design constraints described below. The propellant loadings shown were obtained by adjusting the preliminary sizing data to account for revisions in specific impulse.

The spacecraft is a scaled down Voyager, ten feet in diameter, with payload varied as a function of propellant. The propulsion module has a single engine, a dual-wall meteoroid shield, and is thermally protected by multilayer insulation of double-aluminized mylar with Tissuglas spacers and by filament-wound fiberglass tank support struts.

Table 6

ENGINE DESIGN AND PERFORMANCE CHARACTERISTICS

Parameter	F_2/H_2	$FLOX/CH_4$	OF_2/B_2H_6	N_2O_4/A-50
Thrust, lbf	5000	5000	3500	3500
Mixture Ratio, O/F	12	5.25	3.0	1.6
Engine Feed	Pump	Pump	Pressure	Pressure
Expansion Ratio	100	100	60	60
Chamber Pressure, psia	800	800	100	100
Min. Tank Pressure, psia	20	20	155	155
Engine Weight, lb.	98	98	153	151
Engine Length, in.	43.7	44.1	62.1	63.3
Exit Diameter, in.	20.7	21.0	38.0	39.0
Primary I_{sp}, sec.	464	402	397	311
Throttled I_{sp}, sec.	428	371	362	290

The initial propellant loadings are shown in Table 7.

Table 7

INITIAL PROPELLANT LOADINGS

Propellant	Propellant Loading (lb)	
	Baseline Systems	Alternate Systems
F_2/H_2	3600	6000
$FLOX/CH_4$	4000	6000
OF_2/B_2H_6	4100	–
N_2O_4/A-50	4900	–

Booster Data. The booster specified is the Titan IIID/Centaur with an injection capability for the baseline Mars mission of 9700 lb. Generalized performance capability was shown earlier in Fig. 2.

Design Criteria. The design criteria on which the analysis is based are described below.

Load Factors

- Launch – Axial at max. acceleration: 4 g
  Axial at rebound: -1.5 g
  Lateral: ±1.5 g
- Orbit Insertion – Axial 1 g
- Orbit Trim – Axial .1 g

Design factor of safety = 1.25 to ultimate stress at zero safety margin. Check for no yield at limit which equals 1.1 times maximum applied load.

Meteoroids

- Flux model and penetration model – Ref. 1 and 2, of NASA CR 96988, Lockheed K-19-68-6, Vol. II.
- Probability of no penetrations = 0.99
- Propellant tanks are not to be used as part of the shield

Materials

- Tanks – welded aluminum 2021, room temperature allowables
- Tank supports – low-heat-leak filament-wound fiberglass
- Insulation – smooth double aluminized mylar with Tissuglas spacers – Density = 2.3 lb/ft^3

Missions

- Insulation, meteoroid shield, and pressurization system are designed for each mission
- All other hardware including propellant tanks are constant for all missions

1.5.2.2 Configuration Description. The configuration design analysis consisted of developing the design concepts, analyzing the structural elements, and developing the basic system weights.

The differing properties of the candidate propellants suggested that the vehicle design be grouped into two general categories. The low density of the H_2 in the F_2/H_2 propellant led to one design concept while properties of the remaining propellants led to a second general design as described below.

Two general propulsion stage configurations were studied for the F_2/H_2 propellant combination. These included a four tank arrangement in which the hydrogen tanks were cylindrical in shape and a three tank arrangement which incorporated a single ellipsoidal hydrogen tank and two spherical fluorine tanks. Preliminary analysis indicated that the latter design was simpler and lighter although it produced a somewhat longer stage. This latter design was chosen for the detailed analysis and is shown in Fig. 7.

The F_2/H_2 design utilizes an aluminum space frame, and an aluminum reinforced cruciform beam which supports an aluminum thrust structure. Aluminum tanks are supported by fiberglass struts with high thermal resistance. The propulsion module supports the payload and is in turn carried by the Centaur interstage. The launch vehicle shroud extends down to the interstage so that no air loads are transmitted to the propulsion stage. This design approach was followed for all vehicle concepts.

The FLOX/CH_4 vehicle, which is typical of the non-hydrogen designs, is shown in Fig. 8. This design utilizes four spherical aluminum tanks supported from a space frame consisting of a reinforced cruciform beam surrounded by an octagonally shaped frame. This arrangement allows for a maximum view of space for the propellant tanks and a direct alignment of the propulsion module with an octagonally shaped payload. Each propellant tank is individually insulated and has its own meteoroid shield. The dual-walled meteoroid shield is external to the insulation, providing a good base for appropriate surface coatings and yielding a light-weight design. A multilayer insulation blanket is mounted between the propulsion module and the payload and is

wrapped around each propellant tank to help control the thermal environment of the propellants. The insulation thicknesses and surface coatings on the tanks can be readily changed to adapt the stage to the operating conditions required for the various mission and propellant combinations. Tank support struts are also wrapped with insulation.

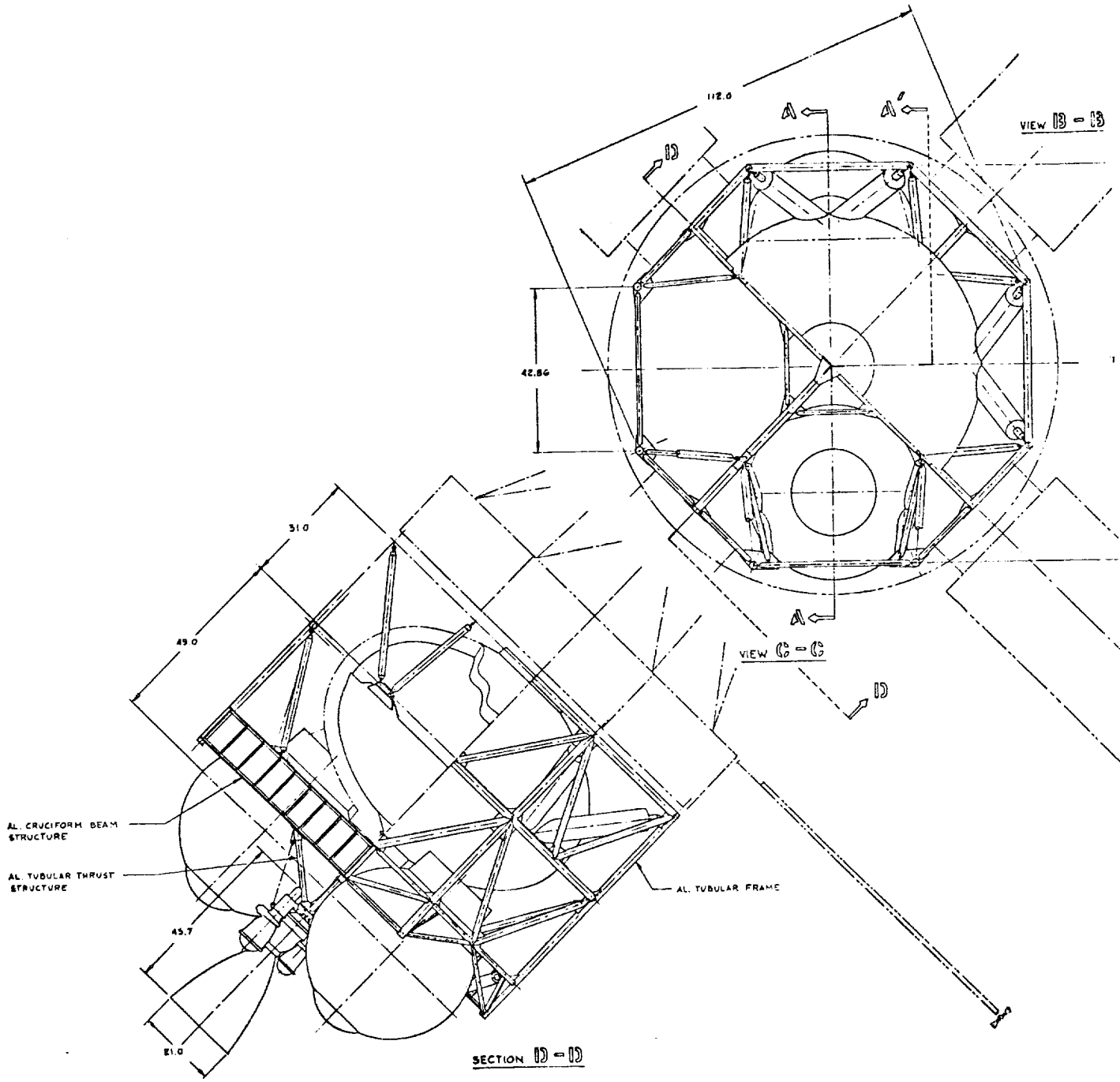
The OF_2/B_2H_6 design is essentially the same as the $FLOX/CH_4$ design concept and is shown in Fig. 9. The primary difference is that this is a pressure-fed system and the engine and pressurization tanks are larger.

The N_2O_4/A-50 design is similar to the OF_2/B_2H_6 design and is shown in Fig. 10. For these Earth storable propellants the pressurization spheres are stored outside of the propellant tanks.

The 6000-pound propellant load propulsion modules are variations of the basic design and provide a more optimum stage for use in the ascent burn mode. Figure 11 shows the 6000 lb F_2/H_2 system which incorporates a larger ellipsoidal hydrogen tank and four spherical fluorine tanks. The 6000 lb $FLOX/CH_4$ system, shown in Fig. 12 utilizes cylindrical sections in the FLOX tanks and slightly larger spherical Methane tanks.

The structural analysis included a detailed static load stress analysis of all structural elements. The analysis encompassed all mission phases and payloads. Conservative material properties were utilized throughout in order to present performance data which is attainable.

Inert weights. The propulsion module weights were divided into propellant sensitive and fixed weights. The basic structure was divided into a basic frame, tank supports, attachments, and bulkhead insulation. The basic frame and tank supports are somewhat propellant sensitive and are therefore varied with propellant load for each design concept. The attachments and bulkhead insulation are assumed to be constants. The



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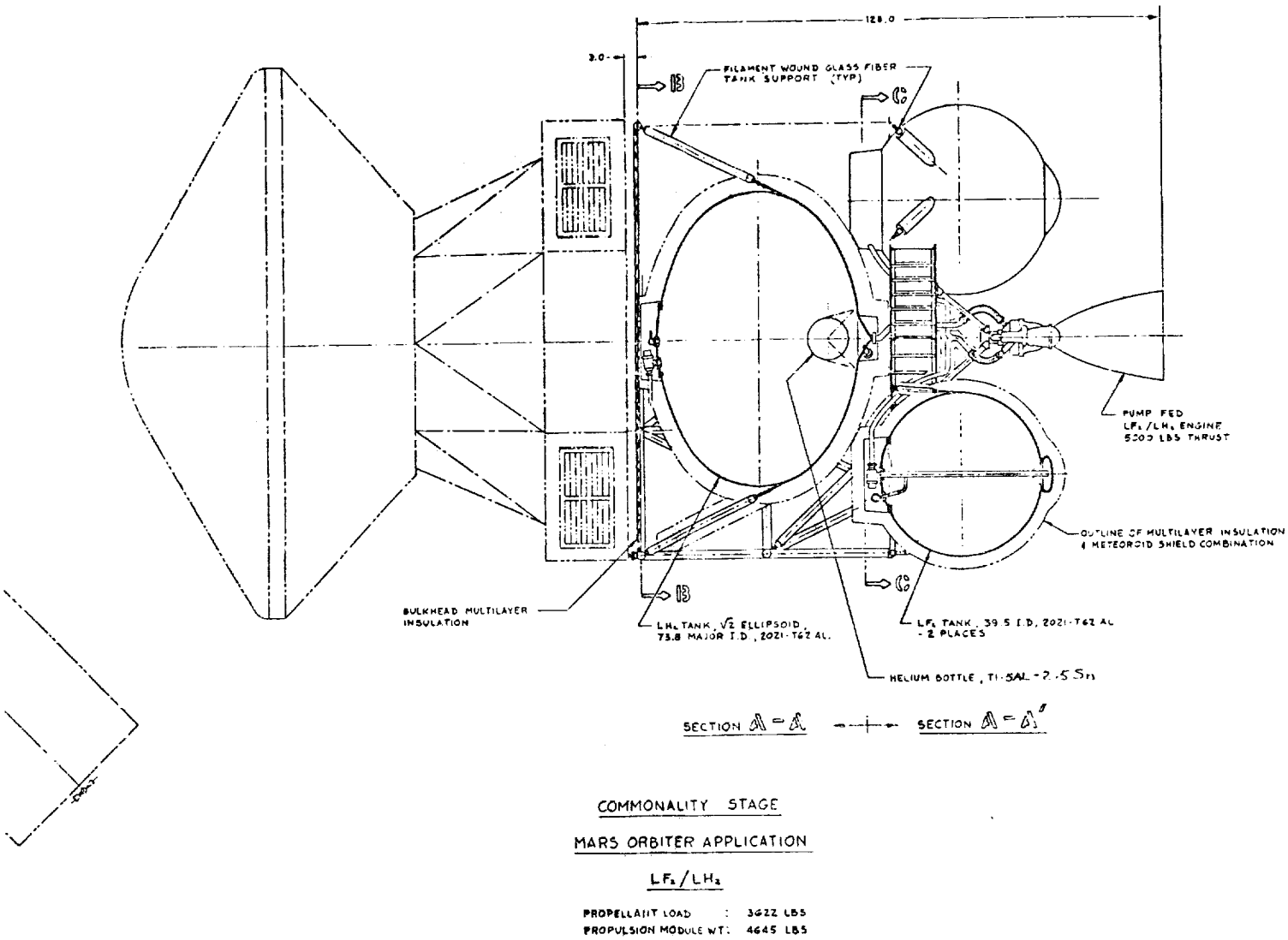

COMMONALITY STAGE

MARS ORBITER APPLICATION

LF_2/LH_2

PROPELLANT LOAD : 3622 LBS
PROPULSION MODULE WT: 4645 LBS

Fig. 7 F_2/H_2 Commonality Stage Design

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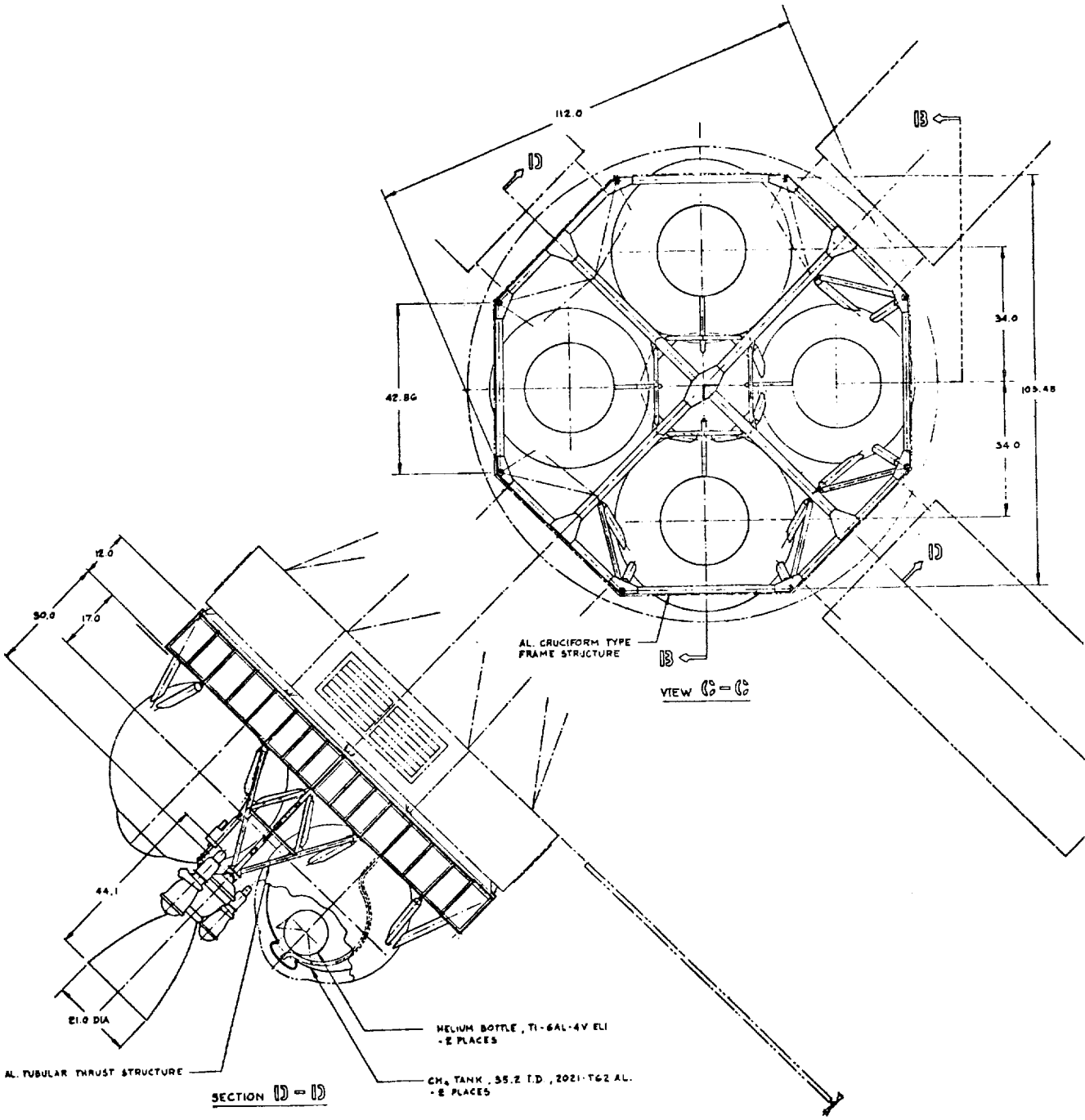

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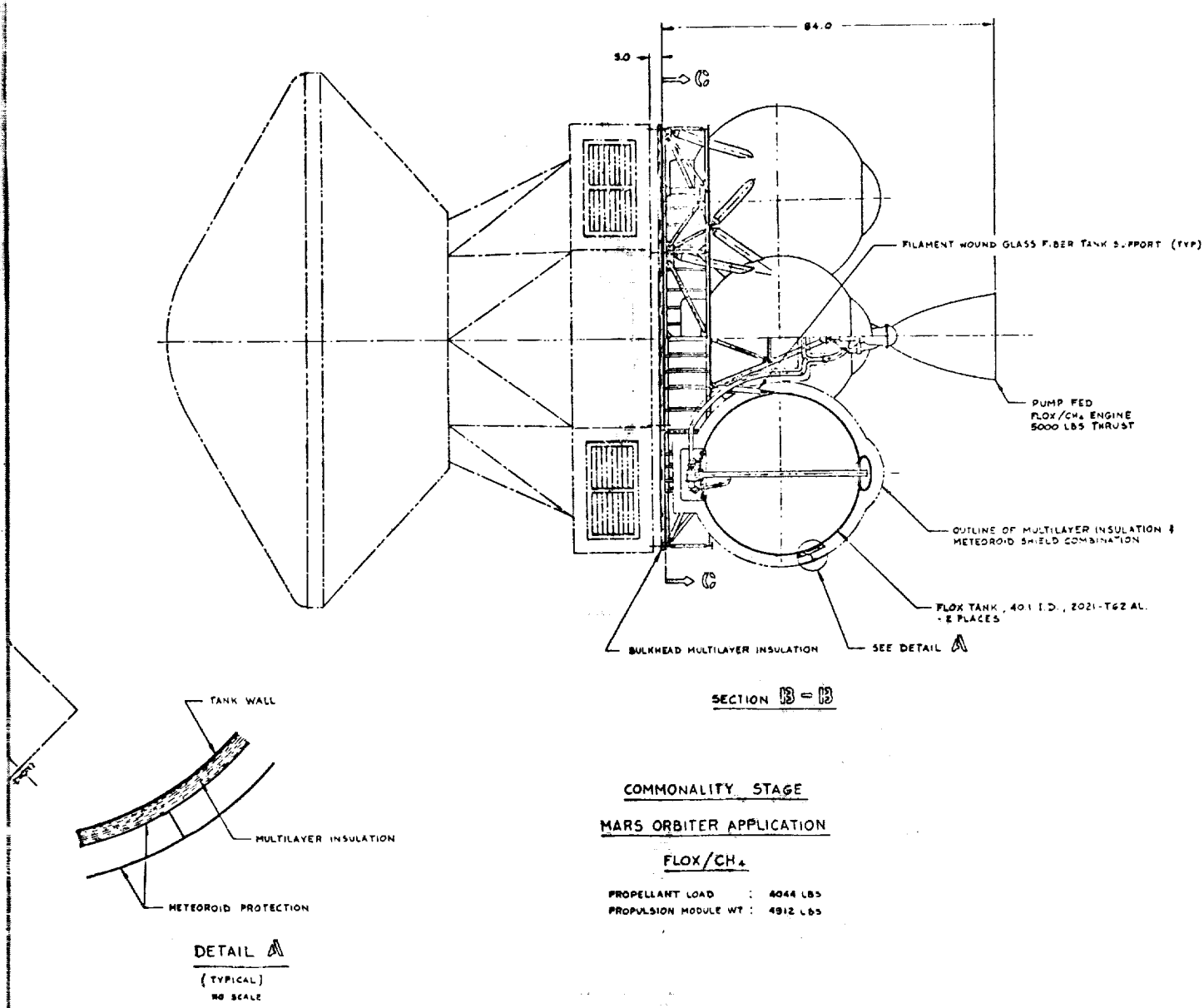

Fig. 8 FLOX/CH_4 Commonality Stage Design

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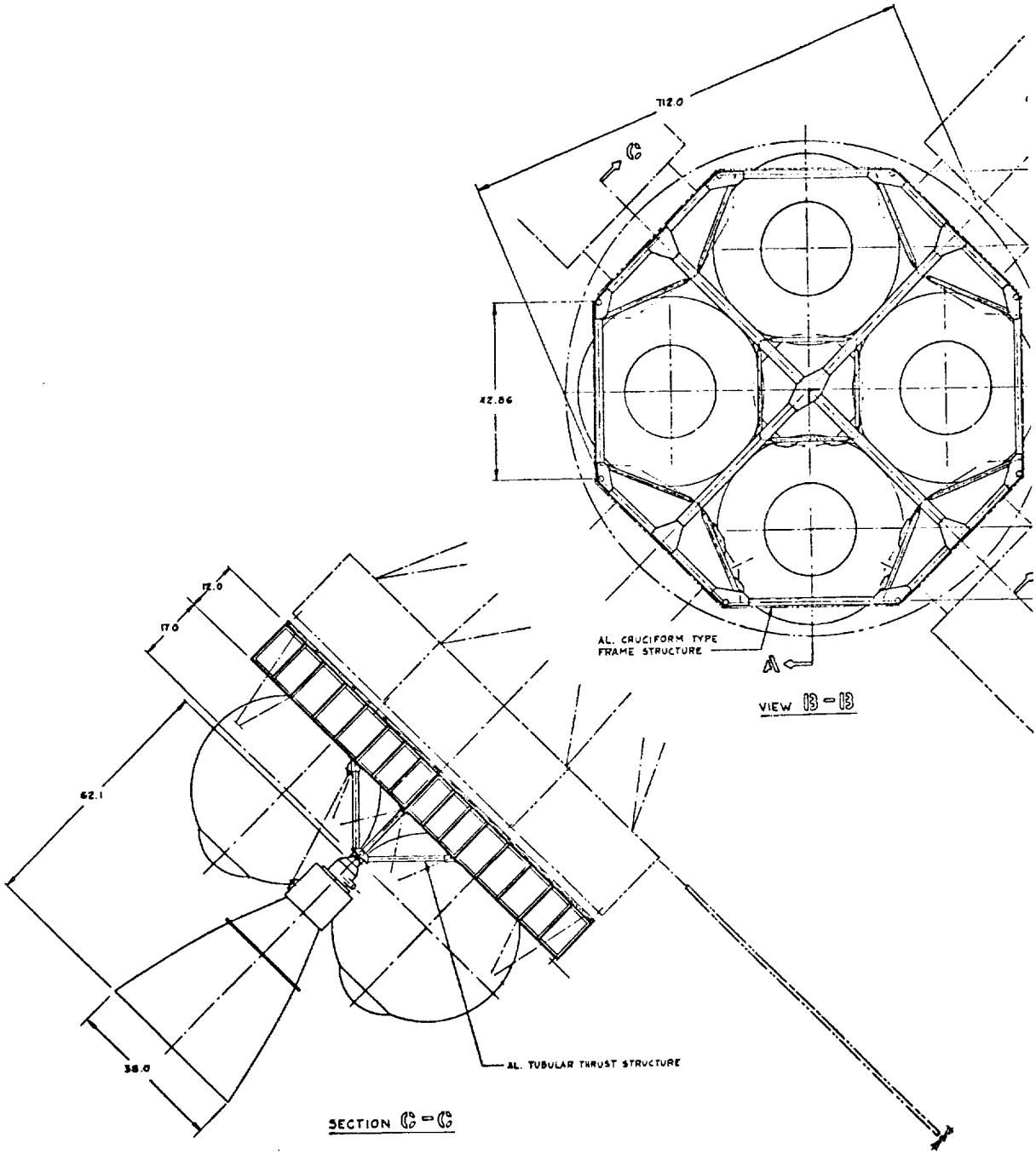

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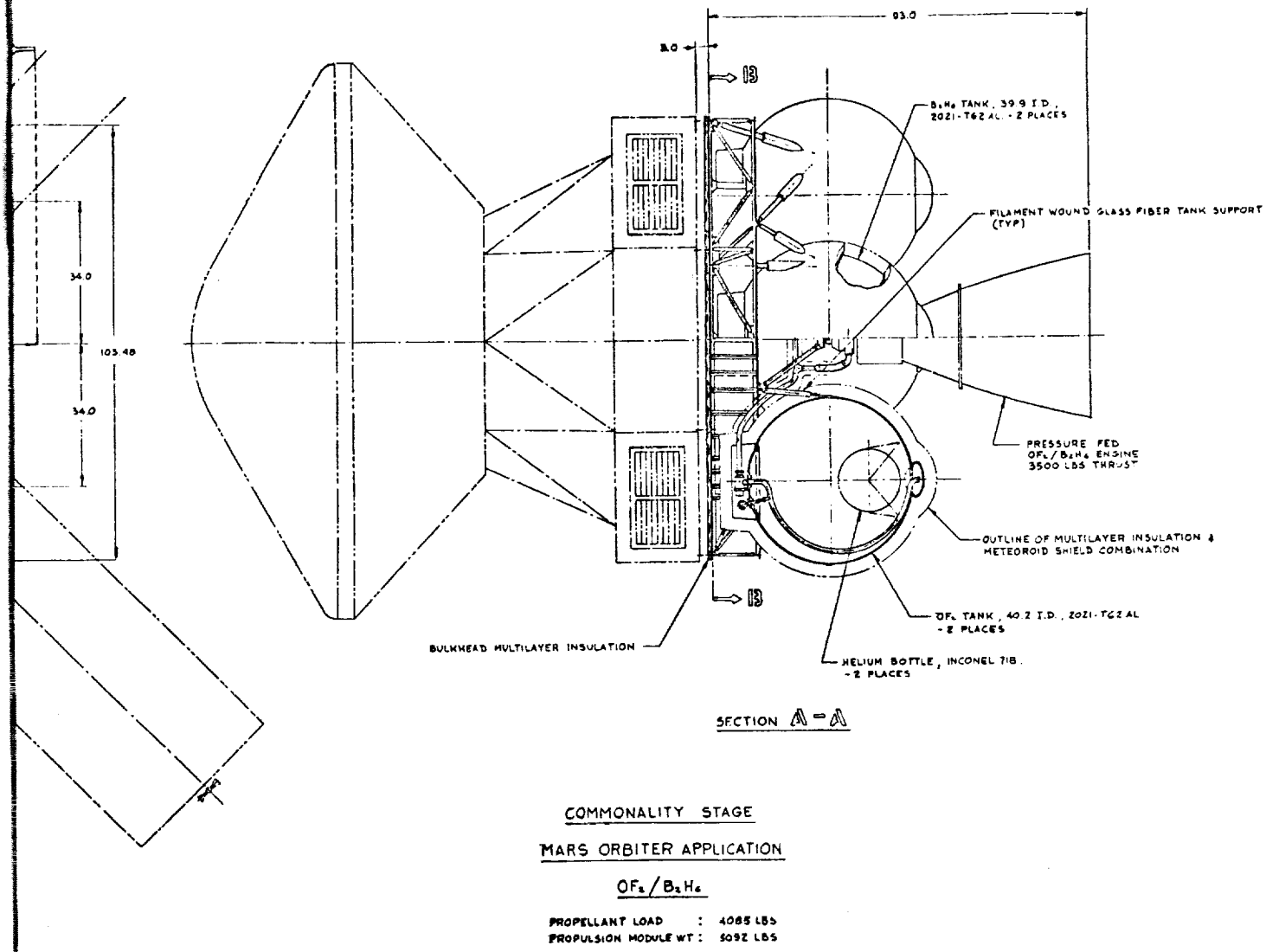


COMMONALITY STAGE

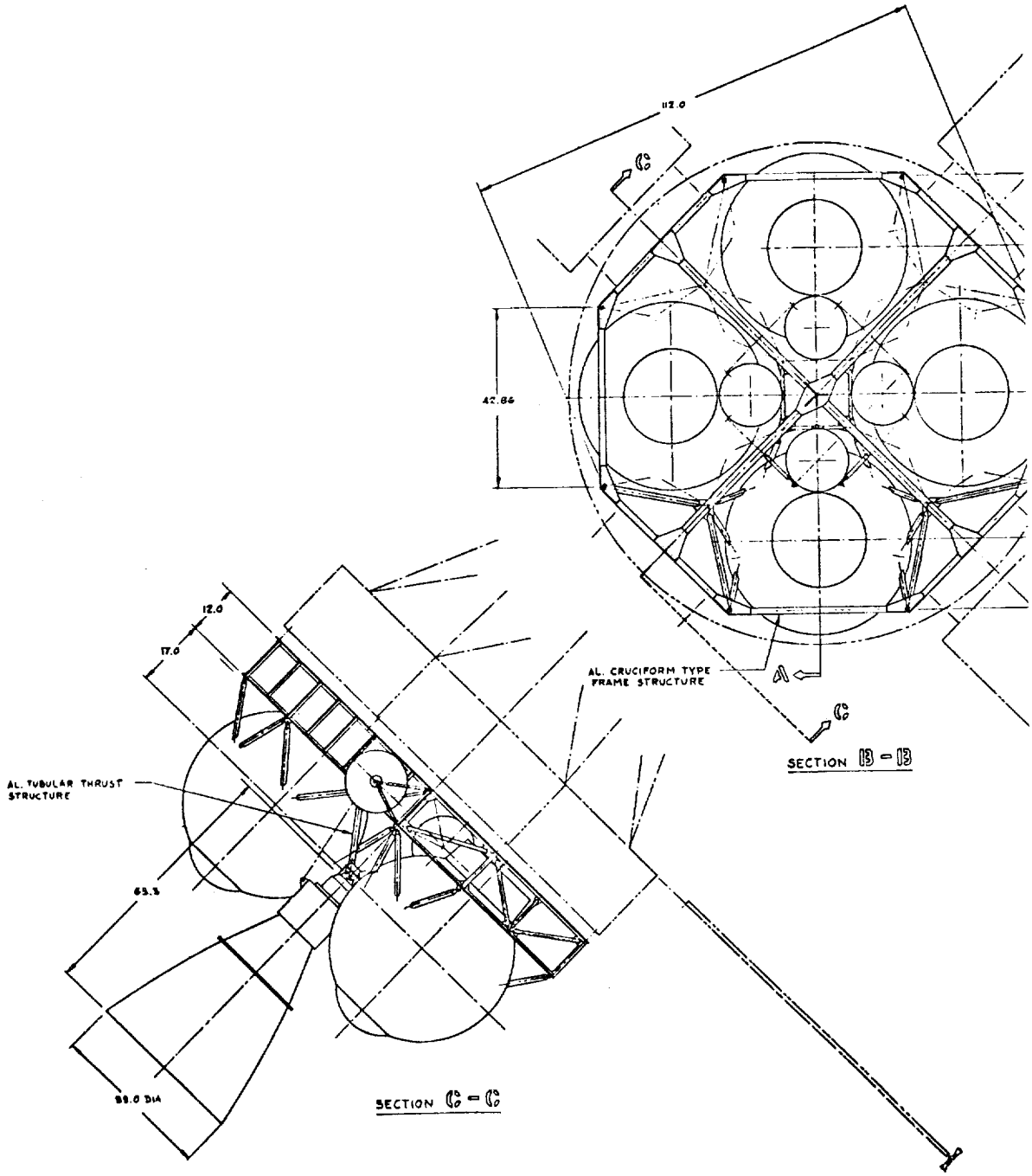
MARS ORBITER APPLICATION

OF_2/B_2H_6

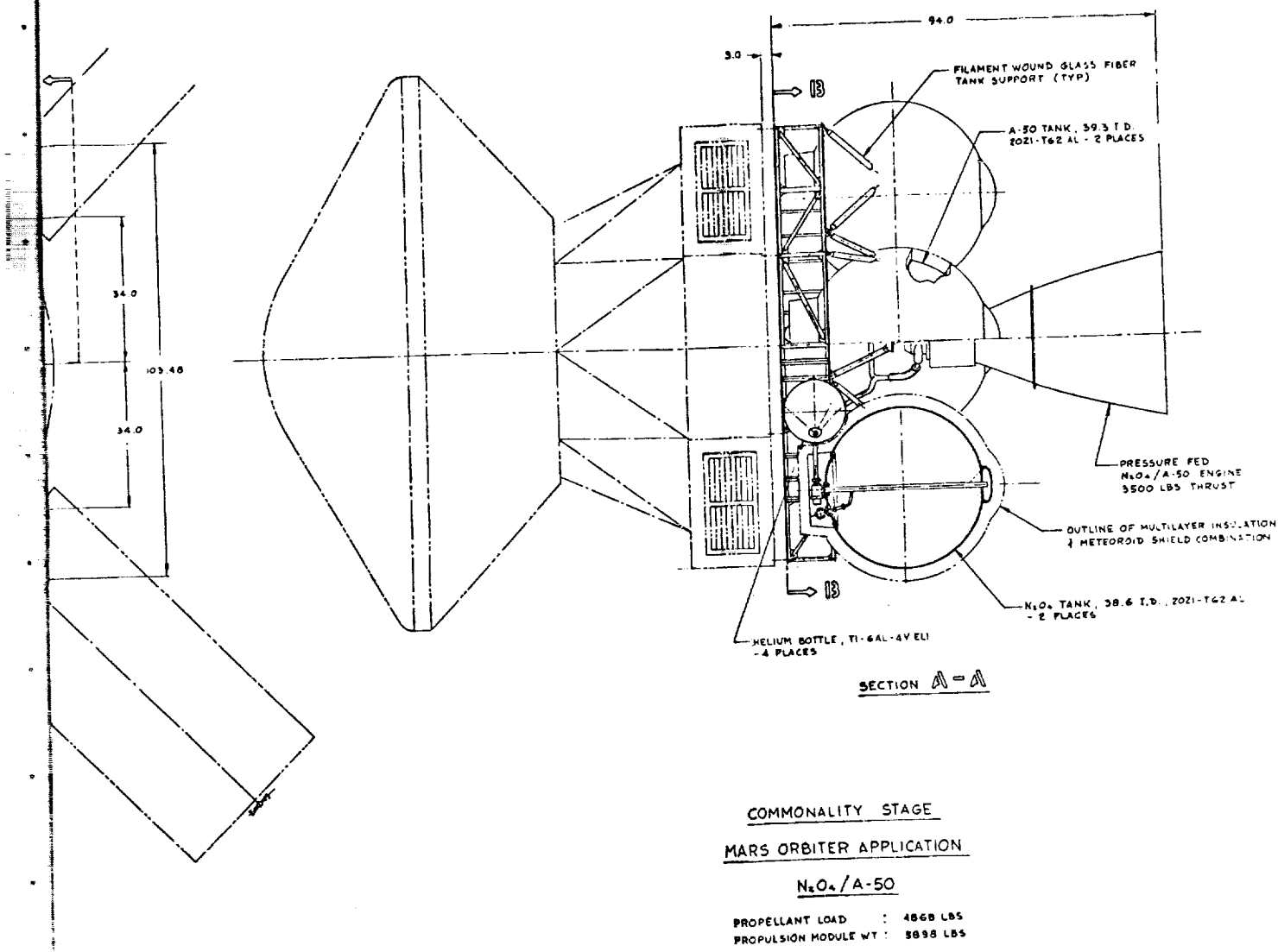
PROPELLANT LOAD : 4085 LBS
PROPULSION MODULE WT : 5092 LBS

Fig. 9 OF_2/B_2H_6 Commonality Stage Design

FOLDOUT FRAME 2



FOLDOUT FRAME 1



COMMONALITY STAGE

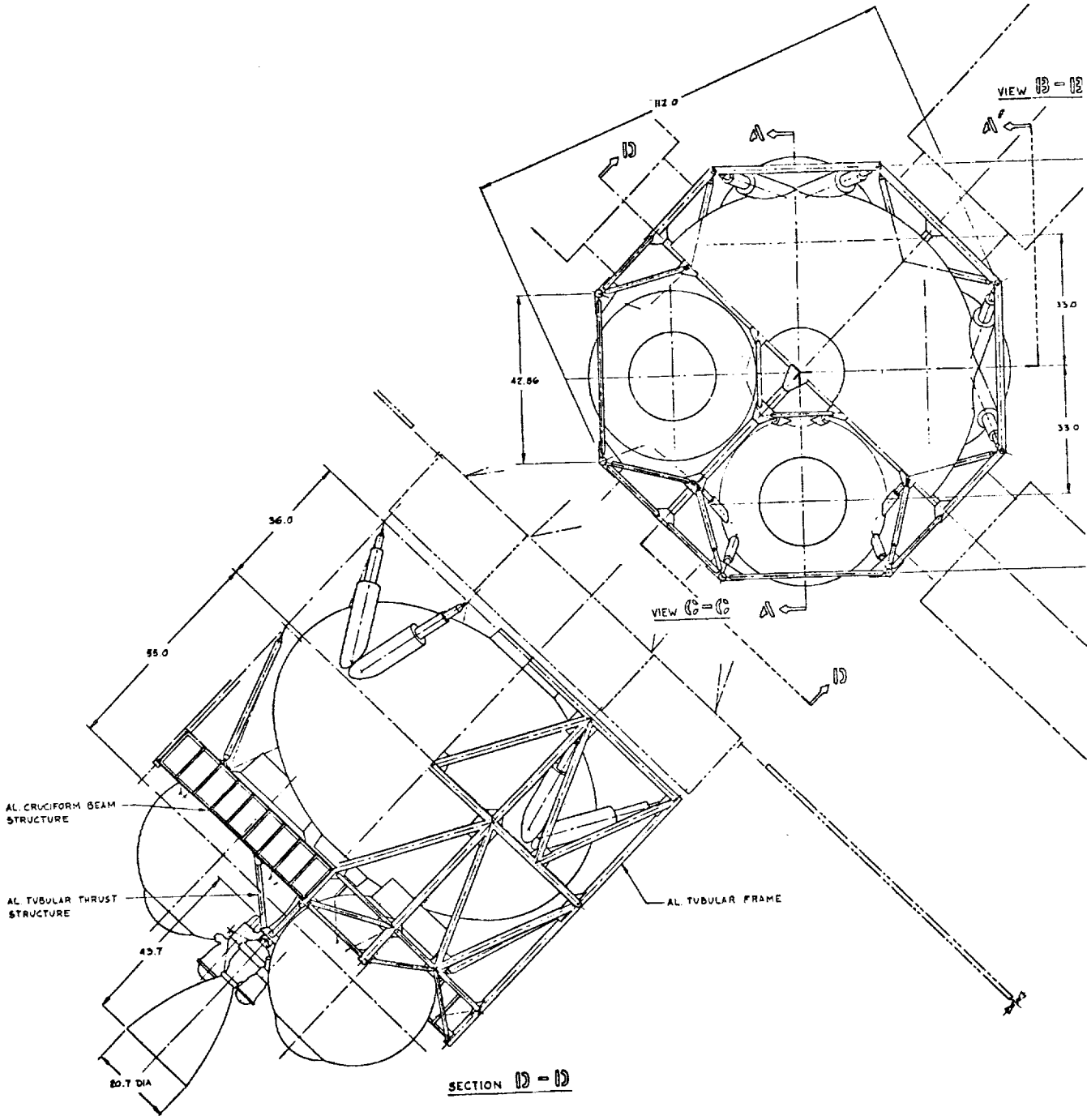
MARS ORBITER APPLICATION

N_2O_4/A-50

PROPELLANT LOAD : 4868 LBS
PROPULSION MODULE WT : 5898 LBS

Fig. 10 N_2O_4/A-50 Commonality Stage Design

FOLDOUT FRAME V



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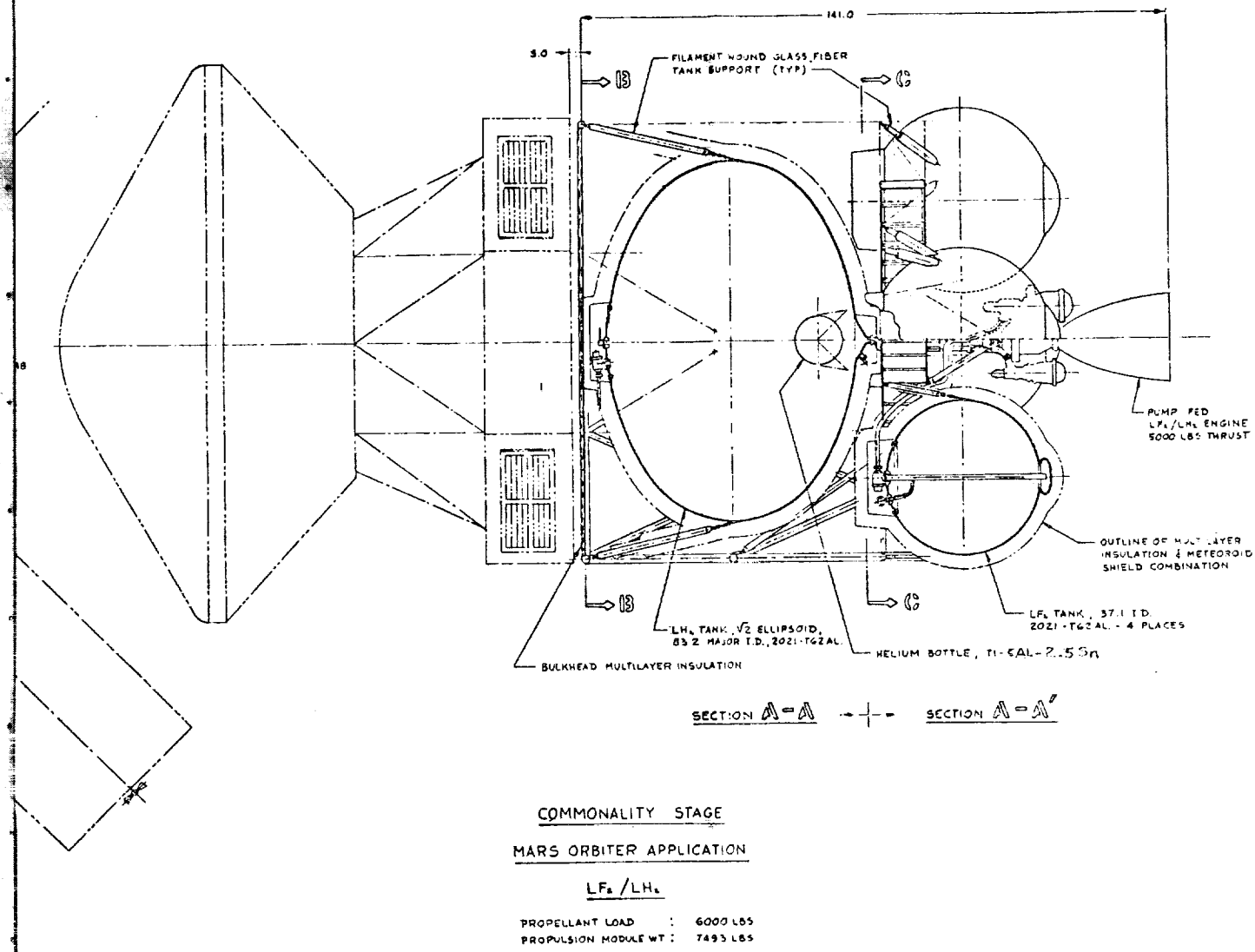


COMMONALITY STAGE

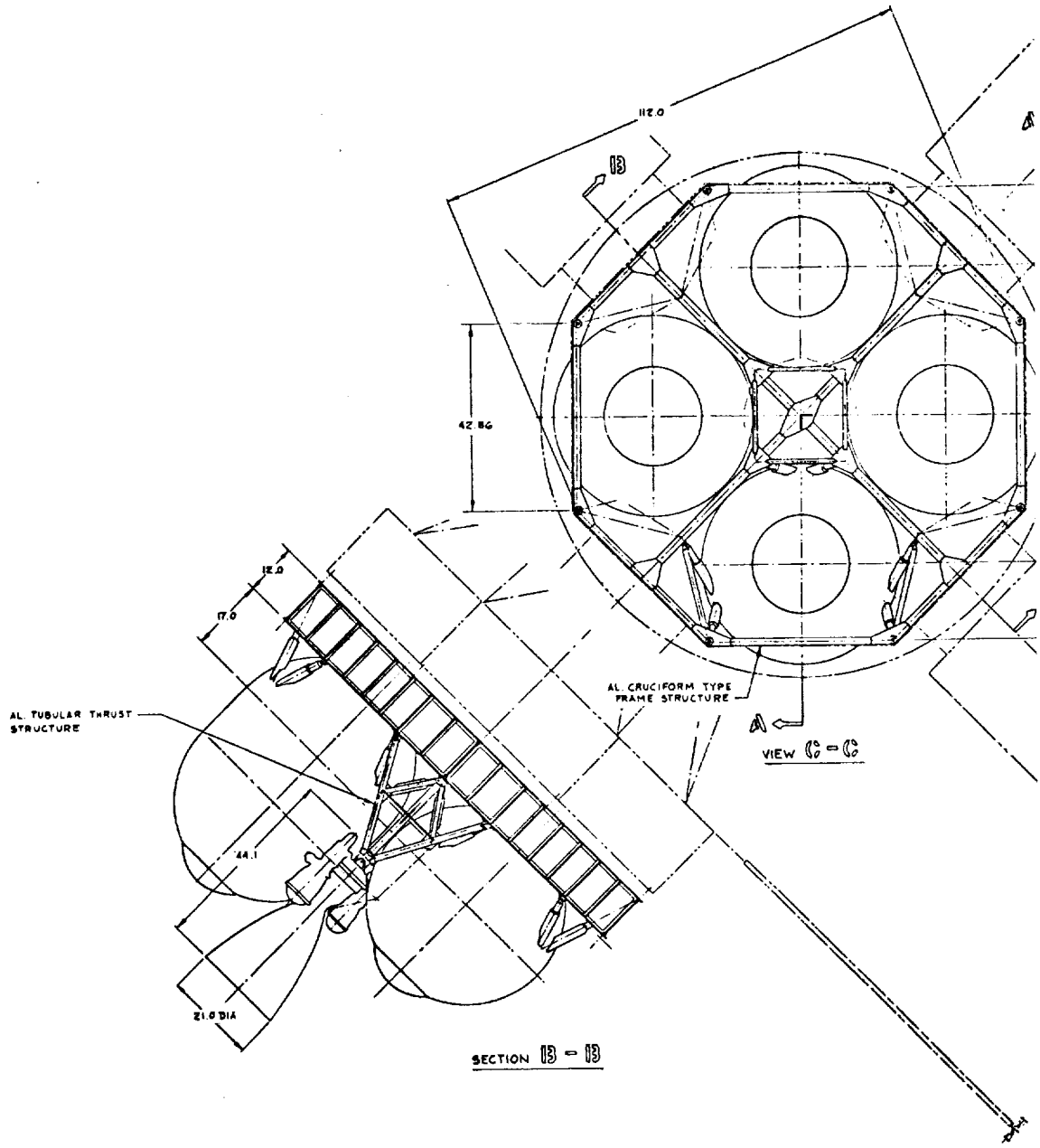
MARS ORBITER APPLICATION

LF_2/LH_2

PROPELLANT LOAD : 6000 LBS
PROPULSION MODULE WT : 7493 LBS

Fig. 11 F_2/H_2 Stage Design With 6,000 Lb Propellant

FOLDOUT FRAME 2



FOLDOUT FRAME 1

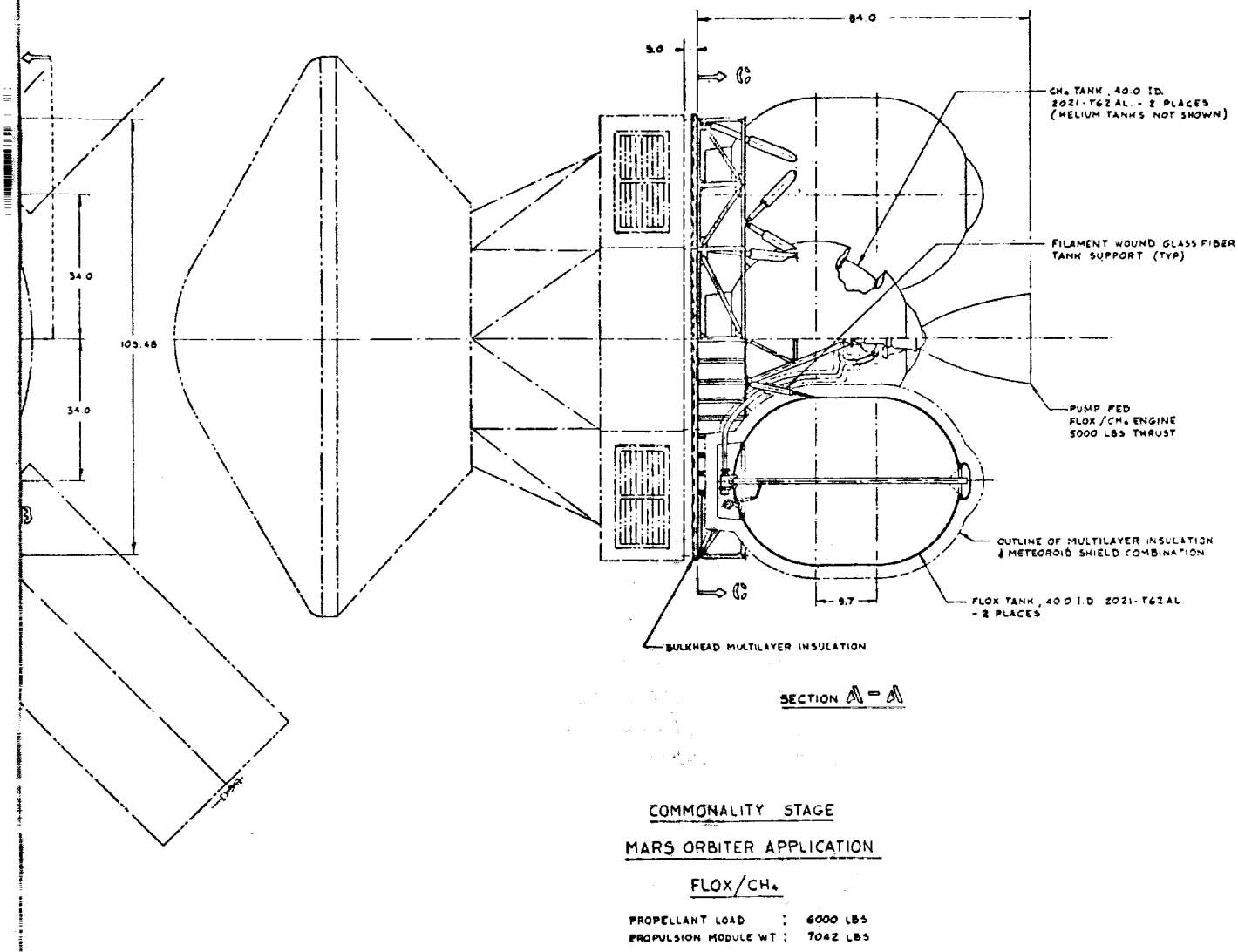


Fig. 12 FLOX/CH_4 Stage Design With 6,000 Lb Propellant

FOLDOUT FRAME 2

tanks, meteoroid shield, and tank insulation were varied with propellant load. Tank weights are also a function of system pressure, and insulation weight is a function of the optimization solution. The meteoroid shield is a function of both tank size and mission. The pressurization system consists of a fixed plumbing network plus a variable tank size based on the pressurization requirements for the mission. The engine system weight is fixed for each propellant combination. The sum of all of these items constitute the dry inert weight of the propulsion module and a 10 percent contingency is added to these weights in order to cover unknowns.

The fluid residuals consist of the liquid and vaporized propellant remaining at the end of the mission and the helium pressurant gas. These residuals include outage, load tolerance, propellant utilization, and line entrapment, plus a performance reserve of 1 percent ΔV incorporated as a contingency for low specific impulse. All of these items are included in the total inert weight of the propulsion module.

Tank and structural weights were based on the program ground rules and general design practice. Valves, plumbing, and pressurization system hardware was sized for each propellant and each configuration in order to make the comparison between propellant combinations as accurate as possible.

The remaining weight elements are the impulse propellant and payload. These values vary for each mission, and weight statements for each propellant and mission combination are therefore presented under performance – Section 1.7 and in Appendix C.

Fluid Systems. The propulsion fluid systems were developed to meet propulsion requirements in a simple and effective manner as described in Section 1.6 of Volume II. Fluid systems schematics for earth-storables, space-storables and cryogens are presented in Figs. 13, 14, and 15 respectively. Pressurant tanks for the cold propellants are shown buried in the oxidizer or fuel tank on the diagrams as analyzed, but could be external and encapsulated with the propellant tank for thermal protection if desired.

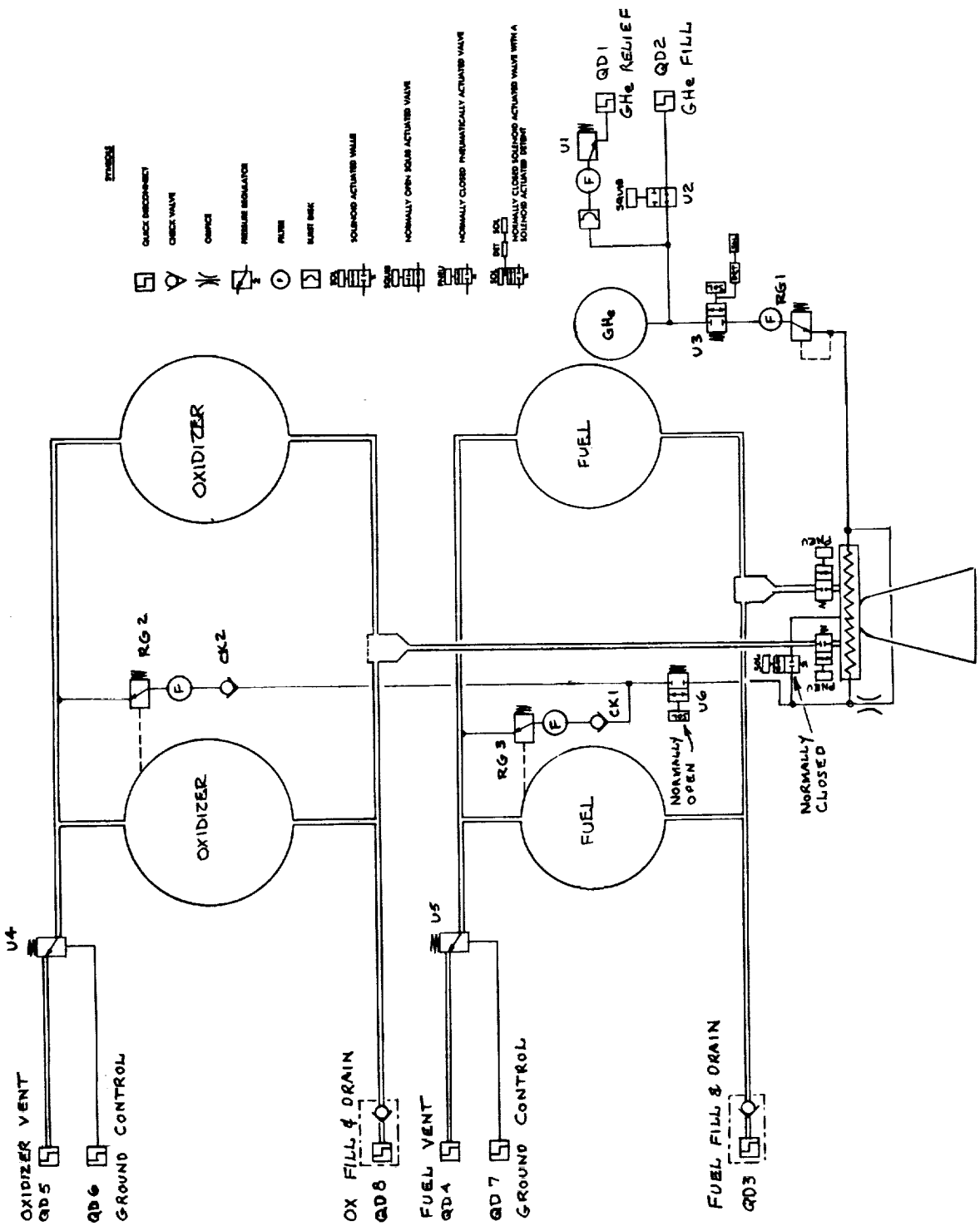


Fig. 13 Earth-Storable Stage Fluid Systems Schematic

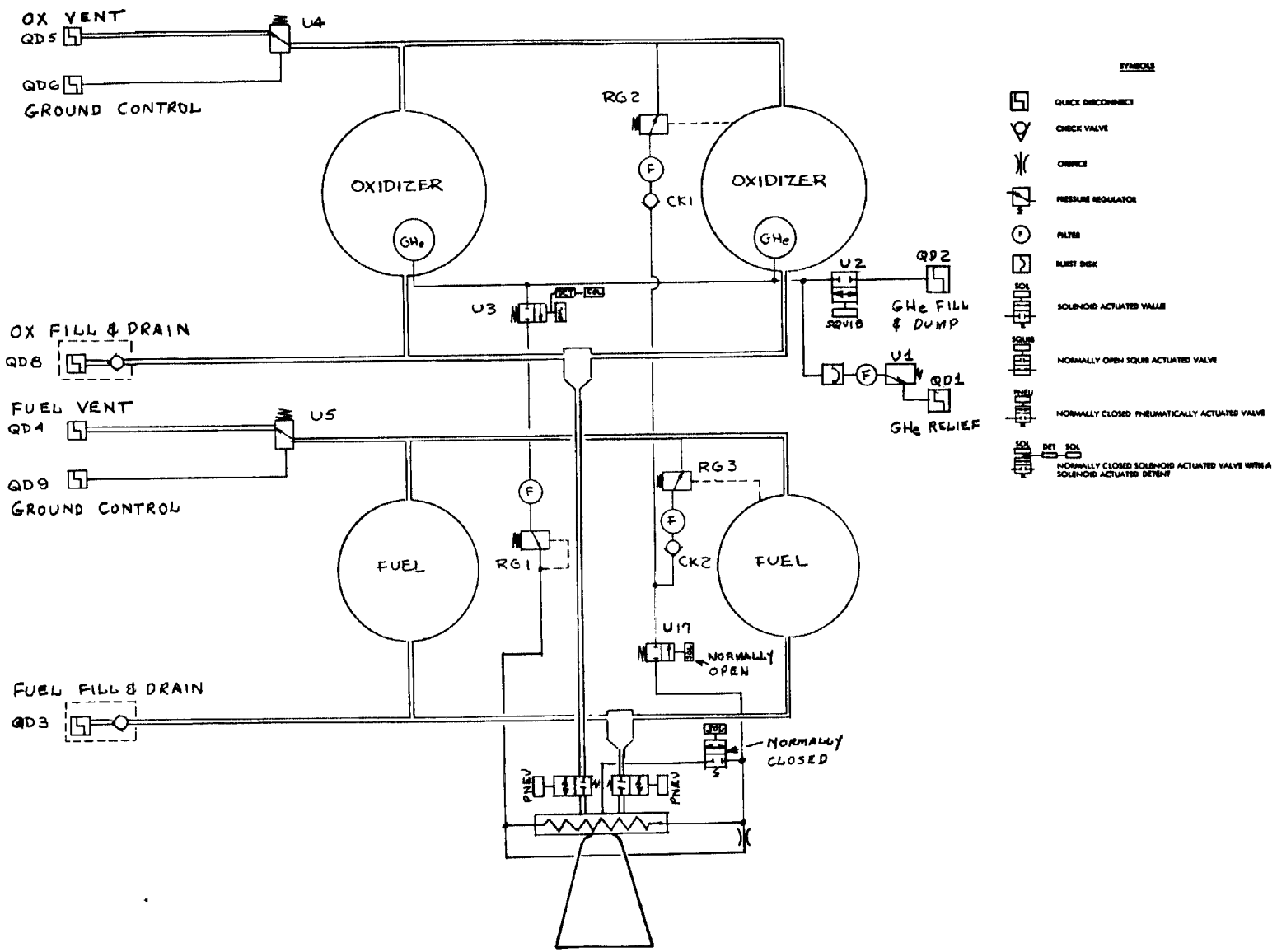


Fig. 14 Space-Storable Stage Fluid Systems Schematic

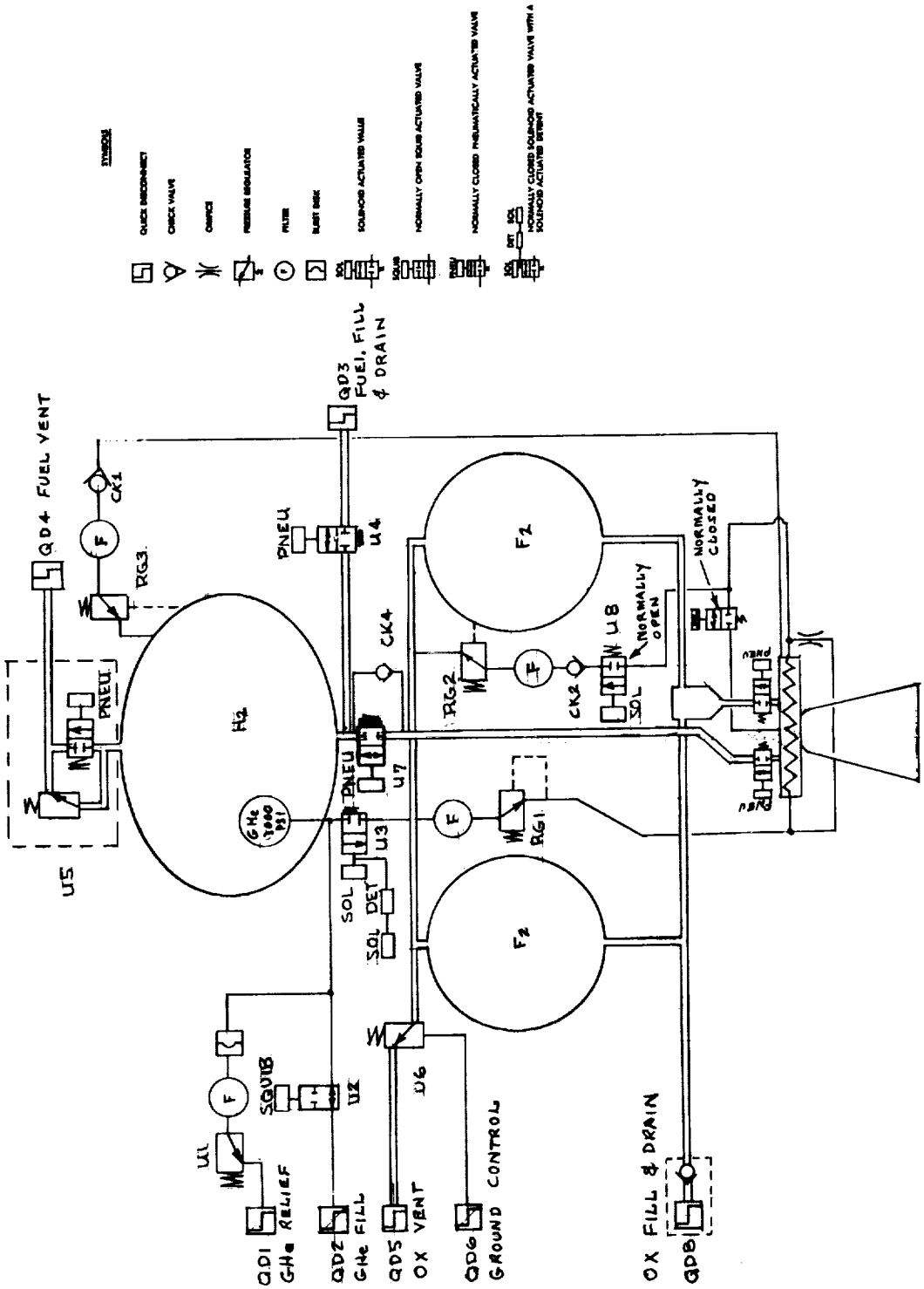


Fig. 15 Cryogenic Stage Fluid Systems Schematic

1.6 THERMODYNAMIC DESIGN AND ANALYSIS

1.6.1 Introduction

Thermodynamic optimization studies were conducted to establish the performance achievable for the propellants and missions under consideration. These studies involved computation of heat transfer, propellant temperature response, pressurization system requirements and determination of optimum values for all thermal/structural parameters. Propellants investigated were:

F_2/H_2
$FLOX/CH_4$
OF_2/B_2H_6
N_2O_4/A-50

Table 8 presents the mission characteristics which significantly affect thermal/structural parameters.

Table 8

THERMODYNAMICALLY SIGNIFICANT MISSION CHARACTERISTICS

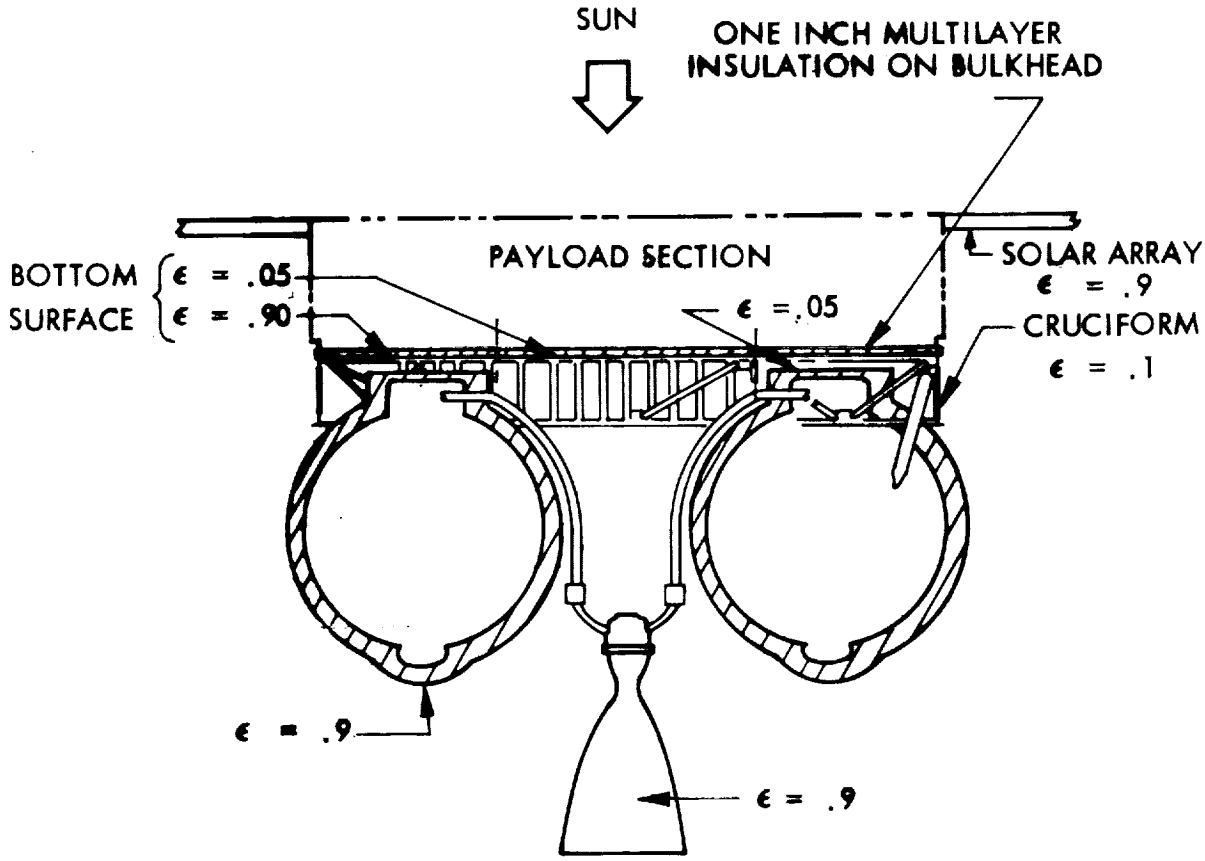
Mission	Duration (Days)	Ascent Burn	Total Burns	Solar Arrays
Mars Orbiter (Baseline)	205	No	4	Yes
Venus Orbiter	170	No	3	Yes
Lunar Orbiter	68.5 Hours	No	4	No
Jupiter Orbiter	900	Yes	5	No
Mars Orbiter (on Titan IIID)	205	Yes	5	Yes
Mars Orbiter (6,000 lb Propellant)	205	Yes	5	Yes

The techniques employed in conducting the thermodynamic and performance analyses in the commonality stage task were improved considerably over the methods used previously. A Thermodynamic Optimization Program (TOP) developed with Lockheed independent development funds was utilized for this task. This program, unlike its predecessors, provides a capability for conducting optimization analyses with any number of simultaneously varying parameters. This constitutes an improvement in efficiency and accuracy over previous methods where parametric analyses were conducted by varying each parameter one at a time and selecting optimum values by graphical techniques.

The overall analysis approach involves defining the environment and configuration, determining temperature distributions and the resultant propellant heating, and optimization of the thermal protection and pressurization systems. The temperature and heat transfer analyses are conducted by establishing thermal mathematical-models for each propulsion module configuration and using a thermal analyzer program (THERM) to solve the heat transfer problems. Using the temperature response information determined with the THERM program, optimization analyses are conducted with the TOP computer program to establish the optimum values of insulation thickness, ullage volume, tank pressure, pressurization gas requirements and inert weights.

1.6.2 Thermal Model Development

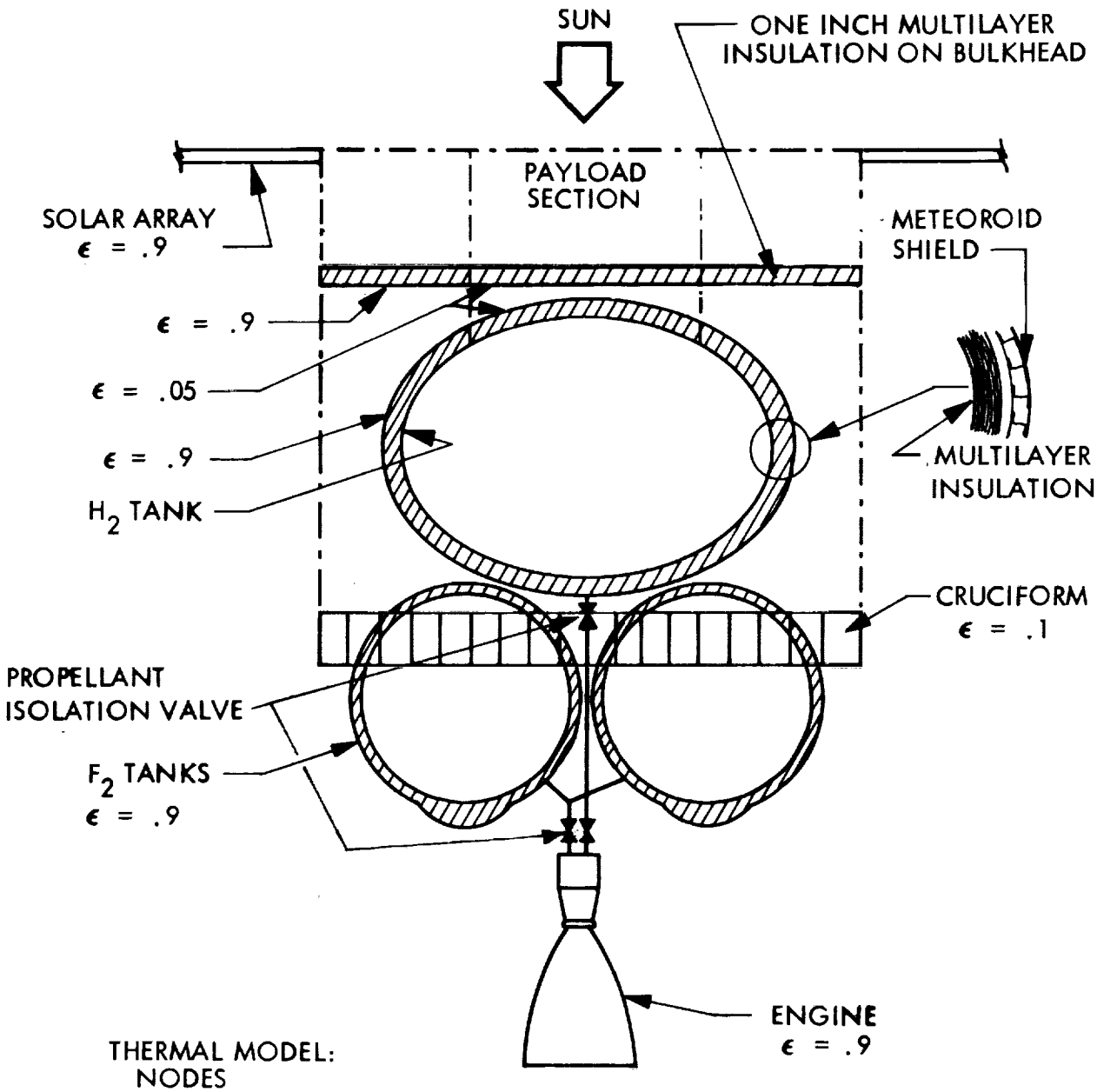
Two basic thermal models were developed, one to represent the configuration which has four spherical tanks and one to represent the F_2/H_2 configuration. Outline sketches of these configurations are shown in Figs. 16 and 17 which also include external surface finishes and the number of elements (nodes, resistors) in the thermal network. Figures 18 and 19 show simplified thermal conduction networks. These networks are not representative of the total number of elements in the networks analyzed, but are presented to illustrate the kinds of elements considered and the manner in which they interact. Surface finishes were not optimized, but favorable values were selected based on experience and are near optimum. For example, emittance values for the forward bulkhead were defined as low values near the center of the module and high near the periphery. External surface emittance is low on the



THERMAL MODEL:

NODES	
FLOX TANKS	36
CH_4 TANKS	36
FWD BULKHEAD	2
PAYLOAD & SOLAR ARRAY	5
ENGINE & SUPPORTS	6
RESISTORS (EACH TANK)	
RADIATION	100
CONDUCTION	16

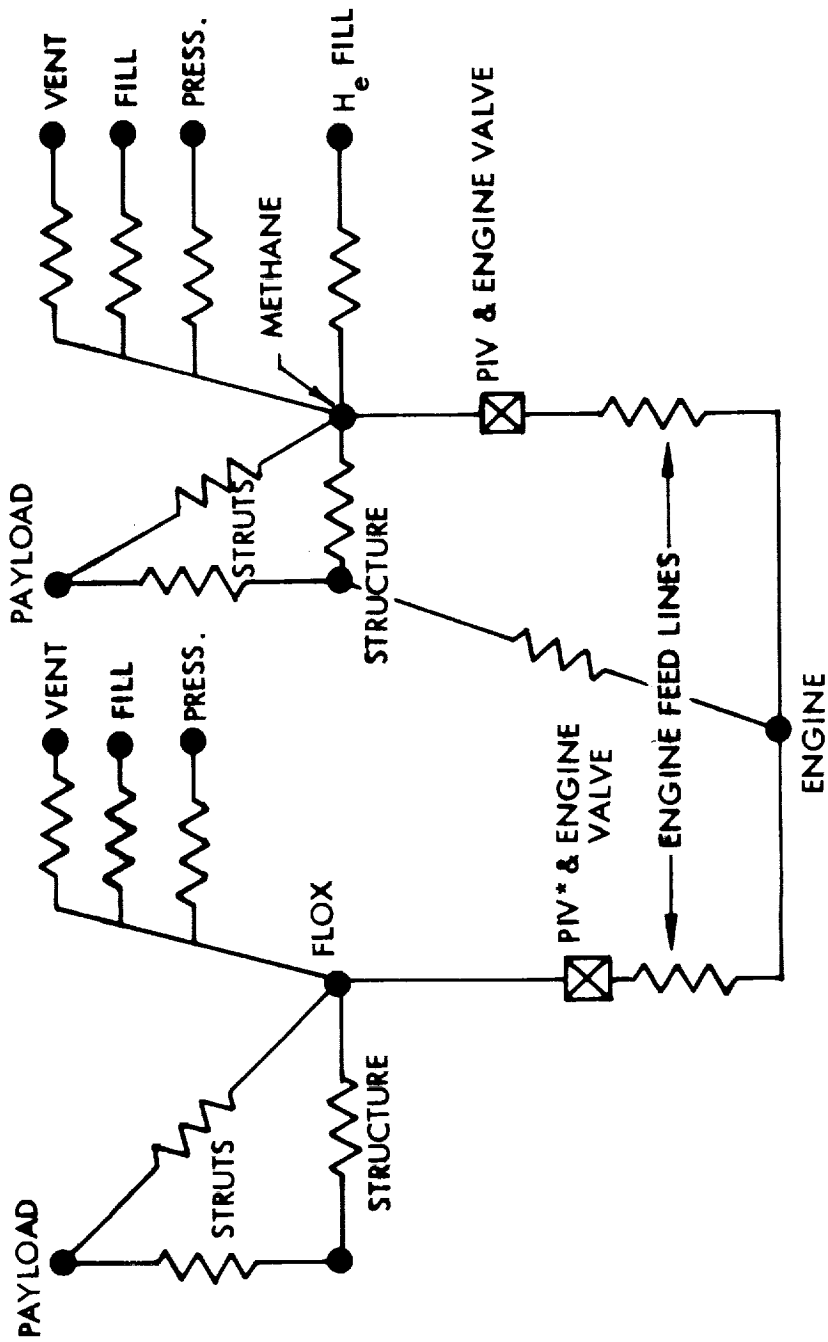
Fig. 16 **FLOX/CH_4 Thermal Model – Four Spherical Tanks**



THERMAL MODEL:

NODES	
H_2 TANK	14
F_2 TANKS	24
FWD BULKHEAD	2
PAYLOAD & SOLAR ARRAY	2
ENGINE & SUPPORTS	6
RESISTORS	
H_2 TANK, 43 RADIATION 7 CONDUCTION	
EACH F_2 TANK, 37 RADIATION 7 CONDUCTION	

Fig. 17 F_2/H_2 Thermal Model



*PROPELLANT ISOLATION VALVES

NOTE: FEED LINES FILLED WITH LIQUID FROM TANK TO PIV

Fig. 18 Simplified Thermal Conduction Network – FLOX/Methane

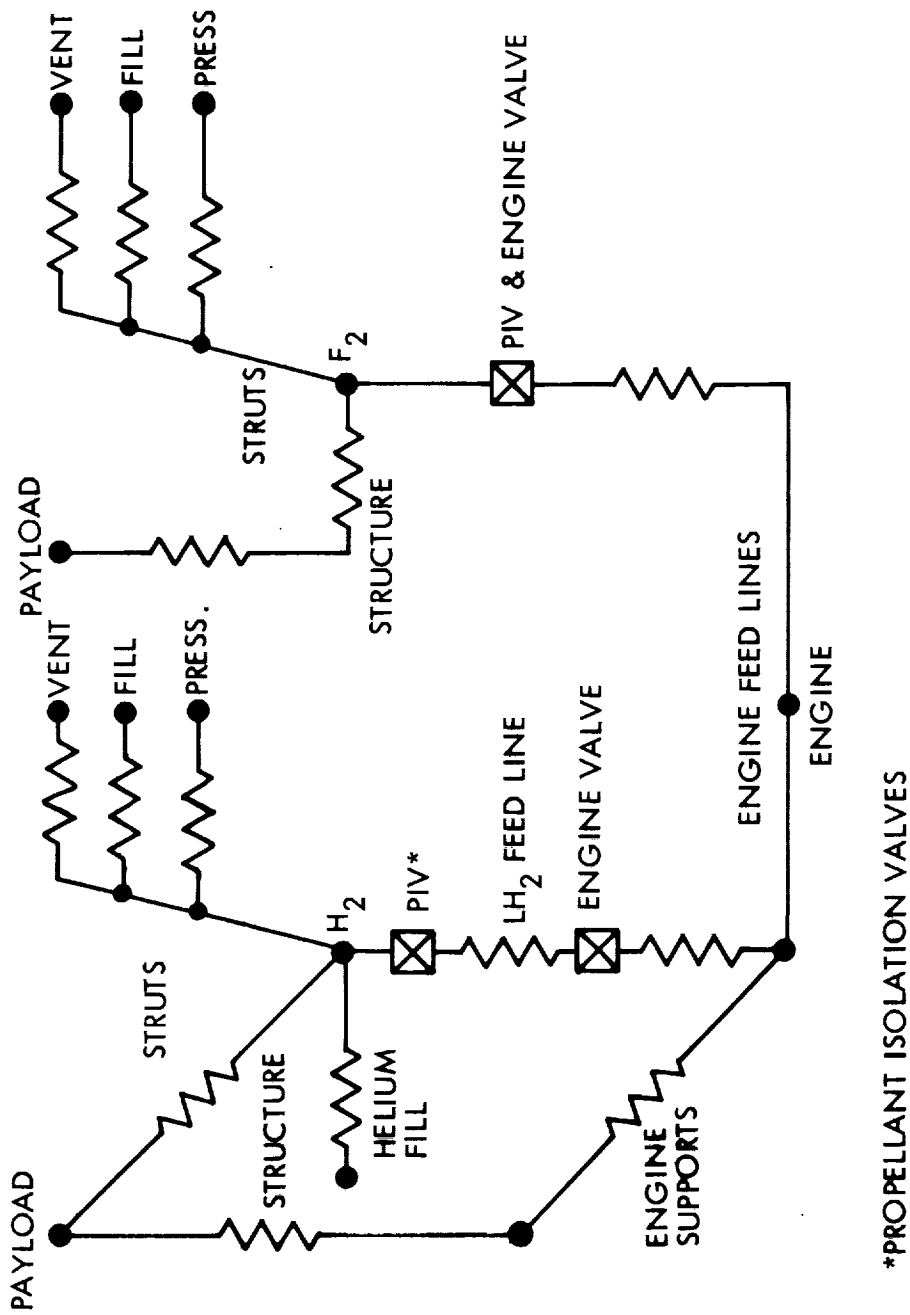


Fig. 19 Simplified Thermal Conduction Network – F_2/H_2

forward surface of the tanks and is high on aft surfaces which can radiate to space. Surface finishes assumed on structural elements were selected to minimize propellant heating (or cooling where applicable).

Solar arrays have a very significant effect on tank surface temperatures and consequently on propellant heating. The location of the arrays relative to the propulsion module is indicated in Figs. 16 and 17. It was assumed that the solar arrays would require a high-emittance aft-facing surface to minimize the array temperature and raise the conversion efficiency. Solar array cooling is anticipated to be a problem near Earth and at less than one AU. Propellant heating could be reduced by moving the arrays forward or by insulating the aft facing surface. Each of these items would have an impact on payload, however.

The equipment section was assumed to be maintained at 70°F (530°R) throughout all missions considered. As in previous studies it was assumed that a thermal bulkhead blanket consisting of one inch of multilayer insulation separates the payload and propulsion module. This thickness of insulation was not optimized. The same thermal resistance could be provided with 5 radiation shields (not in contact with each other) if shield emittances are 0.05 or less. Such a system might consist of aluminized mylar sheets mounted to prevent contact between sheets. This shield could offer a weight improvement over the one inch thick insulation bulkhead.

Tank support-strut heat leaks were computed on the basis of solid conduction in the tube walls. Radiation within the hollow core of the struts is assumed to be negated by packing with a glass fiber filler. Heat transfer through the multilayer on the struts is assumed negligible. This assumption is conservative for the propulsion module which is shaded from the sun because energy is actually lost by radiation from the outer surface of the strut insulation. The strut material is assumed to be S-type glass fiber with an average thermal conductivity of 0.26 Btu/hr-ft-°R.

Propellant line heat leak paths are included in the thermal model. The resistance of each line is based on solid conduction in the line wall. Heat transfer through the line insulation, and radiation within the lines, are assumed negligible. For the analysis, feed lines for all propellants (except H_2) are assumed to be filled with liquid up to the engine valve. In the case of H_2 a tank shutoff valve is assumed which results in a vapor

filled feed line. Table 9 shows the thermal resistance of structural, line and insulation elements and the relative amount of heat transfer contributed by each. Table 10 shows temperatures of major elements in the F_2/H_2 and $FLOX/CH_4$ systems. An example of the external temperature distribution around the tanks and through the structure is shown in Figs. 20 and 21 for F_2/H_2 and $FLOX/CH_4$ systems. Note that the propellant isolation valve temperature and average liquid propellant temperature correspond. The H_2 system was assumed to have a propellant isolation valve at the tank and an evacuated feed line.

Comparison of heat rates through insulation, lines, and structure for the $FLOX/CH_4$ system shows that the engine feed line is the greatest contributor of heat for the Mars Orbiter mission. This results because the engine temperature remains well above the propellant temperature in configurations which have solar panels. Also the engine is assumed to have a high emittance surface. Heating through all lines other than the feed line is very small in the $FLOX/CH_4$ system because temperatures in the module are near the propellant temperature. Those missions which do not use solar panels have no significant positive heating because source temperatures are generally less than the temperature of the FLOX and CH_4.

The F_2/H_2 system heat leaks are only slightly more significant. The feed line contributes slight heating in the configurations which employ solar panels, while insulation and structural penetration heat leaks are dominant. The solar constant variation for all missions is shown in Fig. 22. For vehicle orientations where the propulsion module is shaded from the sun, the solar arrays are the only elements of the model which are directly exposed to the sun.

The temperature of the external surface of the tank insulation (actually external surface temperatures of the meteoroid shields were used) is primarily a function of the external environment. It is, however, somewhat a function of the inner boundary or liquid temperature. The effect of the inner boundary temperature is taken into account. When the "THERM" program is run to determine the insulation temperatures and heat flow rates, computations are made for two thicknesses representing an estimate of the limits within which an optimum will occur. These temperatures, though they are not significantly

Table 9

MARS ORBITER

F_2/H_2 & FLOX/CH_4 PROPELLANT HEATING RATES

Propellant	Resistor	R(HR°F/Btu)	ΔT (°R)	q (Btu/hr)	%q
F_2	R_1 (Insulation)	62.	61.	.985	71
	R_2 (Structure)	587.	161.	.274	20
	R_3 (feed line)	340.	33.	.094	7
	R_4 (vent line)	3060.	113.	.037	less than 3
	R_5 (fill line)	4190.	33.	.001	
	R_6 (press.)	15300.	33.	–	
H_2	R_1 (insulation)	130.	167.	1.285	60
	R_2 (structure)	740.	487.	.659	31
	R_3 (feed line)	1785.	182.	.102	5
	R_4 (vent line)	4580.	208.	.045	less than 5
	R_5 (fill line)	6120.	182.	.029	
	R_6 (press. line)	18450.	182.	.010	
	R_7 (He fill line)	18450.	182.	.010	
FLOX	R_1 (insulation)	59.	14.	.240	34
	R_2 (structure)	900.	191.	.212	30
	R_3 (feed line)	340.	80.	.235	34
	R_4 (Vent line)	4900.	30.	.0061	less than 3
	R_5 (fill line)	4900.	30.	.0061	
	R_6 (pres. line)	15300.	30.	.002	
CH_4	R_1 (insulation)	62.	-10.	-.016	-7
	R_2 (structure)	1660.	152.	.092	39
	R_3 (feed line)	340.	42.	.124	53
	R_4 (vent line)	4900.	-10.	-.0020	less than -2
	R_5 (fill line)	6110.	-10.	-.0016	
	R_6 (press. line)	15300.	-10.	-.0006	

SYSTEM	LOCATION	PROPELLANT		(AVERAGE) TANK ** SURFACE		(AVE.) ENGINE	PIV*		(AVERAGE) THERMAL BULKHEAD	MODULE STRUCTURE	SOLAR ARRAY
		OX.	FUEL	OX.	FUEL		OX.	FUEL			
$FLOX/CH_4$	EARTH	154	201	210	225	265	154	201	249	297	578
	MARS	156	187	171	197	234	156	187	235	290	500
	JUPITER	161	160	135	131	170	161	160	207	275	NONE
	VENUS	167	210	241	249	305	167	210	270	307	683
F_2/H_2	EARTH	154	37	259	225	267	154	37	260	376	600
	MARS	154	51	177	219	237	154	51	243	320	501
	JUPITER	143	57	127	75	186	143	57	196	235	NONE
	VENUS	165	51	241	305	294	165	51	290	429	685

*PIV = PROPELLANT ISOLATION VALVE

**TEMPERATURE ON TANK INSULATION OUTER SURFACE, INSIDE METEOROID SHIELD

Table 10 Typical Propulsion Stage Temperatures (°R)

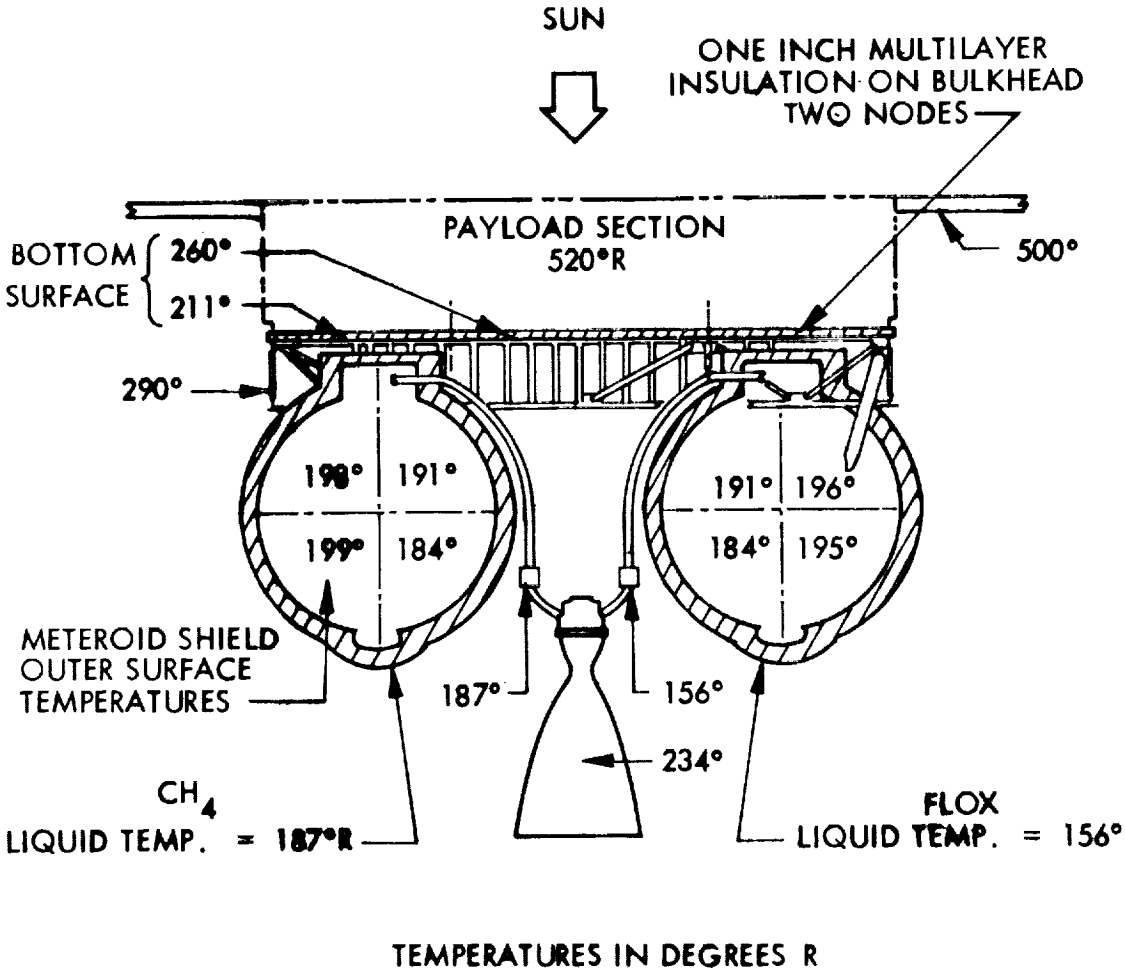


Fig. 20 Mars Orbiter FLOX/CH_4 System Temperatures At Mars

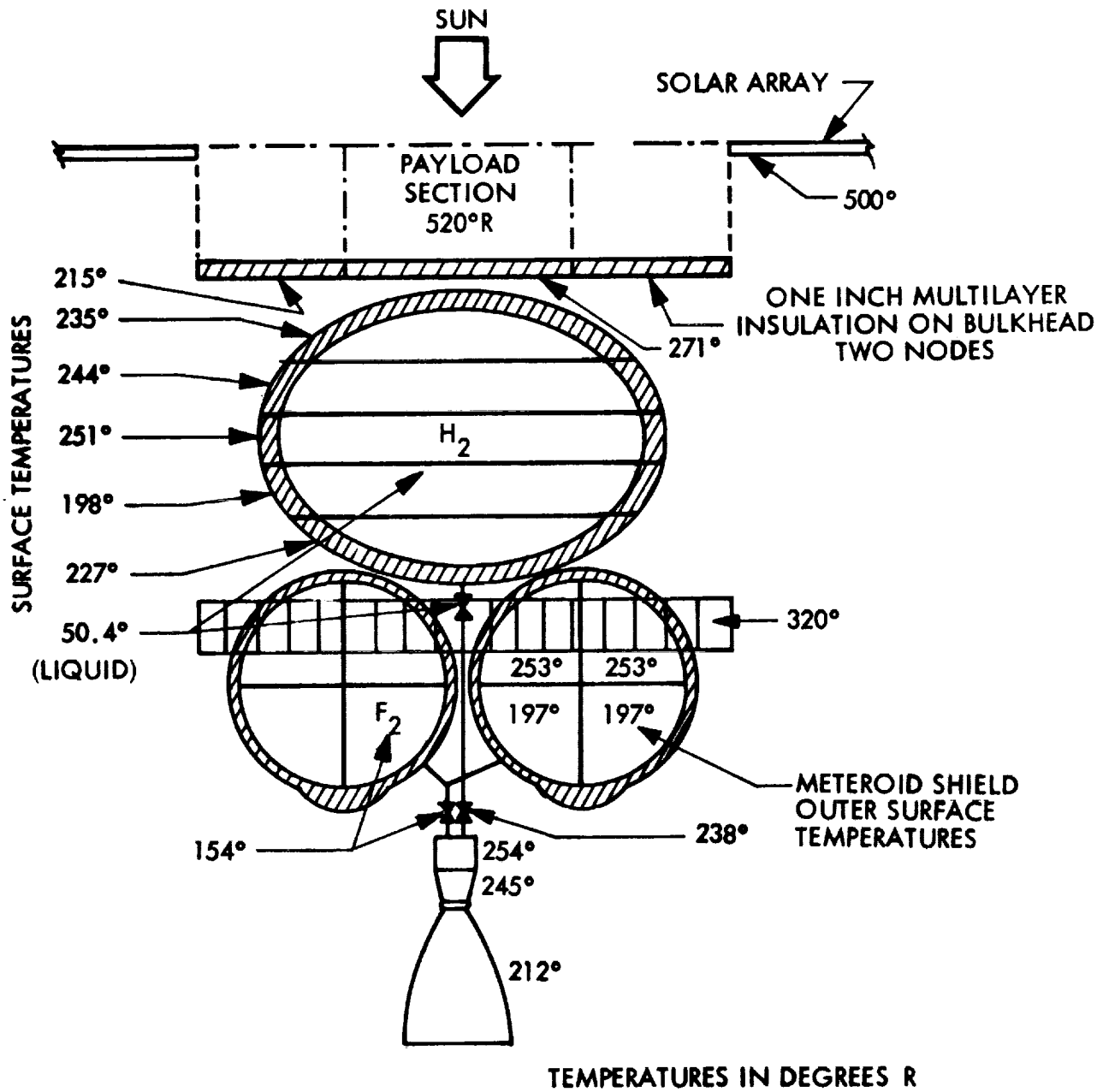


Fig. 21 Mars Orbiter F_2/H_2 System Temperatures At Mars

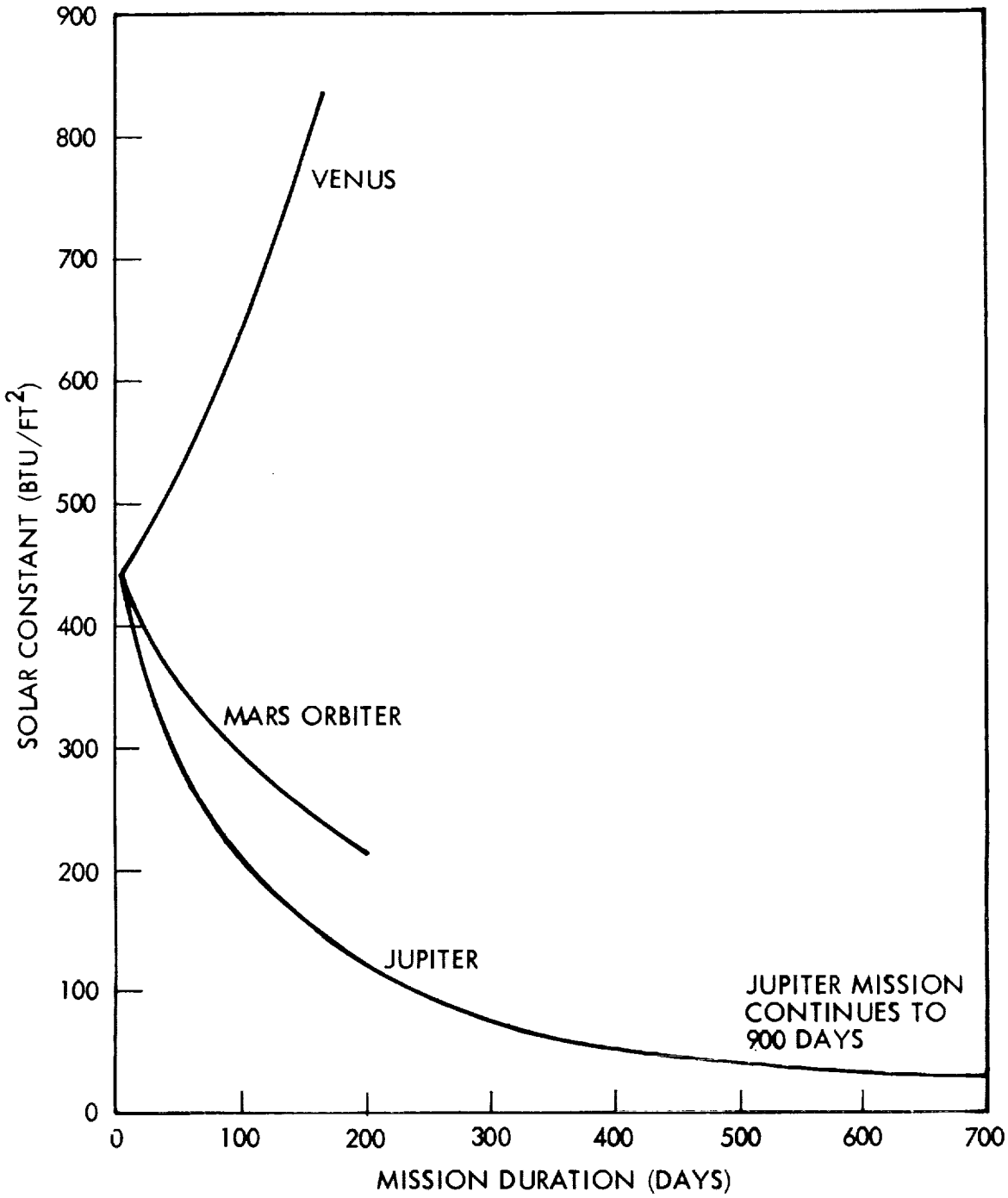


Fig. 22 Solar Constant as a Function of Mission and Duration

different, are an input to the optimization program (TOP) and in the process of optimization, interpolations are made between the two values. The expression used for interpolation between the temperature values for the limiting thicknesses is:

$$T_i = T_{min} \left[\frac{T_{max}}{T_{min}} - \left(\frac{T_{max}}{T_{min}} - 1 \right) e^{-(\Delta t_i - \Delta t_{min})} \right]$$

where

T_i	=	temperature for thickness i
T_{max}	=	steady state temperature for maximum thickness
T_{min}	=	steady state temperature for minimum thickness
Δt	=	insulation thickness

Figure 23 shows an example of the magnitude of insulation surface temperatures and the effect of insulation thickness. Average surface temperatures are shown in Figs. 24 through 26 as a function of time for F_2/H_2 and $FLOX/CH_4$ systems for all missions studied.

The multilayer insulation properties used are indicative of a composite consisting of double-aluminized mylar with Tissuglas spacer material. The assumed packing density is 100 layers/inch which gives a bulk density of 2.3 lb/ft^3. The conductivity of this insulation is given by the following expression:

$$K_{eff} = c_1 \left[1.83 \times 10^{-12} (100)^2 (T_m) + \frac{1.7\sigma \left(T_h^2 + T_c^2 \right) (T_h + T_c) t}{(N - 1)(2/\epsilon - 1)} \right]$$

In the analysis the conductivity computed with this equation was multiplied by a factor c_1 of 5 to account for installation or other degradation. This is a more conservative approach than used in earlier studies where a factor of 2.2 was used.

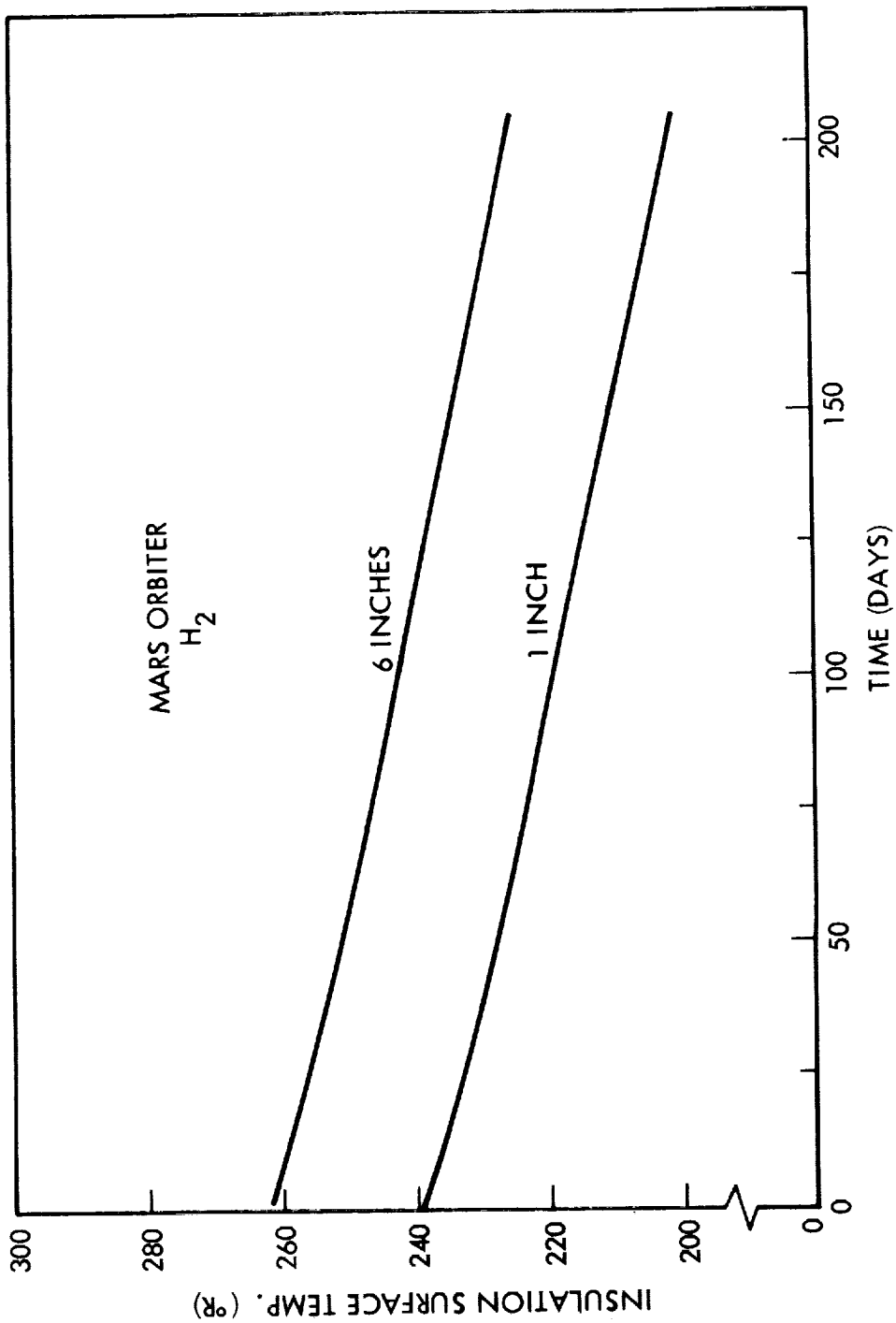


Fig. 23 Effect of Insulation Thickness on Surface Temperature

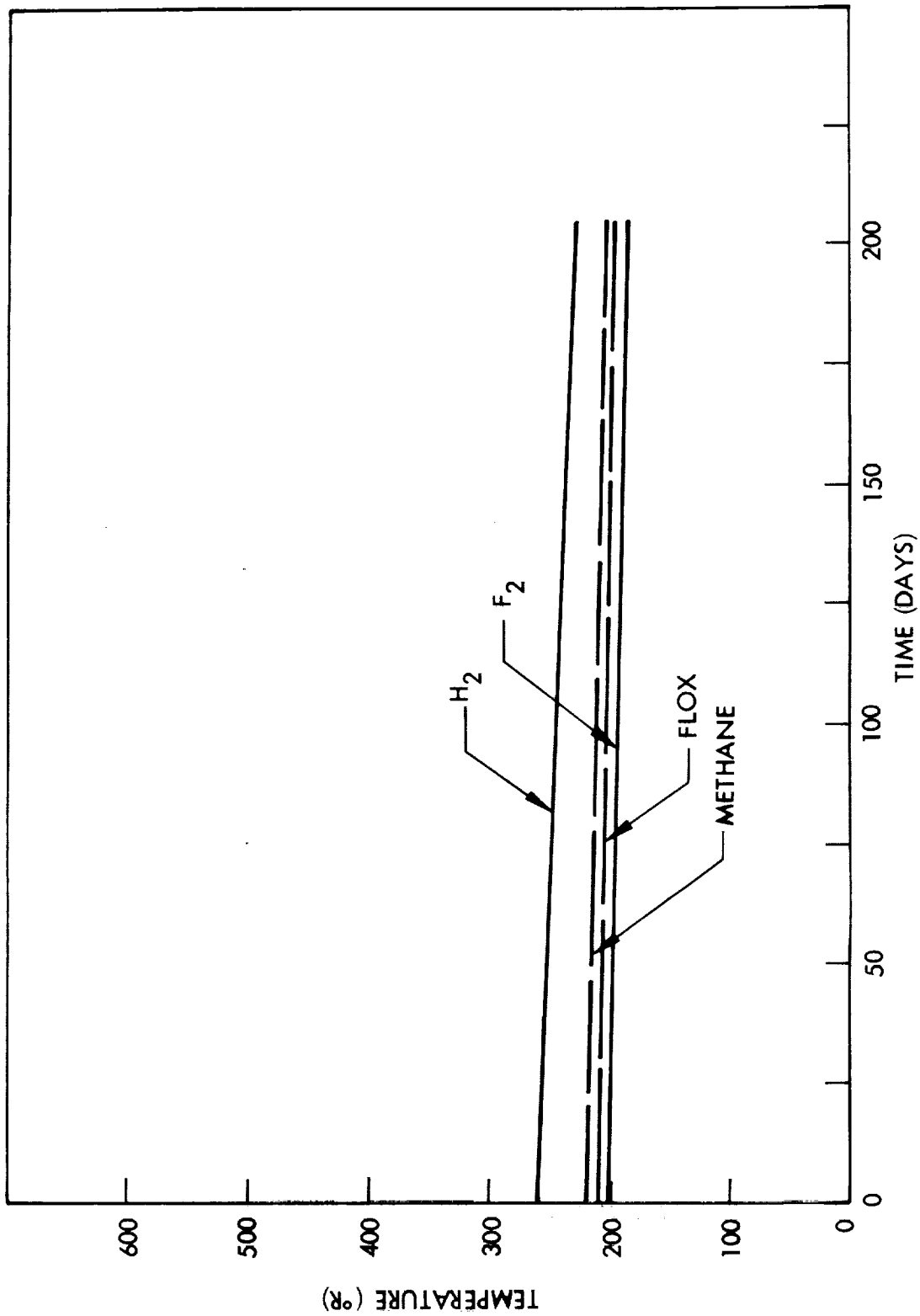


Fig. 24 Mars Orbiter Insulation Surface Temperature as Function of Time

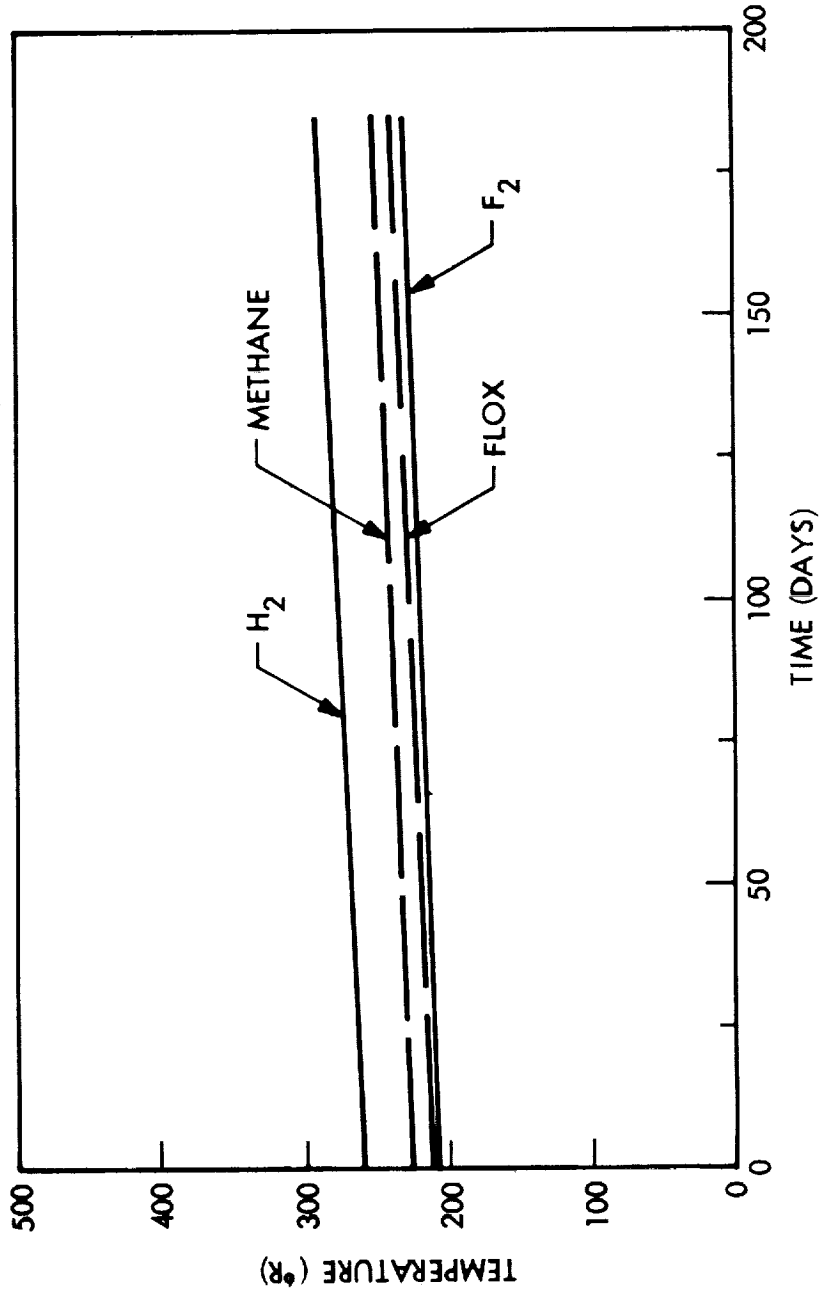


Fig. 25 Venus Orbiter Insulation Surface Temperature as Function of Time

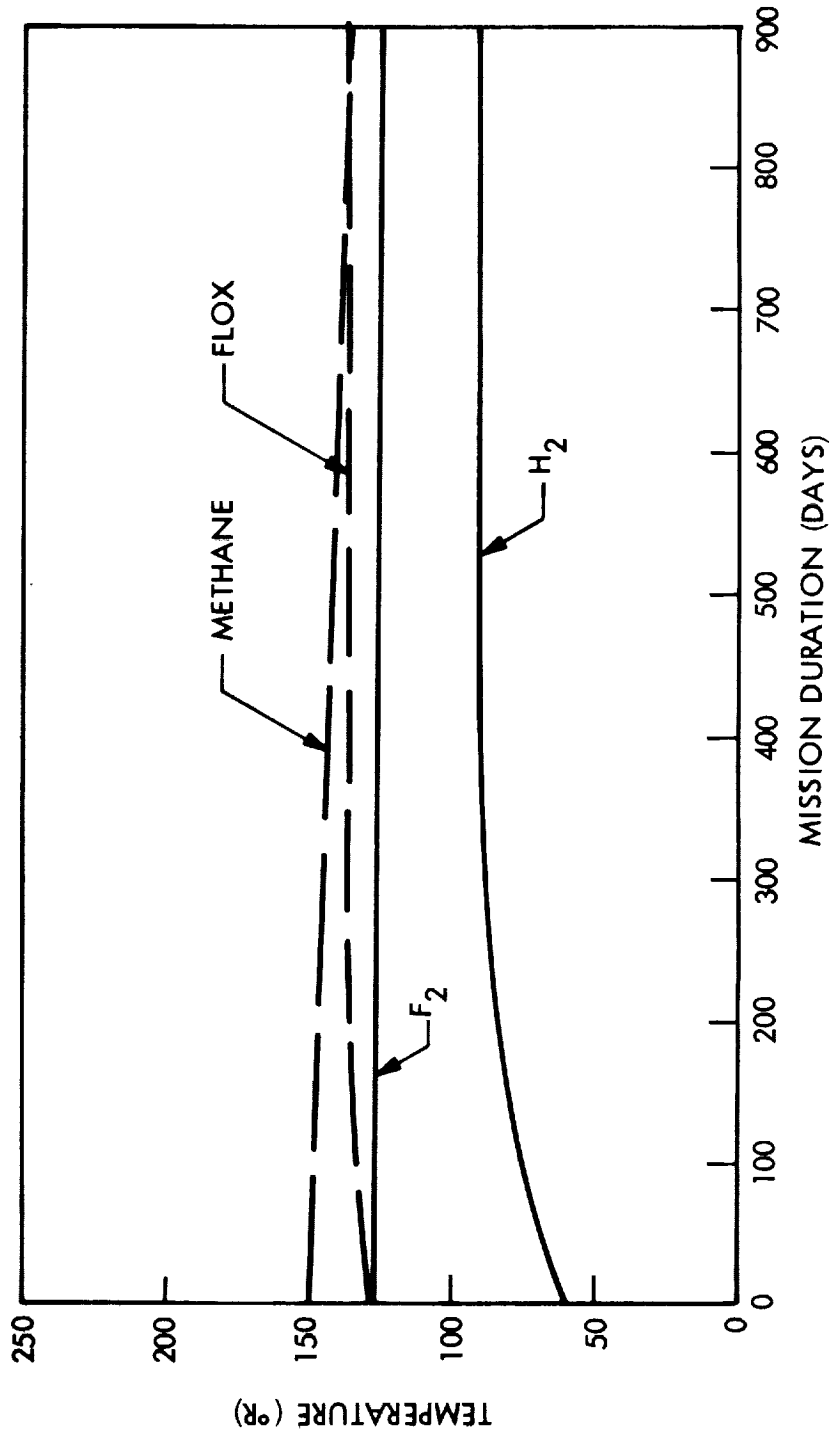


Fig. 26 Jupiter Orbiter Insulation Surface Temperature as Function of Time

where

σ = Stephen-Boltzman constant, BTU/HR-ft^2-°R^4
T_h = warm side boundary temperature (°R)
T_c = cold side boundary temperature (°R)
t = insulation thickness, ft
T_m = $(T_h + T_c)/2$,(°R)
N = total number of insulation layers
ϵ = emittance factor = 0.036

This equation is also used in the TOP program so that the temperature dependence of the insulation conductivity is accurately accounted for.

Considerable emphasis was placed on thermal model detail for the F_2/H_2 and FLOX/CH_4 systems. Less emphasis was placed on thermal modeling for N_2O_4/A-50 and the OF_2/B_2H_6 systems. Temperature data obtained from the FLOX/CH_4 study and hand calculations were employed to establish boundary temperatures for the N_2O_4 and OF_2 systems. The N_2O_4 was assumed to be oriented toward the sun; the OF_2 system away from the sun.

Approximate values of α/ϵ required on N_2O_4 and A-50 tank external surfaces were determined by hand calculation and are presented in Table 11 for all missions. Figure 27 shows the estimated external tank surface temperatures used in the N_2O_4/A-50 and OF_2/B_2H_6 optimization analysis.

Table 11

N_2O_4/A-50 TANK SURFACE CHARACTERISTICS

Mission	α/ϵ
Mars Orbiter (Baseline)	0.66
Venus Orbiter	0.41
Lunar Cargo	0.63
Jupiter Orbiter	2.2
Mars Orbiter (Titan IIID)	0.71
Mars Orbiter (W_p = 6,000 lb)	0.66

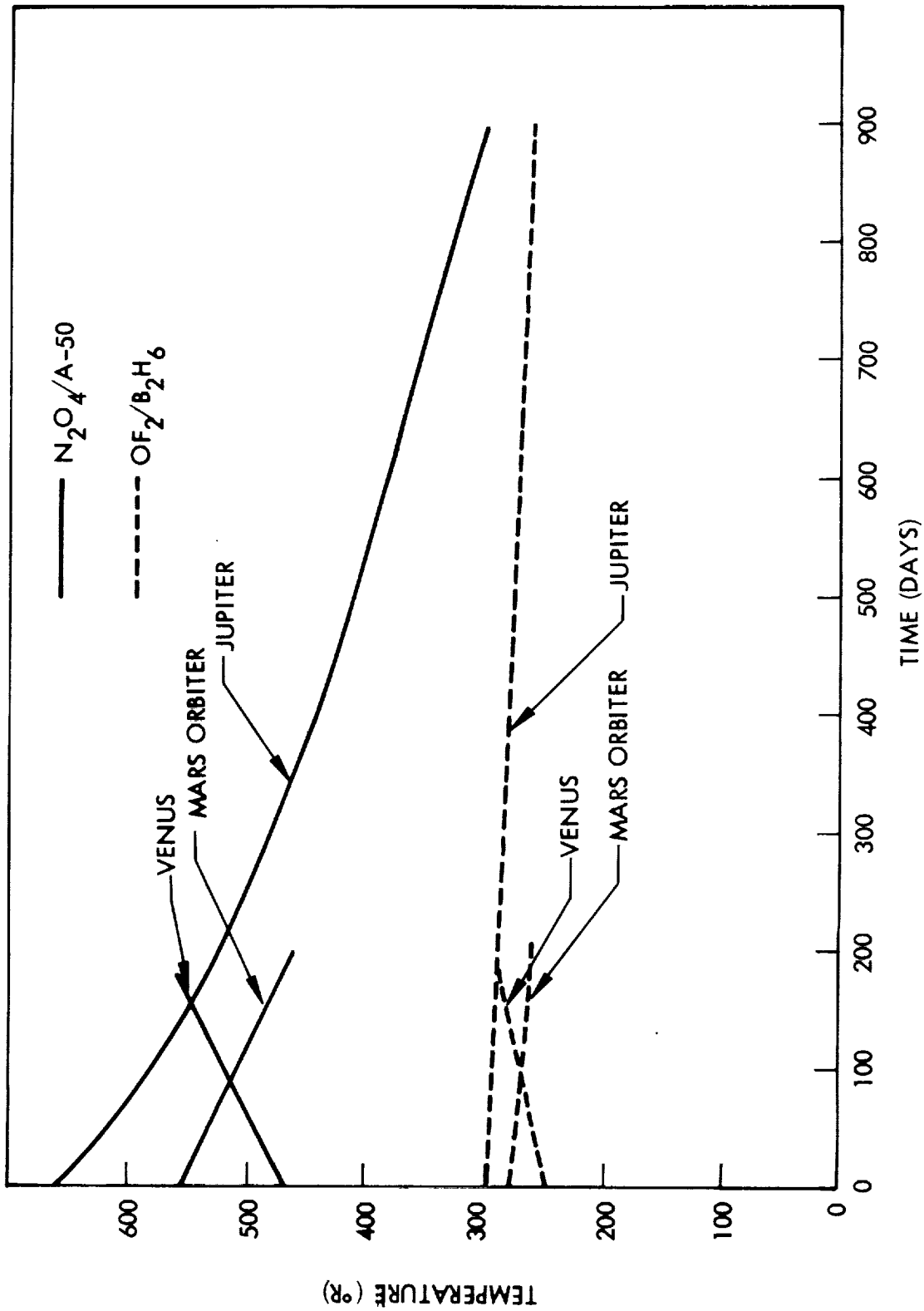


Fig. 27 Insulation Surface Temperature as Function of Time for N_2O_4/A-50 and OF_2/B_2H_6

The temperatures for OF_2/B_2H_6 must be higher than for the $FLOX/CH_4$ system and are achieved by a reduction of insulation thickness between the payload and the propulsion module. The amount of reduction in insulation and increase in heat loss from the payload have not been accurately determined. If heat loss from the payload were a problem, an alternate solution could be employed. This consists of changes in tank surface temperatures by decreasing the tank external surface emittance.

1.6.3 Thermodynamic Optimization Approach and Procedure

The thermodynamic optimization approach for the baseline Mars Orbiter vehicle is directed toward achieving a specified velocity change, ΔV, with maximum payload for a system with fixed initial total weight. Thermal effects on the weights of insulation, tanks, residual vapor, propellants, etc., are calculated to determine propellant system requirements for the lightest weight propulsion stage. The program computes performance, to check that the required velocity is met, by using the standard rocket equation

$$\Delta V = g\, I_{sp} \ln \frac{W_{initial}}{W_{final}}$$

where

ΔV	=	stage velocity increment
g	=	Earth gravitational constant
I_{sp}	=	propulsion system specific impulse
$W_{initial}$	=	initial stage weight
W_{final}	=	final stage weight at burnout

The program can be used to compute the values of parameters, listed below, which give the desired objective; i.e., maximum payload, maximum ΔV, minimum launch weight or minimum inert weight.

- Insulation thickness
- Pressurization system weights

- Ullage volume (initial)
- Tank operating pressure
- Boiloff (if any)
- Tank weight
- Launch weight
- Propellant required

The program contains two major sections; one which performs thermodynamic calculations and a second which performs the mathematical-optimization calculations. The flow diagram of Fig. 28 illustrates the relationship of input information (outside dashed line), the thermodynamic section, the optimization section and output routines. The portion within the dashed line constitutes the "TOP" (Thermodynamic Optimization Program).

The optimization analysis is conducted by an iterative process. As indicated by the general flow diagram, mission information and results of the thermal analyzer study to determine spacecraft temperature distributions are provided as input to the "TOP." In order to initiate the analysis an estimate of tank sizes and propellant requirements are computed within the program. The tank size is computed allowing for the impulse propellants, unusable liquid residuals (known), propellant for a performance reserve (normally for a percentage of ΔV), the volume occupied by pressurant spheres, if any, and ullage. All inert weights are provided as input. Inert weights take two forms; those which are fixed and those which are a function of propellant load. Based on the initially sized propulsion module the iterative analysis is started.

The input to the thermal and pressurization analysis section of the program includes the mission profile and environments. Details of the profile include the number of times the engine is started, the velocity change during each burn, the duration of the burns, time between burns, and times at which weight is dropped (i.e., ejection of payload or capsule).

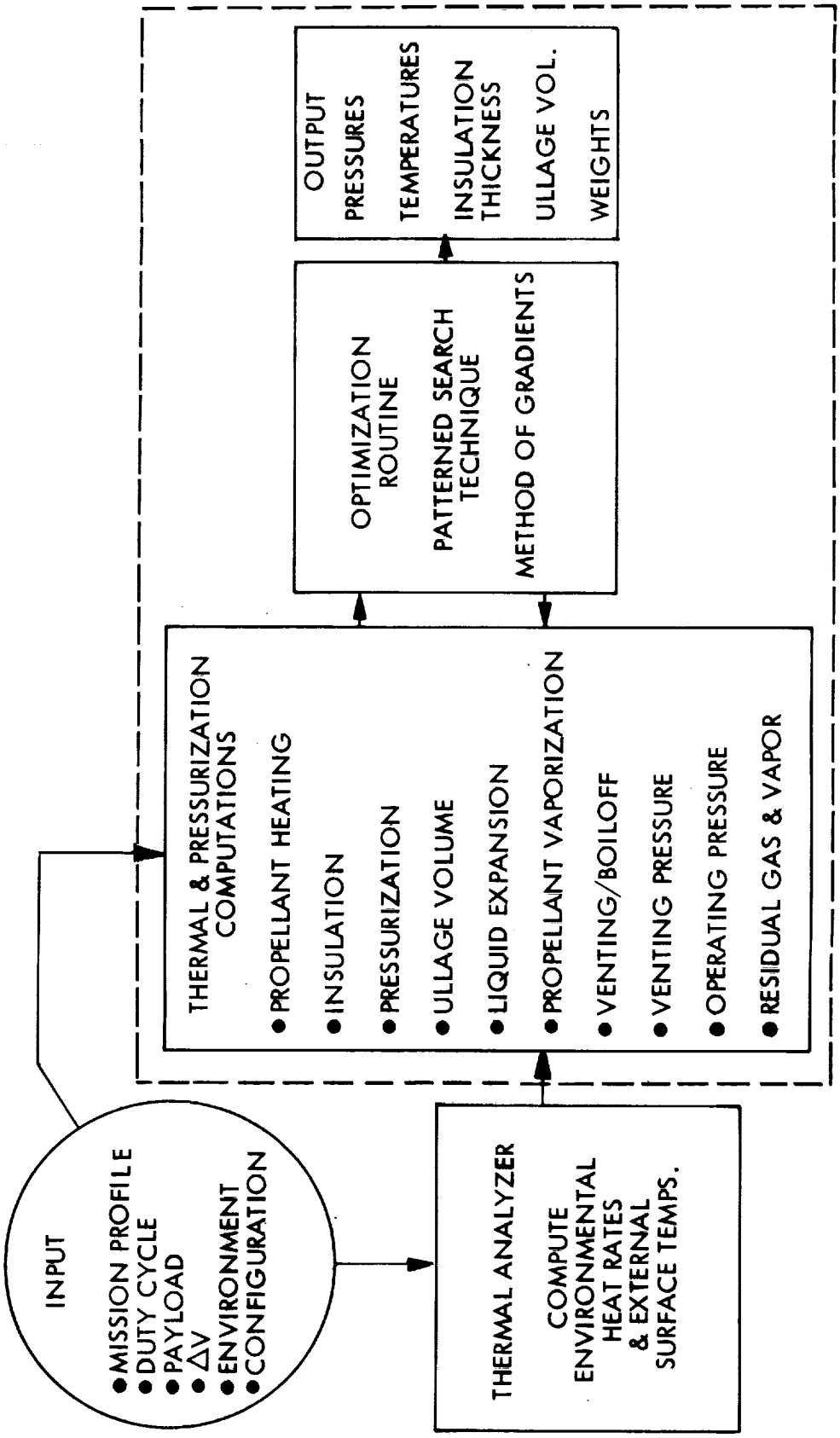


Fig. 28 Thermal Optimization Procedure

Thermal calculations are made for each phase of the mission in sequence. Figure 29 illustrates the series of thermal calculations made for a set of given conditions. As shown in Fig. 29 the thermodynamic and optimization parts of the program are iterative. For the given set of conditions, thermal calculations are made for the entire mission sequence. The performance of the system is computed after each time the thermal calculations are completed. The process of thermal and performance calculations are repeated continually until the objective is achieved; i.e., maximum payload or minimum inert weight.

An energy balance calculation routine is used to determine the propellant temperature and pressure response due to both external and internal heat inputs. External tank surface temperatures and the external temperatures corresponding to each line and penetration heat leak are obtained from the thermal analyzer output and are provided as inputs to the TOP. A subroutine in the TOP provides insulation conductivity as a function of temperature from which the thermal resistance of the tank insulation is computed. Thermal resistance values for structural and line penetrations are computed separately and provided as input to the TOP.

An energy balance subroutine is used to compute the effect of heating during the first coast phase which establishes initial conditions for the first burn. The energy balance expression, from which the propellant temperature (T_2) at the end of any given interval is obtained, is:

$$T_2 = T_1 + \frac{Q - W_\lambda h_\lambda}{(W_L - W_\lambda)C_{PL} + (W_V + W_\lambda)C_{PV} + C_x} \qquad (1)$$

where:

T_1 = liquid and ullage average temperature at initial saturated condition, °R

Q = heat transfer from external sources, i.e., insulation, structural and line penetrations, Btu

W_λ = weight of liquid vaporized, lb

h_λ = heat of vaporization, average over interval from T_1 to T_2, Btu/lb

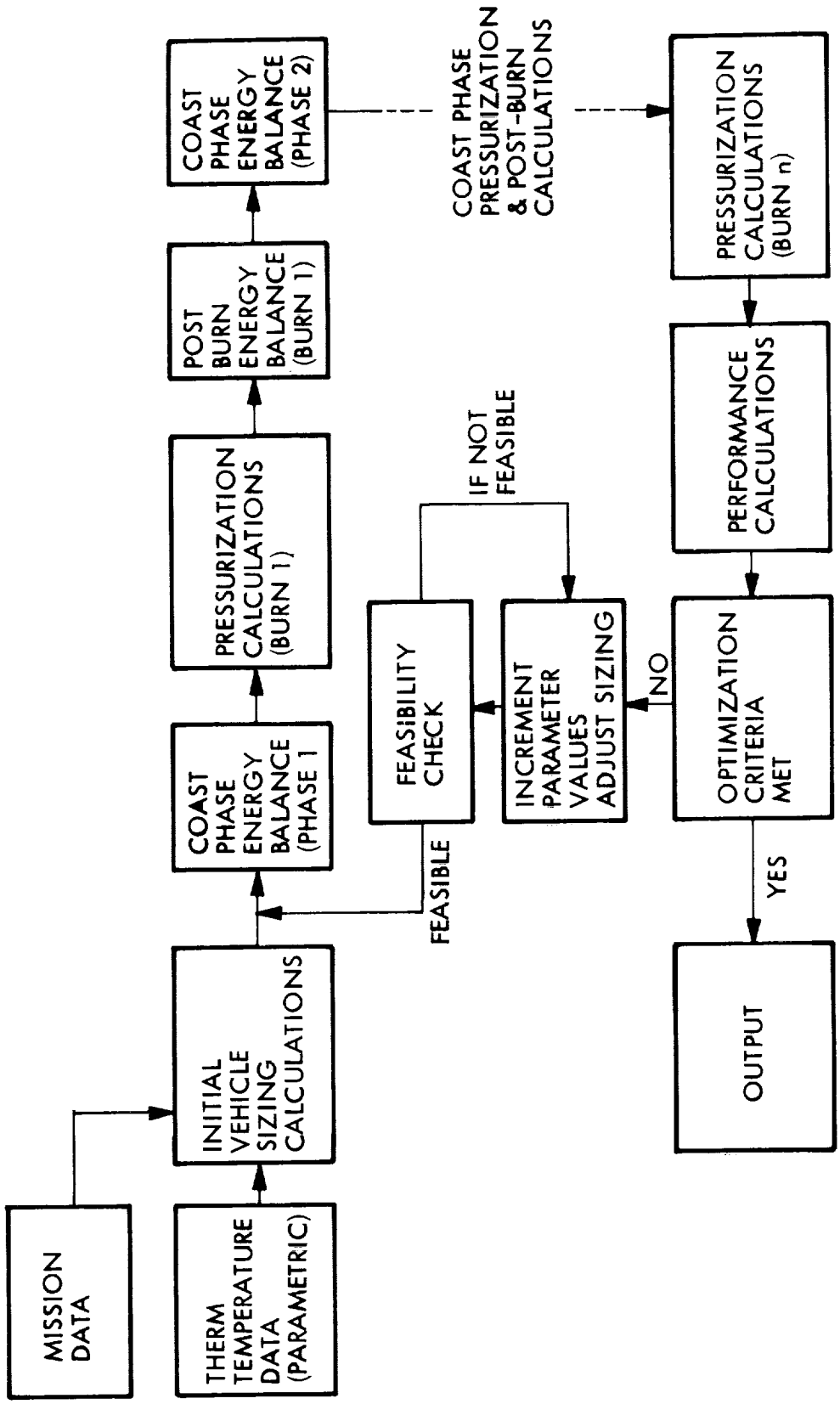


Fig. 29 Thermodynamic and Performance Calculation Sequence

W_L = initial liquid weight, lb
W_V = initial vapor weight, lb
C_{PL} = specific heat (C_P) of liquid at average of T_1 and T_2, Btu/lb °R
C_{PV} = specific heat (C_p) of vapor at average of T_1 and T_2, Btu/lb °R
C_x = thermal capacity of propellant tank wall and pressurant storage spheres in the tank, Btu/°R

The amount of liquid vaporized is computed from:

$$W_\lambda = \frac{(P_2)\, V_{ull}}{Z\, T_2\, R} - W_V$$

where

P_2 = vapor pressure at T_2, lb/ft^2
Z = compressibility factor, dimensionless
R = specific gas constant
V_{ull} = ullage volume, ft^3

The ullage volume is obtained by:

$$V_{ull} = V_T - \frac{W_L - W_\lambda}{\rho}$$

where

V_T = total tank volume, ft^3
ρ = liquid density at T_2, lb/ft^3

All propellant properties are stored in a computer subroutine as a function of temperature. All calculations are made by using the temperature-dependent property values. By so doing, liquid density is always considered a function of temperature and therefore liquid expansion or contraction is taken into account.

The manner in which the mission duty cycle is described as input to the program is shown in Fig. 30. For the mission analyzed in this study the heating from external sources was input in the form of temperatures which are constant over each of ten time segments. The TOP program calculates the Q (equation 1) based on the external temperatures and the temperature of the propellant. The engine burns are defined as events 1, 7, 9 etc., and occur at specific times as indicated in the figure. At events where there are no engine burns, the program performs a coast phase calculation only. The ΔV and I_{sp} for each burn are specified input. Thus, the propellant temperature rise (or decrease), the liquid expansion due to heating, the change in ullage volume, and amount of propellant vaporized or condensed is computed. The new bulk average liquid and ullage vapor temperature is now known.

With conditions in the tank at the time for initiation of the 1st burn known, the amount of pressurant required to provide the necessary NPSP is computed. This depends on the ullage temperature, volume, and pressurant inlet flow rate.

Two options or routines exist in the program for the computation of pressurant requirements. First, and most often used in this study is a simplified routine in which an energy balance is used to determine the effect of mixing the injected pressurant with the existing ullage vapor (and pressurant from previous burns, if any). Heat transfer to the tank wall and liquid is not computed in this routine, but is accounted for by applying an empirical collapse factor obtained from other more rigorous analyses.

The second option is the use of the Epstein correlation which allows direct calculation of heat transfer to the tank walls and liquid.

The first method has been employed. In this case it is necessary to compute the required mass of pressurant gas and resultant average ullage temperature which is necessary to establish the required NPSP before the burn is started and to maintain it during the expulsion period. The thermodynamic processes involved are constant volume for pressurizing prior to expulsion and constant pressure during expulsion.

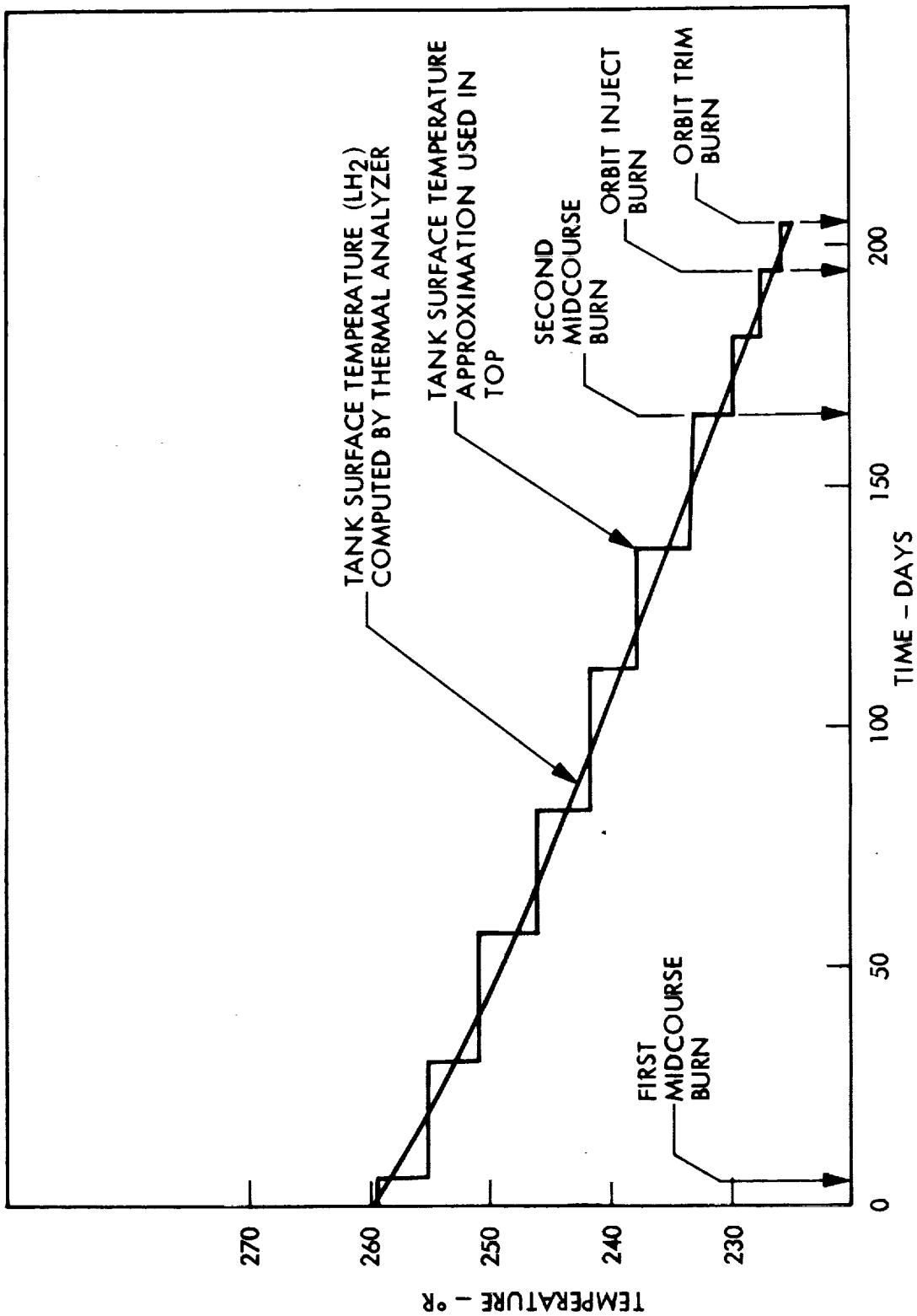


Fig. 30 Typical Tank Surface Temperatures Profile – Mars Orbiter

The pressurant mass required is computed as follows:

$$M_R = \left(\frac{\Delta P \cdot V_{ull}}{R \cdot T_{ull} \cdot Z}\right) CF \tag{2}$$

where

ΔP = increase in partial pressure of pressurant, lb/ft^2
Z = compressibility factor at T_{ull}
T_{ull} = ullage temperature, °R
CF = thermodynamic collapse factor

The mixed ullage gas temperature, T_{ull} is a function of the amount of pressurant added and therefore an iterative calculation procedure is required. The mixed ullage temperature, which does not include heat loss to the liquid or tank walls (taken into account by using collapse factor), is determined by:

$$T_{ull} = \frac{T_S (M_V C_{V_1} + M_P C_{P_1}) + M_N \cdot C_{N_1} T_{in}}{M_V \cdot C_{V_2} + M_P C_{P_2} + M_N C_{N_2}} \tag{3}$$

where

T_S = initial ullage temperature (°R)
T_{in} = pressurant inlet temperature (°R)
M_V = mass of vapor (lb)
M_P = mass of liquid (lb)
M_N = mass of pressurant (lb)
C_{V_1} = specific heat of vapor before pressurizing (Btu/lb °R)
C_{P_1} = specific heat of liquid before pressurizing (Btu/lb °R)
C_{N_1} = specific heat of pressurant before pressurizing (Btu/lb °R)
C_{V_2} = specific heat of vapor after pressurizing (Btu/lb °R)
C_{P_2} = specific heat of liquid after pressurizing (Btu/lb °R)
C_{N_2} = specific heat of pressurant after pressurizing (Btu/lb °R)

Separate calculations are performed for the period prior to liquid expulsion and during liquid expulsion. Expulsion of the propellant required to provide the appropriate ΔV occurs, and the corresponding liquid and ullage volume changes are computed.

During prepressurization the specific heats are values for constant volume process and during expulsion they are for a constant pressure process.

Immediately following the burn an energy balance subroutine is again used to determine the propellant temperature response resulting from energy added to the system from the heated pressurant and due to vaporization of some propellant to establish equilibrium with the new (larger) ullage volume.

During the expulsion period it is assumed that liquid leaves the tank and the ullage volume correspondingly increases, but that vaporization of liquid propellant into the ullage space is negligible until after the burn is complete. After the burn, liquid continues to vaporize within the tank until the ullage and liquid come to an equilibrium state. Vaporization can result in a significant decrease in liquid temperature particularly in the case where a large quantity of liquid is expelled and little liquid remains after the burn. The energy balance conducted to determine post-burn equilibrium conditions in the tank accounts for energy addition to the system resulting from the heated pressurant that was injected, and for both heat and mass transfer involved in the vaporization process.

The temperature change (decrease) and amount of liquid vaporized in bringing the liquid and ullage into vapor pressure/temperature equilibrium is computed as follows:

$$T_1 - T_2 = \frac{\Delta W \, h_\lambda}{C_{P\ell} \cdot W_2 + (W_V + \Delta W)_1 C_{PV} + C_x} \tag{4}$$

$$\Delta W = \frac{(P_2)(V_{ull})}{T_2 R \, Z} - W_V \tag{5}$$

where

T_1 = post burn liquid temperature before evaporation, °R
T_2 = post burn liquid temperature after evaporation, °R
W_1 = amount of liquid before burn, lb
ΔW = amount of liquid vaporized, lb
W_2 = amount of liquid remaining after vaporization, lb
W_V = initial vapor weight, lb
$C_{P\ell}$ = specific heat of liquid

The energy balance calculation is an iterative process. Estimates of T_2 are made and ΔW is repeatedly computed; when the estimated value of T_2 is such that ΔW from Eq. (4) and Eq. (5) match (within a given tolerance), the energy balance is satisfied.

Equation (4) results in the temperature change which is caused by vaporization within the tank. To account for the energy added to the system by injection of the heated pressurant, the energy balance equation (1) is used. In this case the value of Q is that energy added via the pressurant rather than from external heat sources.

The described series of energy balance and pressurization calculations are repeated for each coast and burn phase in the mission. This then provides tank pressure and weight for a given thickness of insulation. If the given constraint of ΔV or payload (whichever is given) is achieved, the calculation is repeated for a lesser thickness of insulation.

If at any point in the calculation an unfeasible situation results, i.e., insufficient ullage volume, the calculation is terminated and restarted with an increase in the appropriate variable which may be ullage volume, impulse propellant, or insulation thickness.

The definition of a starting point and the method used to converge on an optimum operating condition are discussed in the following paragraphs. Selection of initial values for all variables is based on judgment, prior experience or trial runs. Given a set of parameter values, the thermodynamic calculations are conducted in the sequence indicated in Fig. 29.

The basic approach to optimization is a "box" search. Boundary, or limiting values are assigned to all parameters; for example, insulation thickness limits may be set at 1/2" and 6", ullage volume at 2 percent and 20 percent and so on. The established boundaries give an n dimensional envelope within which computations will be made. An example of boundary values is shown in Fig. 31 for a simple two dimensional problem. Constraints on the system may also exist such as, within some portion of the defined region, propellant heating may tend to expand the liquid beyond the tank volume. This is a physical constraint. An initial error search band is defined for each variable which is the range over which calculations will be made. Checks are employed to determine whether a point is located within an unfeasible range. If so, calculations at the point are discontinued.

In Fig. 31 the box search approach is illustrated. The initial calculation and "box" are shown and the sequence of boxes utilized in progressing to the optimum point.

Calculations are made at nine points on each box as illustrated in Fig. 31. With n variables this results in n^3 calculations because all variables are perturbed over the applicable range. After calculations on the first box are complete, and a point other than the origin gives higher performance, the new point is selected for an origin in establishing a second box with a tolerance range of 1/2 that of the previous box. This process of selecting new points is continued until the optimum point is reached. The optimum point is defined as that which lies within given minimum allowable tolerances for all variables.

The input information required by the TOP is listed below for a typical study application.

- Total ΔV required and ΔV for each burn
- NPSP
- Thermal resistance of lines and struts
- Helium storage pressure
- Propellant mixture ratio
- Source temperatures for penetrations (output from THERM)
- Valve tolerance (adds to NPSP if applicable)
- Thrust level – used in flow rate calculation
- Liquid residuals (known unusable propellant). This is a finite thermal capacity

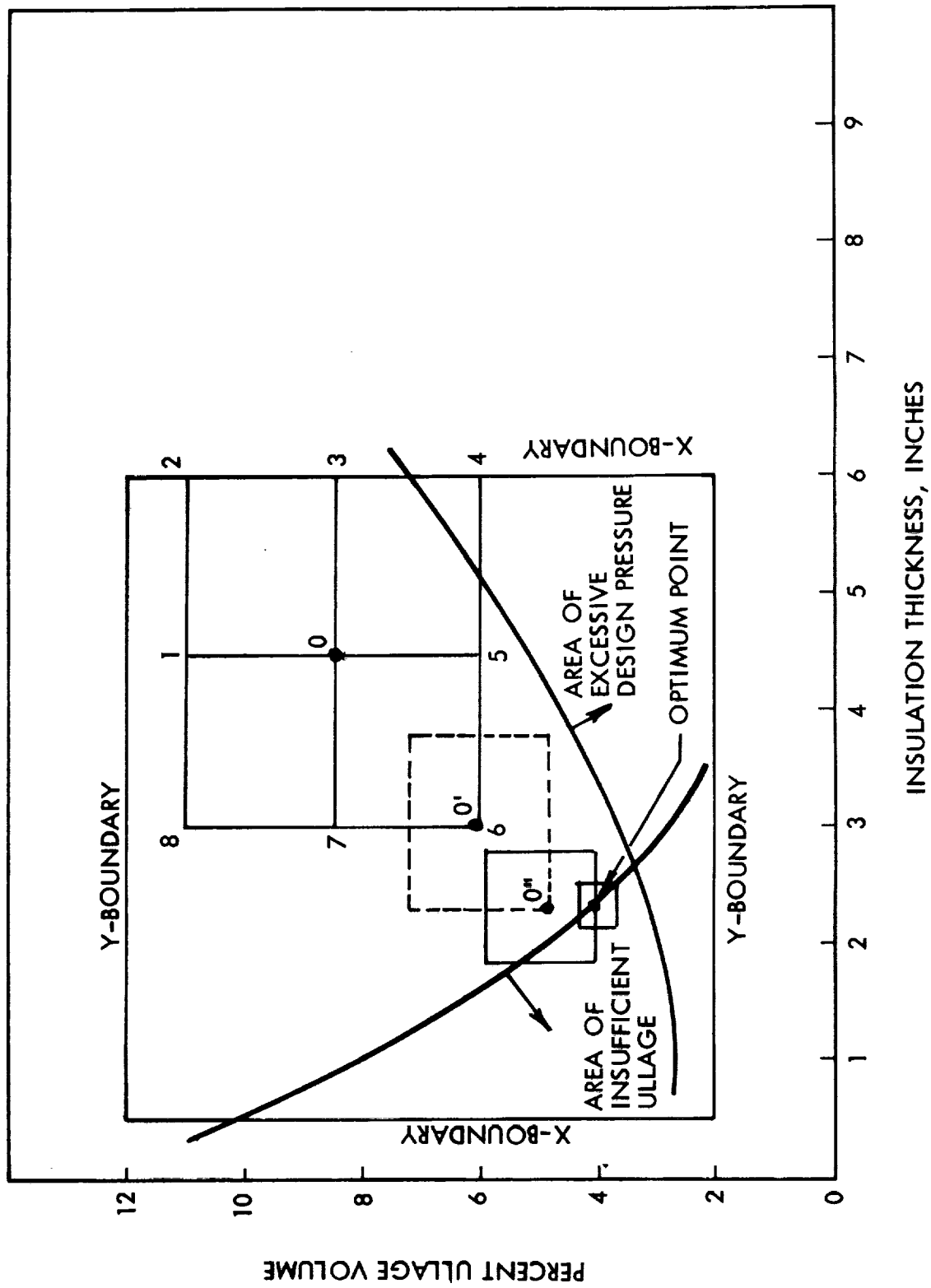


Fig. 31 Box Search Optimization

- Structural weight (constant component)
- Attachment (structural) weights
- Insulation weight – fixed portion
- Pressurization system plumbing weight (fixed portion)
- Pressurant inlet temperature
- Payload weight (if given)
- Fixed shield (including insulation) weights
- Engine system weight
- Plumbing weight (feed system)
- Propellant (type and weight)
- Number of tanks
- Storage location of He sphere
- Type of tanks (i.e., spherical, elliptical, etc.)

The program includes equations for the computation of tank weights as a function of size and pressure. Equations are also included to compute tank surface areas which are used in the computation of insulation weights. The types of tanks accommodated are spherical, cylindrical, and elliptical.

The mission duty cycle is an input in terms of event times such as the time at which burns occur, weights are dropped, etc.

The oxidizer and fuel are optimized independently by making a computer run for each. Since each run is based on optimization of stage performance the interaction of oxidizer and fuel must be included. Due to complexity and the number of total calculations involved, a method was developed to do this. For example; in a F_2/H_2 system a run is made in which all parameters involving the H_2 are varied. An estimate of the fluorine tank and all related weights is made for purposes of the performance calculations. When all parameters directly related to the H_2 are being optimized the program changes the F_2 system weights based on the mixture ratio; i.e. enough F_2 is provided to use the optimized quantity of H_2. Insulation thickness, ullage volume, residual vapor, pressurant gas, etc., are optimized in this run for the H_2 but not the F_2. Taking the output from this run (all values relating to H_2) a run is made for F_2 in which

all thermal/structural parameters for the F_2 are optimized. The two runs, if all parameters are truly optimized result in the same performance. If not, another run is made using the output of the optimization runs as initial conditions for the next run. This procedure has been very effective in the studies conducted.

When calculations have been completed for enough iterations that an optimum set of parameter values are known, they are printed out together with all corresponding inert weight and performance results. An example of an output for the F_2 and H_2 tanks of a F_2/H_2 system is shown in Tables 12 and 13, respectively. Table 14 is the output showing stage performance for the Mars Orbiter F_2/H_2 system.

Note that the output information includes the following:

- Insulation thickness
- Launch weight
- Ullage volume in percent of tank volume
- Vent pressure (if applicable). In all studies conducted under this contract, tanks were considered non-vented. The value of vent pressure shown on the computer output listing is the propellant critical pressure. This, of course, is never reached.
- Initial pressurant load – This is the optimized value of pressurant required for the specific oxidizer or fuel which is being optimized. It does not include requirements for the other tank or residual left in the He storage sphere.
- Maximum tank pressure – In the example shown in Table 12 the value indicated is simply an estimate of the maximum pressure the tank will experience. In cases where optimum tank pressures exceed minimum gage values, this maximum tank pressure is truly optimized.
- Initial propellant load – This is the optimized value of propellant loaded.

Under the Mission Particulars section of the printout the final propellant temperature is given. This is the liquid temperature immediately before the final burn.

The weight summary gives propellant tank and pressurant storage sphere dimensions, volumes and weights. The propellant tank surface areas and insulation weights are given. Weights of impulse propellant, residuals, pressurant, and vapor are given.

```
                    INTERMEDIATE OPTIMIZATION OUTPUT
                    ********************************

PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME      = .20000+01 (  %  )
        VENT PRESSURE                         = .82280+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               = .16909+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                            CALCULATION TIME          =  06.881   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .15430+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  LF2(H)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .21482+03 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .43700+02 ( PSI )
        PROPELLANT TANK                       = .66115+02 ( LBS )     PROPELLANT TANK SURFACE AREA      = .33995+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16449+01 (FEET )     RESIDUAL                          = .30692+02 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .63278+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .97121-00 ( LBS )     INSULATION                        = .32579+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .35554-00 (FEET )     PRESSURANT SPHERE WEIGHT          = .17498+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .18832-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .28790-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .22220+01
   NONOPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        PROPELLANT TANK VOLUME                = .85929+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .96295+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .30727+01 (FEET )     INSULATION                        = .41527+02 ( LBS )
        INITIAL PROPELLANT LOAD               = .31936+03 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .10817+01 ( LBS )
        PROPELLANT TANK                       = .10924+03 ( LBS )     RESIDUAL                          = .51154+01 ( LBS )
   TOTALS
        PAYLOAD                               = .50527+04 ( LBS )     LAUNCH WEIGHT                     = .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY       = .28527+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70199+04
   BURN #   1
        DELTA VELOCITY OBTAINED               = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .16375+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .60253-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .59679-01 ( LBS )
        OPERATING PRESSURE                    = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .35479+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .55537+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .26989-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .27387-01 ( LBS )
        OPERATING PRESSURE                    = .23698+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .12033+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .15948+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .76280+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .53431+01 ( LBS )
        OPERATING PRESSURE                    = .24084+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .34553+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               = .33170+03 (FT/SC)     WEIGHT OF PROPELLANT USED         = .68019+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .28000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .14737+03 ( LBS )
```

Table 12 F_2 Optimization for F_2/H_2 Mars Orbiter

```
                    INTERMEDIATE OPTIMIZATION OUTPUT
                    ********************************

PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =      2.250 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME      = .14965+02 (  %  )
        VENT PRESSURE                         = .18770+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 = .96000+02 ( PSI )
        INITIAL PROPELLANT LOAD               = .31936+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                            CALCULATION TIME              =   06.497   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .50526+02 (DEG-R)

. WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  1 LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .96331+02 ( PSI )       MAX PRESSURE FOR THIS MISSION     = .90702+02 ( PSI )
        PROPELLANT TANK                       = .10924+03 ( LBS )       PROPELLANT TANK SURFACE AREA      = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .85930+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .30727+01 (FEET )       RESIDUAL                          = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )       CALCULATED RESIDUAL PROPELLANT VAPOR= .39539+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .00000    ( LBS )       INSULATION                        = .41527+02 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # =     4 - LH2(F)PRESSURANT SPHERE INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .3555A-00 (FEET )       PRESSURANT SPHERE WEIGHT          = .17498+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .18832-00 (FT-3 )       PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .22220+01
   NONOPTIMUM TANK(S)   =  2 LF2(M)TANK(S)
        PROPELLANT TANK VOLUME                = .18638+02 (FT-3 )       PROPELLANT TANK SURFACE AREA      = .33995+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16448+01 (FEET )       INSULATION                        = .32579+01 ( LBS )
        INITIAL PROPELLANT LOAD               = .16909+04 ( LBS )       CALCULATED INITIAL PROPELLANT VAPOR = .12546-00 ( LBS )
        PROPELLANT TANK                       = .66115+02 ( LBS )       RESIDUAL                          = .30692+02 ( LBS )
   TOTALS
        PAYLOAD                               = .50550+04 ( LBS )       LAUNCH WEIGHT                     = .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )       1 % PERFORMANCE CONTINGENCY       = .29904+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70231+04
   BURN #   1
        DELTA VELOCITY OBTAINED               = .50500+02 (FT/SC)       WEIGHT OF PROPELLANT USED         = .27291+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.30976-00 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER= -.89741-01 ( LBS )
        OPERATING PRESSURE                    = .20453+02 ( PSI )       TOTAL WEIGHT OF MIXTURE USED      = .35479+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               = .17170+02 (FT/SC)       WEIGHT OF PROPELLANT USED         = .92562-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.32754-01 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER= .36378-01 ( LBS )
        OPERATING PRESSURE                    = .81899+02 ( PSI )       TOTAL WEIGHT OF MIXTURE USED      = .12033+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               = .66205+04 (FT/SC)       WEIGHT OF PROPELLANT USED         = .26579+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .33471-00 (DEG-R)       WEIGHT OF PROPELLANT VAPORIZED AFTER= .28825+02 ( LBS )
        OPERATING PRESSURE                    = .80274+02 ( PSI )       TOTAL WEIGHT OF MIXTURE USED      = .34553+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               = .33484+03 (FT/SC)       WEIGHT OF PROPELLANT USED         = .11442+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00000    (DEG-R)       WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .90702+02 ( PSI )       TOTAL WEIGHT OF MIXTURE USED      = .14875+03 ( LBS )
```

Table 13 H_2 Optimization For F_2/H_2 Mars Orbiter

PROPELLANTS :	LF2(M)+LH2(F)		
USABLE WEIGHT	.36217+04		
STRUCTURE			.25078+03
BASE STRUCTURE		.16679+03	
TANK SUPPORTS		.50919+02	
ATTACHMENTS		.15000+02	
BULKHEAD INSULATION (1")		.18000+02	
PROPELLANT FEED ASSEMBLY			.41873+03
TANKS		.24147+03	
VALVES,FILTERS,PLUMBING,ULLAGING		.37000+02	
INSULATION (FIXED AND VARIABLE)		.63043+02	
METEOROID BUMPER		.77214+02	
PRESSURIZATION SYSTEM (PLUMBING + TANKS)			.26950+02
ENGINE SYSTEM			.98000+02
INERT SUB-TOTAL			.79439+03
CONTINGENCY 10%			.79439+02
RESIDUALS			.12326+03
PROPELLANT		.66508+02	
VAPOR		.52314+02	
HE GAS		.44440+01	
PERFORMANCE RESERVE (1% ΔV)			.28527+02
IMPULSE PROPELLANTS			.36217+04
PROPULSION MODULE WEIGHT			.46473+04
PAYLOAD WEIGHT			.50827+04

Table 14 Weight Summary For Optimized F_2/H_2 Mars Orbiter

The maximum total pressure experienced is printed out in this section. This value can be the operating pressure. or if at some time between burns a higher pressure actually exists. that is defined as maximum tank pressure. Also under the weight section are given the calculated value of payload and the stage launch weight.

The fourth major heading of the otuput is a summary of data related to each burn. For each burn the following is given:

- ΔV
- The change in propellant temperature from immediately before the burn to after the burn when thermal equilibrium is established.
- Operating pressure – This is vapor pressure + NPSP. Or in any case a minimum of 20 psia total pressure
- Weight or propellant used during the burn
- Weight of propellant vaporized in re-establishing equilibrium after a burn
- Weight of both oxidizer and fuel burned.

Table 14 shows a typical summary of the program output which is the optimized stage weights and performance. This includes the

- Impulse propellant weight.
- Structural weights divided into appropriate groups some of which are fixed and some of which are a function of stage size and weight.
- Propellant feed assembly – some components (tanks) are a function of propellant load and some components such as valves are given fixed weights.
- Pressurization system – Includes sphere, plumbing, regulators, etc.
- Engine system – Fixed weight in this study.
- Contingency – All inert weights are totalled and a 10 percent contingency is added.
- Residuals – Includes liquids, vapor, and pressurization gas
- Performance Reserve – Enough propellant is included to allow for a performance reserve equivalent to 1 percent of the total ΔV.
- Impulse propellant
- Propulsion module weight
- Payload weight

The total information obtained from the thermal optimization analyses for all missions is presented in Appendix C, Computer Output of Thermodynamic Optimization. A summary of thermodynamic design variables is presented in Section 1.6.4 and a summary of performance data is presented in Section 1.7.

1.6.4 Thermodynamic Optimization Results

The thermodynamic optimization analysis results provide performance data on the basis of which a comprehensive comparison of propellants can be made. In addition, characteristics which may influence the practicality of using certain propellants for specific missions are pointed out.

Summaries of achievable performance and the values of corresponding parameters are shown for each propellant and mission in Section 1.7. Workable stages can be designed for all propellants and missions; however, the practicality of using certain propellants for certain missions varies.

Insulation thicknesses, and maximum tank pressure level requirements are shown in Table 15. The variation in the values of these parameters from mission to mission results from duty cycle effects, whether solar arrays are used, mission duration, and to some extent the environment (distance from sun). The effect of distance from the sun is very significant for configurations where the sun is directly incident on the propellant tanks (N_2O_4/A-50). When the tanks are shaded from the sun, solar array temperature levels change with distance but the impact of this variation is slight relative to the effect of distance when tanks are oriented toward the sun.

1.6.4.1 F_2/H_2 System Optimization. The F_2/H_2 system is particularly effective for all missions considered. With the vehicle oriented to shade the propellant tanks from the sun, optimum insulation thicknesses lie in practical ranges (i.e., 1 in. to 3 in. for H_2 and 1/2 in. to 5/8 in. for F_2).

The pressure levels at which the tanks optimize are generally at or slightly below the minimum gage capability for H_2 and considerably below this point for F_2. Minimum

Table 15

SUMMARY OF INSULATION THICKNESS AND MAXIMUM TANK PRESSURES

Mission	F_2/H_2				FLOX/Methane				OF_2/B_2H_6				N_2O_4/A-50			
	F_2		H_2		FLOX		CH_4		OF_2		B_2H_6		N_2O_4		A-50	
	Thk*	Press.†	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.
Mars Orbiter (Baseline)	1/2	44	2 1/4	91	1/2	65	1/2	31	1/2	241	1/2	155	1/2	160	5/8	155
Venus Orbiter	1/2	213	3	94	1/2	210	1/2	48	1/2	228	1/2	155	1/2	174	1/2	155
Lunar Cargo	1/2	33	1	25	1/2	33	1/2	33	1/2	166	1/2	166	1/2	158	1/2	167
Jupiter Orbiter	1/2	33	1	166**	1/2	41	1/2	32	5/8	226	1/2	165	5	219	4	190
Mars Orbiter w/ Titan IIID	5/8	44	2 3/8	127**	5/8	51	1/2	35	1/2	216	1/2	166	1/2	166	1/2	166
Mars Orbiter W_p = 6000 lb	1/2	44	2 7/8	98	1/2	43	1/2	35								

*Thk = Insulation thickness, inches
**Above minimum gage design pressure.
†Press. = max. Tank Pressure, psia

gage capability is exceeded only for the H_2 tank for the Jupiter and Mars Orbiter W/Titan IIID missions. A high optimum tank pressure for the Jupiter mission resulted primarily from the heating over the very long duration. The relatively high pressure for the Mars Orbiter W/Titan IIID results from the fact that a large portion of the propellant is expelled at the start of the mission and is therefore unavailable as a heat sink during the remainder of the mission

Some typical pressure profiles for F_2 and H_2 are shown in Figures 32 and 33, respectively for the Mars Orbiter mission. The fluorine tank pressure increased due to heating during the coast phases. A substantial total pressure rise occurs immediately after the first midcourse burn because the tank is pressurized with non-condensible helium and the helium partial pressure is greater than the NPSP requirement. In the case shown, the helium partial pressure is great enough to provide the required NPSP throughout the second midcourse burn expulsion without adding more helium. A propellant partial pressure sensing device is required. Note that the divergence between F_2 vapor pressure and total pressure occurs as a result of liquid expansion due to heating. Also note a considerable drop in F_2 liquid temperature following the orbit injection burn where about 90% of the propellant is expelled. This temperature drop results from the vaporization of F_2 to bring the entire tank ullage volume back into partial pressure equilibrium following the burn.

The H_2 tank pressure profile shown is for an autogenous H_2 gas bleed pressurization system. The required NPSP is provided each time a burn occurs by injecting heated GH_2. Thermal equilibrium is re-established following each burn.

Review of the thermal analysis results for the F_2/H_2 system show different characteristics for different missions. For all missions, except the Jupiter orbiter, heating of the H_2 tank by heat transfer through the insulation is greater than through penetration heat leaks. In the Jupiter mission, where no solar panels are employed, the H_2 tank heating is predominately from structural heat leaks. As a result the H_2 tank pressure

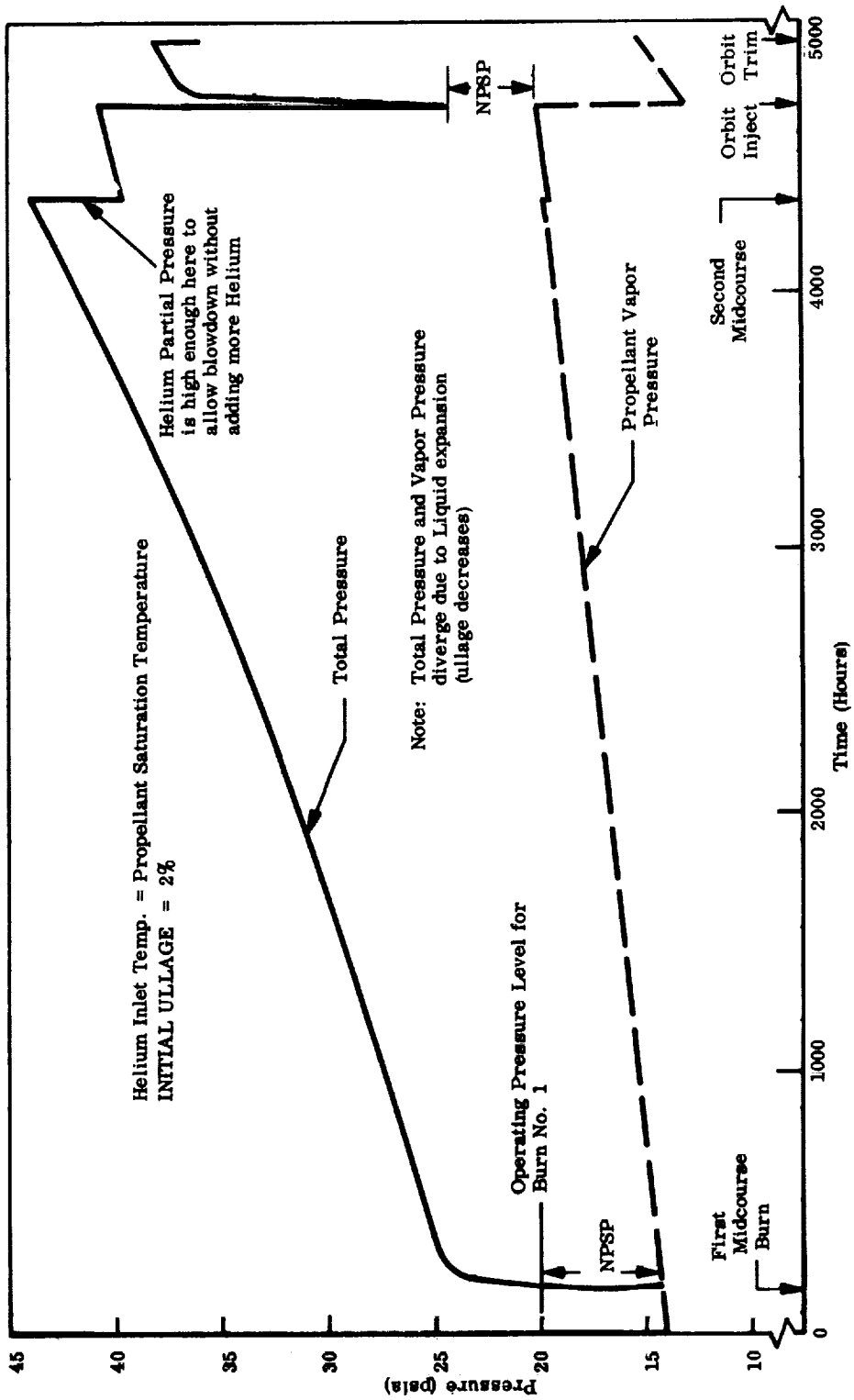


Fig. 32 F_2 Pressure Profile for Mars Orbiter

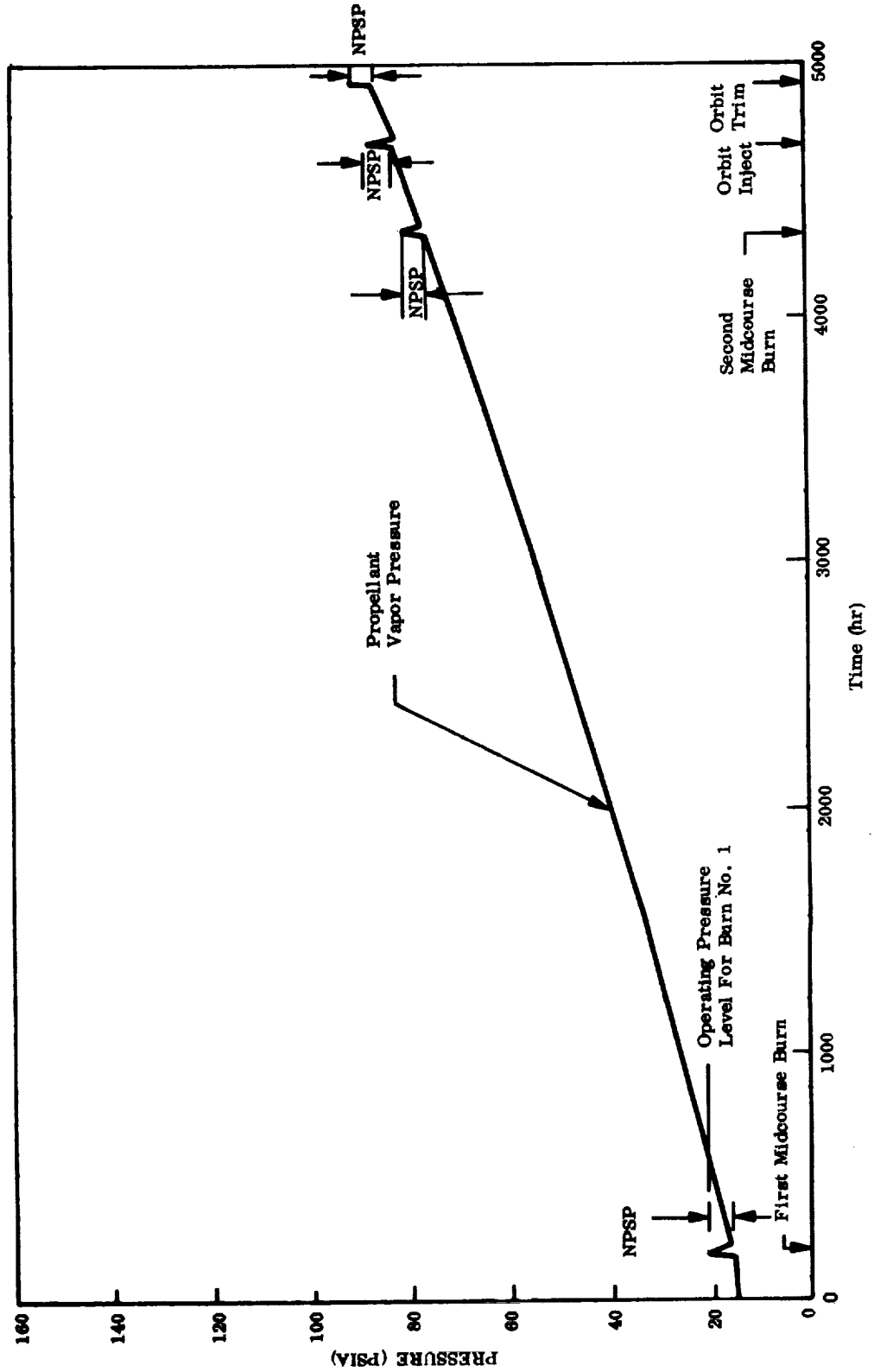


Fig. 33 H_2 Pressure Profile for Mars Orbiter

cannot be controlled significantly by adding insulation. In this study the importance of accurately evaluating the conduction heat leaks from the connection of the propulsion module and payload became evident. As a result, the analysis of the H_2 tank strut heat leaks was refined to take into account radiation losses from the surface of the insulation applied to the struts. In the Jupiter configuration with no solar panels consideration of this radiation loss reduced the heat leak substantially. Figure 34 shows the ratio of actual heat transferred to the tank through the strut relative to heat transfer computed neglecting radiation to or from the insulation surface. With 1/4 inch of insulation on the struts, a practical minimum for ground hold and maneuvers, heat transfer to the tank through the struts is only about 16 percent of the heat transfer computed while neglecting radiation from the strut insulation surface. With a configuration which includes solar panels, this effect is much less significant because the insulation surface temperature is controlled largely by heat transfer from the panels.

1.6.4.2 FLOX/Methane System Optimization. This system proves to be particularly well suited thermally to all the space environments considered. With the vehicle oriented to shield the propellant tanks from the sun the optimum insulation thicknesses vary from 1/2 to 1 inch, a very practical range. Maintaining liquid in the feed line during long periods is not a problem because the engine and other hardware which would normally be a source of energy to the feed line are very low in temperature.

The pressure levels at which these tanks optimize are generally below the minimum gage pressure capability. Presure profiles for FLOX and methane are shown in Figs. 35 and 36. The FLOX pressure profile is similar to that for F_2 in that the vapor and total pressure generally increases with time due to heating. The methane pressure profile remains relatively flat throughout the mission because it is virtually in thermal equilibrium. The only significant variations in the methane pressure level results from the effects of the engine burns.

1.6.4.3 OF_2/B_2H_6 System Optimization. This system is well suited to the environment for all missions. Previous analyses have shown this to be the case for tanks

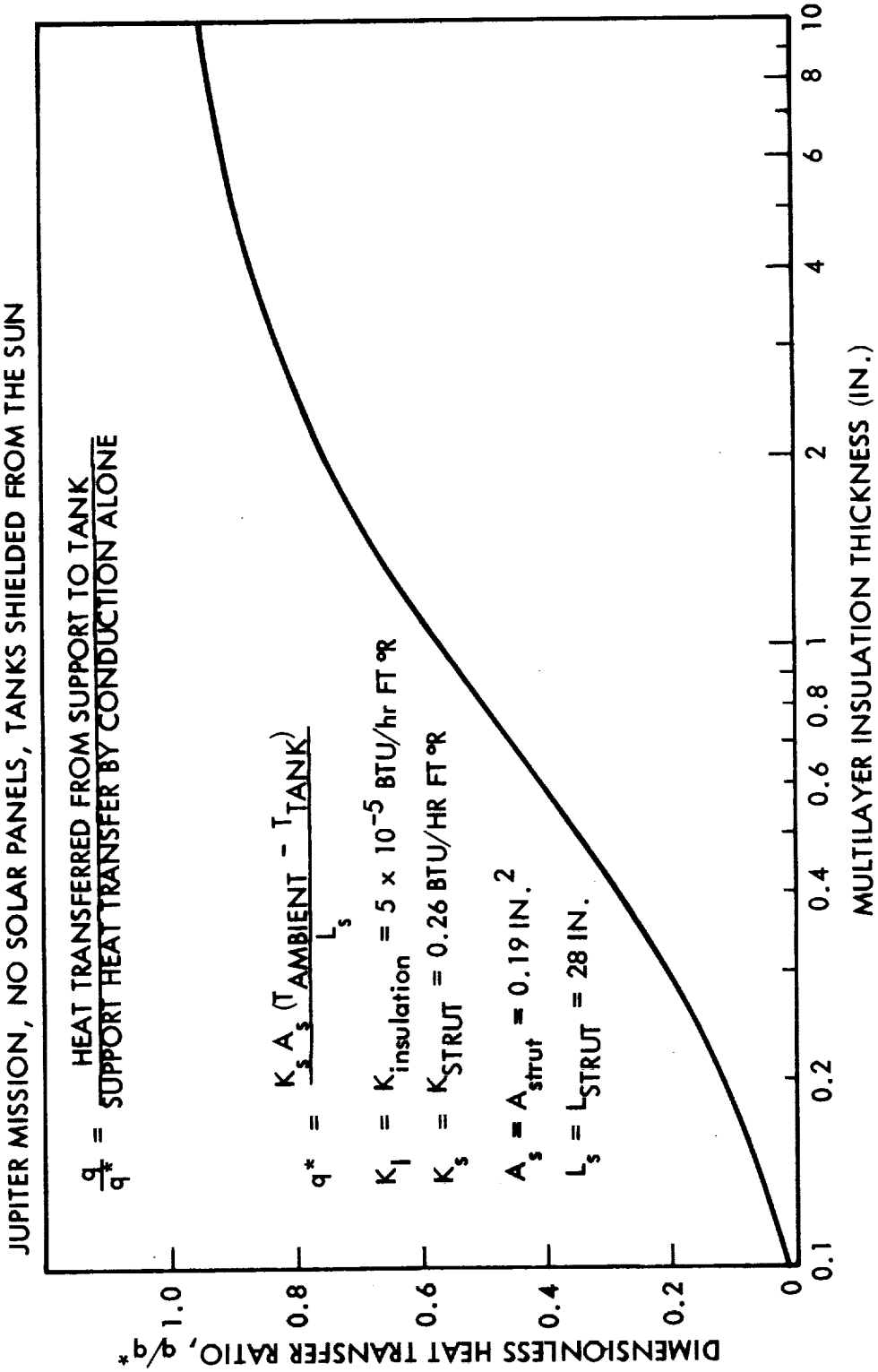


Fig. 34 Effect of Radiation Losses on Support Strut Heat Transfer

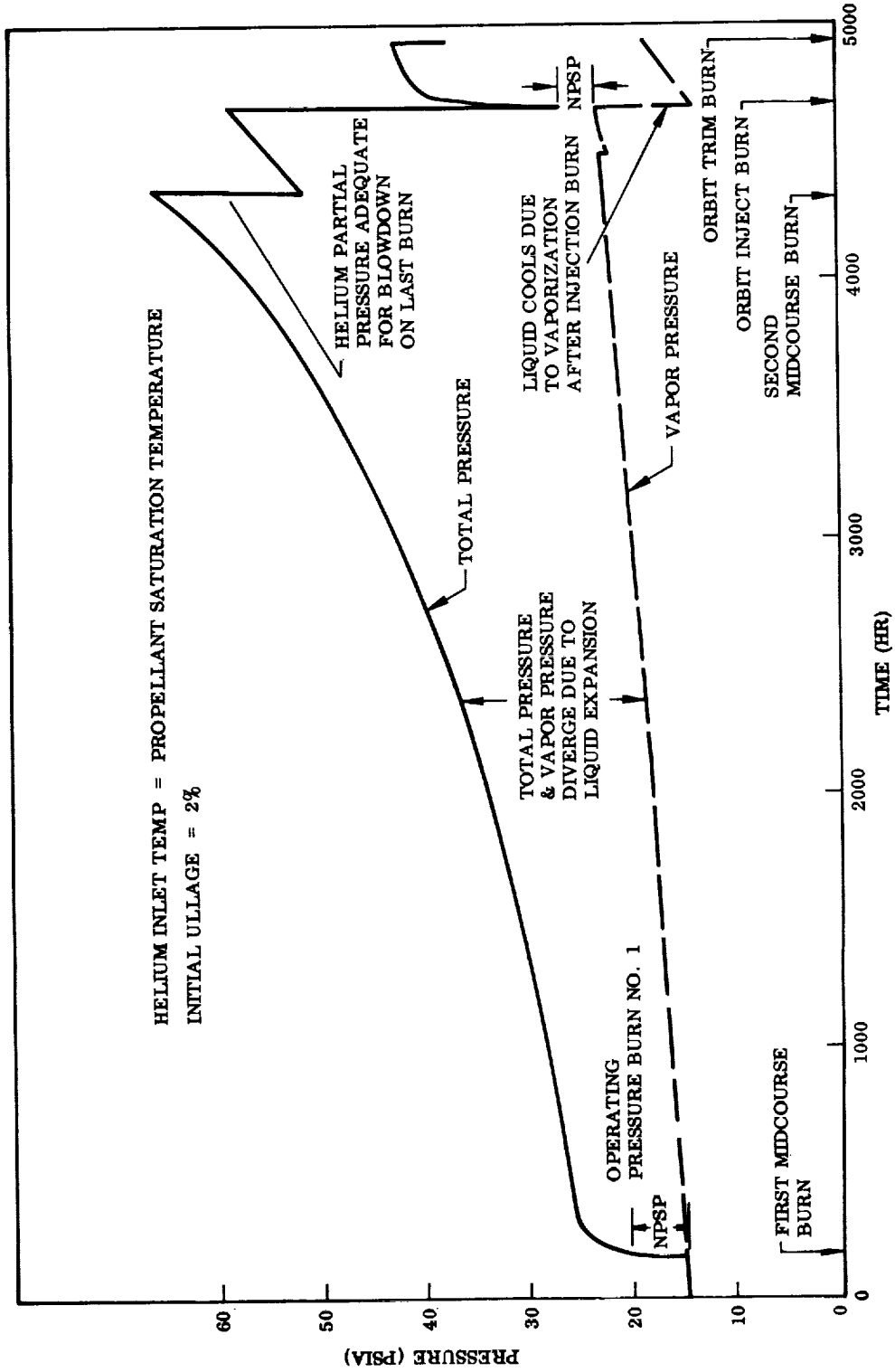


Fig. 35 FLOX Pressure Profile for Mars Orbiter

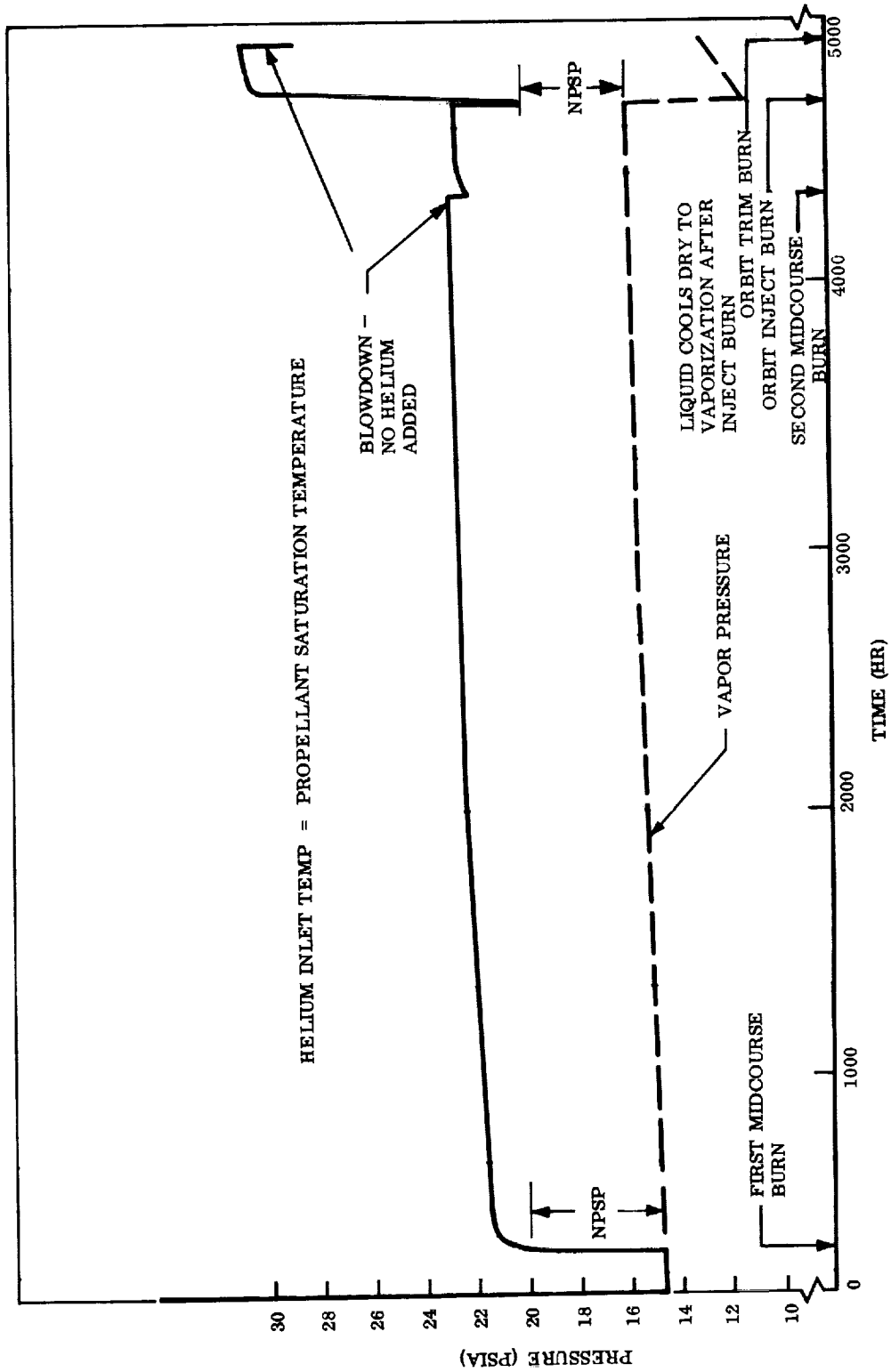


Fig. 36 Methane Pressure Profile for Mars Orbiter

oriented toward the sun or shaded from it. Performance is improved, however, if the tanks are shaded. In this study it was determined that thermal isolation of the tanks from the payload section and solar arrays needs to be carefully controlled to establish tank temperature levels which will effectively maintain the propellants in or nearly in temperature equilibrium. Results of the optimization analyses show that insulation requirements and pressure levels are nominal.

1.6.4.4 N_2O_4/A-50 System Optimization. The N_2O_4/A-50 propellant system is relatively well suited to the environments of all missions except Jupiter. Insulation requirements and pressure levels lie in very practical ranges except for the Jupiter mission. In all cases the N_2O_4/A-50 tanks were assumed oriented toward the sun. Thermal control surface properties can be selected such that external surfaces do not get too warm near Earth or Venus nor too cold at Mars. Values of α/ϵ used are shown in Table 11. For Mars missions, α/ϵ ratios were selected to give external surface temperature higher than the liquid temperature when near earth and lower than the liquid temperature when near Mars. For the Venus mission an α/ϵ was selected which gives surface temperatures lower than the liquid at Earth and higher at Venus.

Because of the rather large change in solar constant between Earth and Jupiter (443 Btu/hr-ft^2 to 16 Btu/hr-ft^2) passive thermal control techniques are not practical. To prevent freezing the N_2O_4/A-50 either five inches of insulation or active propellant heating are required.

1.6.5 Thermodynamic Design and Analysis Conclusions

Thermal analyses conducted during the course of this study have been performed in sufficient depth to draw specific conclusions about the space storability and use of the candidate propellants for the missions considered. The resulting data can be effectively employed in making a comprehensive comparison of the candidate propellants. Specific conclusions are:

- F_2/H_2, FLOX/CH_4, and OF_2/B_2H_6 can be effectively employed for all missions studied.

- N_2O_4/A-50 is well suited to all mission environments except that for Jupiter. Unless active thermal control measures are used for the Jupiter mission, insulation thicknesses required to prevent propellant freezing became impractically large.
- Insulation thicknesses are nominal (less than 3 in.) for all propellants (except N_2O_4/A-50 at Jupiter).
- Solar panels have a very significant propellant heating effect.
- Ullage volumes required are very small (2%) for all propellants except for OF_2 and H_2 which require 8 and 15% respectively for the baseline mission.
- Presently available thermal control surfaces are satisfactory. Further development of low α/ϵ surfaces would be beneficial if tanks were oriented toward the sun.
- Radiation shields are very effective in reducing propellant heating.

1.6.6 Thermodyamic Analysis and Development Recommendations

- Additional work should be directed at development of effective prelaunch thermal control systems.
- Analyses conducted under this contract assumed the use of a pressure control device which at any burn provides only enough pressurant to make up the required NPSP. Such a pressure control device should be developed.
- Additional analytical studies should be performed to investigate more thoroughly engine heat soak back and start mode effects.
- Additional analytical studies should be performed to optimize shadow shields, particularly between the solar arrays and the propellant tanks.

1.7 PERFORMANCE ANALYSIS

The performance analysis includes not only the determination of payload weight, propulsion module weight, and propellant load, but also the design parameters as determined from the thermodynamic/structural/performance optimization analysis. The Thermodynamic Optimization Program (TOP) was used for the performance calculations. The ability of the program to handle several sequential propulsive burns, with varying specific impulse and variable launch weight, provided great flexibility in the optimization at considerable savings in time. The performance is calculated by inputting parametric data for subsystems such as propellant tanks whose weights vary with propellant load. Propellant tank weights vary both with size and with pressure above the minimum gauge values. The weights of fixed systems such as engines are also included as input, as are the mission parameters such as time of burn, value of ΔV, and values of specific impulse. Thermodynamic data relating to insulation, pressurization, etc., are included parametrically and weights and dimensions uniquely defined only after the optimization has been completed.

For each propellant combination the optimization procedure is first conducted using the baseline Mars Orbiter mission profile and an injection weight of 9700 pounds. This weight includes the payload, propellant load, and propulsion module inert weight. Subsequent to the optimization and definition of the propulsion module for each propellant, the Mars Orbiter propulsion module is flown on the alternate missions. The only changes allowed are in insulation thickness, pressurization requirements and meteoroid shield, which are optimized for each mission. The injection capability of the Titan IIID/Centaur varies for each mission and poses a further constraint on the overall system weight. The performance results are classified as to single primary burn systems such as planet orbiter mission and those systems which also incorporate ascent burn. For the ascent burn system the initial staging weight is determined parametrically so as to optimize the payload, and varies for each propellant.

The following paragraphs present summary weight statements for each stage and mission. The detailed weights are presented in Appendix C where the actual computer output is reproduced.

1.7.1 Single Primary Burn Stage Performance

The single primary burn systems include the Mars Orbiter, Venus Orbiter, Lunar Cargo, Jupiter Flyby, Grand Tour of the Jovian Planets, and 0.2 A. U. Solar Probe.

The baseline mission is the Mars Orbiter with the propulsion module sized for a Titan IIID/Centaur booster and predetermined mission profile. The Mars orbiter propulsion stage is then flown on the other missions, with minor mission-peculiar changes, and the performance and operating conditions determined. The mission requirements and booster capability were presented earlier in Section 1.5.2.1.

1.7.1.1 Mars Orbiter Stage Weights and Design Parameters. Summary weight statements for the F_2/H_2, $FLOX/CH_4$, OF_2/B_2H_6, and N_2O_4/A-50 systems are shown in Table 16. These weight statements give a breakdown of all of the inert weight elements, the impulse propellant, the total propulsion module, and the payload. The summary weight items shown in Table 16 and subsequent tables are made up of the following:

Structure:
- Base Structure, Tank supports, attachments, and bulkhead insulation

Propellant feed assembly:
- Tanks, valves, filters, plumbing, ullaging, insulation, meteoroid shield

Pressurization System:
- Tanks and plumbing

Engine system: no breakdown

Contingency: 10% of dry inert weight

Residuals:
- Propellant
- Vapor
- He gas

Performance Reserve: 1% on total ΔV

Inert: sum of above weights

Table 16

WEIGHTS AND DESIGN PARAMETERS FOR BASELINE MARS ORBITER STAGE
(Weights in Pounds)

Item	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	419	381	406	381
Pressurization System	27	27	59	79
Engine System	98	98	153	151
Contingency	79	68	80	82
Residuals	121	86	100	93
Performance Reserve	28	31	31	34
Inert	1023	868	1007	1030
Impulse Propellant	3622	4044	4085	4868
Propulsion Module	4645	4912	5092	5898
Payload	5055	4788	4608	3802

Design Parameters

Propellant	Tank Pressure (psia)	Insulation Thickness (inches)	Initial Ullage (percent)
F_2	44	1/2	2
H_2	91	2-1/4	15
FLOX	65	1/2	2
CH_4	31	1/2	2
OF_2	241	1/2	8
B_2H_6	155	1/2	2
N_2O_4	160	1/2	2
A-50	155	5/8	2

Impulse Propellant: Propellant used for ΔV

Propulsion Module: Inerts plus impulse propellant

Payload: All mass carried by the propulsion stage

In addition to the weights, the design parameters such as maximum propellant tank pressure, insulation thickness, and initial ullage values are shown. These parameters have all been carefully optimized in order to maximize the payload.

1.7.1.2 Venus Orbiter Stage Weights and Design Parameters. Subsequent to the optimization for the Mars Orbiter mission, the baseline propulsion stage is flown on the alternate missions such as the Venus Orbiter. The propellant tank sizes and gauges are fixed, but the pressurization spheres, insulation thickness, initial ullage, and meteoroid shielding are allowed to vary in the optimization procedure. The Venus Orbiter, because of its flight path, receives increasing solar radiation during its mission. The effect of this is to increase the tank pressures. For hydrogen the ullage volume and insulation thickness were also increased. Summary weight statements and design parameters are shown in Table 17 for the four propellants.

1.7.1.3 Lunar Cargo

The lunar cargo mission, because of its short duration, poses no special problems. Table 18 presents the weight statements and design parameters for the four propellants.

1.7.1.4 Direct Ascent Missions

The direct ascent missions do not pose any additional propellant handling and storing problem. In general, they are simpler and ground operations dictate the thermal design features that need to be incorporated. The design conditions for these missions and propellants are shown in Table 19. Figure 37 shows the generalized performance curve for the four propellant combinations and notes the payload capability for the Jupiter Flyby, Grand Tour of Jovian Planets, and 0.2 A.U. Solar Probe.

Table 17

WEIGHTS AND DESIGN PARAMETERS FOR VENUS ORBITER STAGE

(Weights in Pounds)

Item	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	428	378	402	376
Pressurization System	27	29	58	68
Engine System	98	98	153	151
Contingency	80	68	79	80
Residuals	82	69	81	89
Performance Reserve	28	30	31	33
Inert	994	849	982	1007
Impulse Propellant	3620	4037	4088	4869
Propulsion Module	4614	4886	5070	5876
Payload	4779	4505	4352	3555

Design Parameters

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage (percent)
F_2	213	1/2	2.4
H_2	94	3	22.4
FLOX	210	1/2	2.6
CH_4	48	1/2	2
OF_2	228	1/2	8.5
B_2H_6	155	1/2	2
N_2O_4	174	1/2	2
A-50	155	1/2	2

Table 18

WEIGHTS AND DESIGN PARAMETERS FOR LUNAR CARGO STAGE

(Weights in Pounds)

Item	F_2/H_2	$FLOX/CH_4$	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	351	353	371	350
Pressurization System	26	25	60	69
Engine System	98	98	153	151
Contingency	73	65	76	78
Residuals	81	74	87	94
Performance Reserve	26	28	28	30
Inert	906	820	953	982
Impulse Propellant	3632	4058	4083	4868
Propulsion Module	4538	4878	5036	5850
Payload	3434	3170	2995	2317

Design Parameters

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage (percent)
F_2	33	1/2	2
H_2	25	1	23.3
FLOX	33	1/2	2
CH_4	33	1/2	2.8
OF_2	166	1/2	8.7
B_2H_6	166	1/2	2
N_2O_4	158	1/2	2
A-50	167	1/2	2

Table 19

DESIGN PARAMETERS FOR DIRECT ASCENT MISSIONS

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage* (percent)
F_2	20	1/2	2
H_2	20	1	23
FLOX	20	1/2	2
CH_4	20	1/2	2.8
OF_2	155	1/2	8.7
B_2H_6	155	1/2	2
N_2O_4	155	1/2	2
A-50	155	1/2	2

*Dictated by fixed tank sizes.

1.7.2 Ascent Burn Stage Performance

The ascent burn missions include the Jupiter Orbiter and the Titan IIID (no Centaur) boosted Mars Orbiter for all propellant combinations, and the 6000-pound propellant load Mars Orbiter mission with F_2/H_2 and FLOX/CH_4 propellants. The baseline Mars Orbiter propulsion stage is used for the Jupiter Orbiter mission and Titan IIID (no centaur) Mars Orbiter while new stages are optimized for the 6000-pound propellant load Mars Orbiter vehicles.

1.7.2.1 Jupiter Orbiter Stages Weights and Design Parameters. The Jupiter Orbiter is the most demanding of all the missions evaluated. Not only is more than half of the propellant utilized during the ascent burn, but the long mission duration produces a significant amount of conductive heating to the cryogenic propellants. This effect is such that the F_2/H_2 commonality stage must be modified slightly for the Jupiter orbiter mission. Weight statements and design parameters are shown in Table 20 for the four propellants.

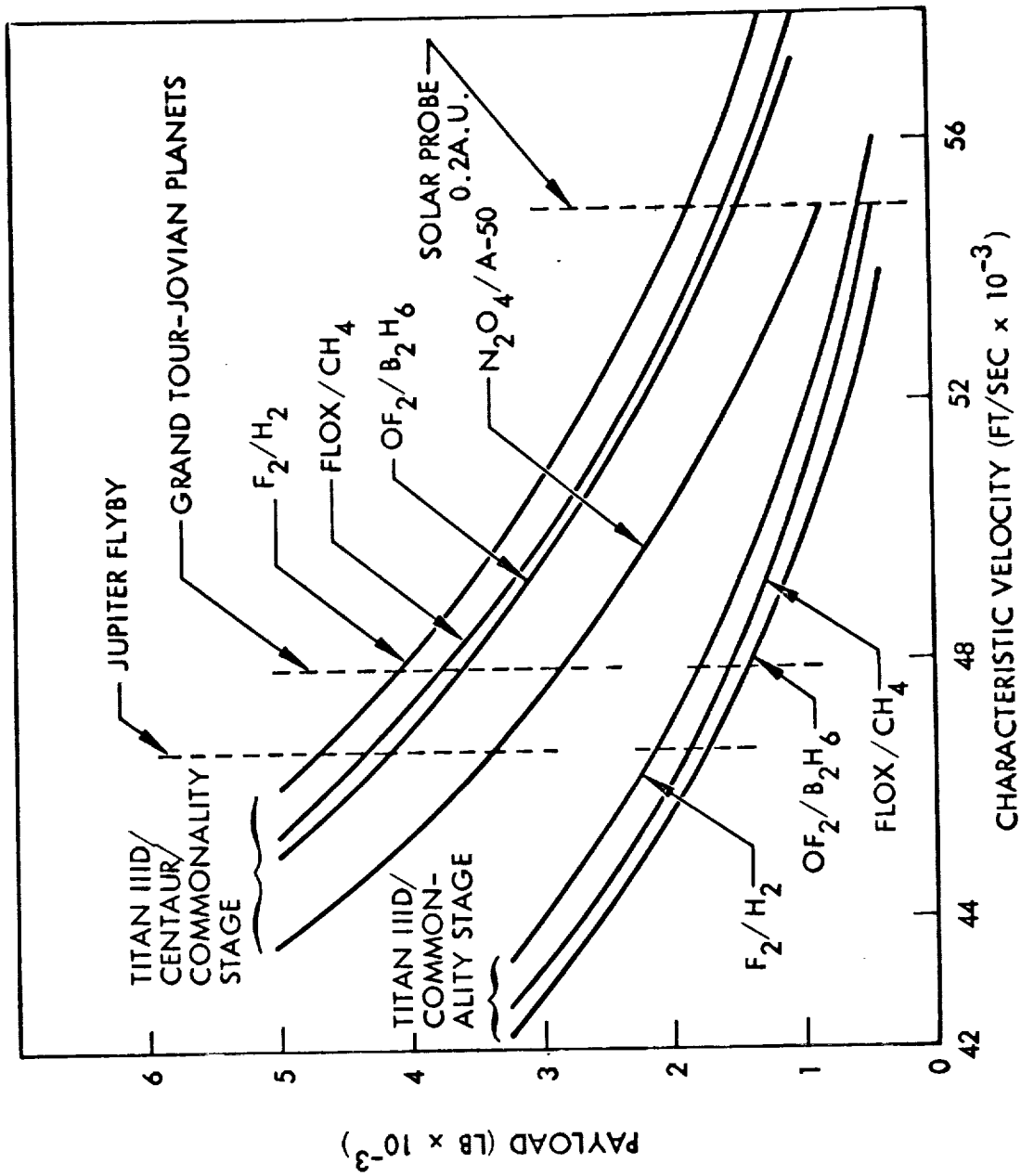


Fig. 37 Direct Ascent Payload Capability

Table 20

WEIGHTS AND DESIGN PARAMETERS FOR JUPITER ORBITER STAGE
(Weights in Pounds)

Item	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	493	415	449	518
Pressurization System	27	26	64	79
Engine System	98	98	153	151
Contingency	87	72	84	96
Residuals	128	77	90	92
Performance Reserve	24	24	24	24
Inert	1108	889	1042	1170
Impulse Propellant	3635	4056	4094	5007
Propulsion Module	4743	4945	5136	6177
Payload	1769	1661	1479	784

Design Parameters

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage (percent)
F_2	33	1/2	2
H_2	166*	1	10.4
FLOX	41	1/2	2
CH_4	32	1/2	2.9
OF_2	226	5/8	7
B_2H_6	165	1/2	2
N_2O_4	219	5	2
A-50	190	4	6.1

*Exceeds Mars Orbiter Stage design pressure and imposes a 45 lb increase in tank weight.

1.7.2.2 Mars Orbiter Stage Weights and Design Parameters for Titan IIID Without Centaur. For this mission a significant amount of the commonality stage propellant is consumed during the ascent burn at Earth. Again the F_2/H_2 system is most sensitive insofar as the commonality requirements are concerned. In this case, more than six inches of insulation would be required in order not to exceed the commonality stage design pressures. A 20 pound increase in tank weights to allow higher pressures permits a reduction in insulation thickness to 2-3/8 inches. Weight statements and design parameters are shown in Table 21 for the four propellants.

1.7.2.3 Mars Orbiter Stage Weights and Design Parameters for Stages With 6000 lb Propellant Loading. Only F_2/H_2 and FLOX/CH_4 propellants were studies for the stages with 6000 lb propellant loading. This loading delivers approximately the maximum payload to Mars on the Titan IIID/Centaur and it is also of near-optimum size for a Lunar Mission. Ascent burn is required for the Mars Orbiter mission. The resulting payloads compared to the baseline commonality stage payloads are shown in the following table for the Mars Orbiter.

Mars Orbiter Payload

Propellant	Payload, lb	
	W_p = 6000 lb	Baseline
F_2/H_2	5370	5055
FLOX/CH_4	5040	4788

The difference in payload is more significant in the Lunar Cargo Vehicle as shown below:

Lunar Cargo Payload

Propellant	Payload, lb	
	W_p = 6000 lb	Baseline
F_2/H_2	5650	3434
FLOX/CH_4	4860	3170

Weight statements and design parameters for the 6000 lb propellant load stages are shown in Table 22.

Table 21

WEIGHTS AND DESIGN PARAMETERS FOR MARS ORBITER STAGE ON TITAN IIID WITHOUT CENTAUR

(Weight in Pounds)

Item	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	443	382	406	380
Pressurization System	26	26	57	79
Engine System	98	98	153	151
Contingency	82	68	80	82
Residuals	144	91	104	94
Performance Reserve	24	26	26	27
Inert	1068	868	1004	1023
Impulse Propellant	3622	4042	4091	4874
Propulsion Module	4690	4910	5095	5897
Payload	2255	2120	1945	1328

Design Parameters

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage (percent)
F_2	44	5/8	2
H_2	127*	2-3/8	9.7
FLOX	51	5/8	2
CH_4	35	1/2	2.8
OF_2	216	1/2	8.1
B_2H_6	166	1/2	2
N_2O_4	166	1/2	2
A-50	166	1/2	2

*Exceeds Mars Orbiter Stage design pressure and imposes a 45 lb increase in tank weight.

Table 22

WEIGHTS AND DESIGN PARAMETERS FOR MARS ORBITER STAGE WITH 6,000 LB PROPELLANT LOADING

Weight Statement

Item	Weight, lb	
	F_2/H_2	$FLOX/CH_4$
Structure	320	228
Propellant Feed Assembly	673	453
Pressurization System	30	29
Engine System	98	98
Contingency	112	80
Residuals	217	112
Performance Reserve	43	42
Inert	1493	1043
Impulse Propellant	6000	6000
Propulsion Module	7493	7042
Payload	5370	5040

Design Pressures

Propellant	Tank Pressure (psia)	Insulation Thickness (in.)	Initial Ullage (percent)
F_2	44	1/2	2
H_2	98	2-7/8	2
FLOX	43	1/2	2
CH_4	35	1/2	2

1.8 COMMONALITY STAGE DESIGN SUMMARY

The propulsion stages developed in the commonality stage task have had the benefit of iterative and detailed structural, thermodynamic, performance, and operation analyses. The configuration drawings have been shown in Figs. 7 through 12, and weight statements presented in Tables 16 through 22 and in Appendix C.

The basic structure of the module consists of aluminum tubing and sections that carry the primary loads. The propellant tanks are supported by filament-wound fiberglass struts in order to reduce the heat transfer to the tanks. There is also an insulation blanket on the module bulkhead in order to control the heat transfer from the payload. The propellant tanks are made of 2021 aluminum with room temperature design allowables. Each tank has a manhole cover through which the penetrations pass. Valves, fittings, regulators, and plumbing lines were sized, and the sum of these component weights are shown in the weight statements. Each tank is individually insulated with multiple layers of double-aluminized Mylar and Tissuglas. Each tank also has a dual-wall aluminum meteoroid shield. The pressurization spheres are located in the propellant tanks except for the N_2O_4/A-50 system. These spheres could be external for all propellants and enclosed within the primary tank insulation. Titanium is used for all pressurant tanks except for the OF_2/B_2H_6 system where Inconel was used. The pressurant was stored at 4500 psi. Helium was used for the pressurant for all propellants except that heated hydrogen was used with hydrogen propellant. The engine systems selected are representative of current engine company design capability.

1.9 COMMONALITY CONCEPT ANALYSIS

The basic objective of the Commonality task was to determine the feasibility of utilizing a stage optimized for a Mars orbiter mission for other missions. This commonality stage concept proved to be very valid with study results showing that a broad range of missions could be performed with a Mars orbiter stage. Nearly all missions could be flown with minimal mission-peculiar modifications.

1.9.1 Performance Comparison

The Commonality stages optimized for the Mars Orbiter mission and Titan IIID/ Centaur have certain characteristics which are fixed and others which are variable for alternate missions. The basic assumption was that the only items variable with mission were insulation thickness, meteoroid shielding, pressurization system tanks and loading, and propellant loading. Stage structure, propellant tanks, and engine system were fixed. The following paragraphs describe how closely this commonality philosophy was met, and display the variable characteristic values for each propellant combination and mission.

1.9.1.1 Fluorine/Hydrogen Commonality Stage Performance. The F_2/H_2 Commonality Stage performed well on all missions when provided with the optimized system characteristics presented in Table 23. For two missions, the Jupiter Orbiter and the Mars Orbiter with Titan IIID (no Centaur), the Commonality stage design pressures were exceeded for the H_2 tank. This discrepancy can be overcome in one of several ways:

1. H_2 tank weight could be increased 45 lb for the Jupiter Orbiter (20 lb required for the Mars Orbiter w/Titan IIID). This weight penalty would then be imposed on all missions using the Commonality stage.
2. Greater separation (one foot) could be specified and a shadow shield added between propulsion stage and payload for the critical missions only. This would reduce tank pressures and impose no penalties on the Commonality stage.
3. Detailed redesign to reduce structural heat leaks might eliminate the problem at little or no penalty.
4. A hydrogen vent system could be added.

1.9.1.2 FLOX/Methane Commonality Stage Performance. The FLOX/CH_4 Commonality Stage performed well on all missions with no violations of the commonality concept. Payloads are slightly lower than for for F_2/H_2. Optimized characteristics for each mission are presented in Table 24.

Table 23

F_2/H_2 COMMONALITY STAGE CHARACTERISTICS

Mission	Payload (lb)	Impulse Propellant (lb)	Meteoroid Shield (lb)	Pressurization System (lb)	Tank Operating Pressure (psi)		Insulation Thickness (in.)		Initial Ullage (%)	
					F_2	H_2	F_2	H_2	F_2	H_2
Mars Orbiter	5055	3622	77	27	44	91	1/2	2-1/4	2	15
Venus Orbiter	4779	3620	72	27	213	94	1/2	3	2.4	22.4
Lunar Cargo	3434	3632	33	26	33	25	1/2	1	2	23.3
Jupiter Orbiter	1769	3635	130	27	33	166*	1/2	1	2	10.4
Mars Orbiter w/Titan IIID	2255	3622	77	26	44	127**	5/8	2-3/8	2	9.7
Mars Orbiter – W_p = 6000 lb	5370	6000	131	30	44	98	1/2	2-7/8	2	2

*Exceeds design pressure of commonality stage and imposes a 45 lb increase in tank weight.
**Exceeds design pressure of commonality stage and imposes a 20 lb increase in tank weight.

Table 24

FLOX/CH_4 COMMONALITY STAGE CHARACTERISTICS

Mission	Payload (lb)	Impulse Propellant (lb)	Meteoroid Shield (lb)	Pressurization System (lb)	Tank Operating Pressure (psi)		Insulation Thickness (in.)		Initial Ullage (%)	
					FLOX	CH_4	FLOX	CH_4	FLOX	CH_4
Mars Orbiter	4788	4044	52	27	65	31	1/2	1/2	2	2
Venus Orbiter	4505	4037	50	29	210	48	1/2	1/2	2.6	2
Lunar Cargo	3170	4058	25	25	33	33	1/2	1/2	2	2.8
Jupiter Oribter	1661	4056	87	26	41	32	1/2	1/2	2	2.9
Mars Orbiter – W/Titan IIID	2120	4042	52	26	51	35	5/8	1/2	2	2.8
Mars Orbiter – W_p = 6000 lb	5040	6000	75	29	43	35	1/2	1/2	2	2

1.9.1.3 Oxygen difluoride/Diborane Commonality Stage Performance. The OF_2/B_2H_6 Commonality Stage performed well on all missions with no violations of the Commonality concept. Payloads are lower than for either F_2/H_2 or $FLOX/CH_4$. Optimized characteristics for each mission are presented in Table 25.

1.9.1.4 Nitrogen Tetroxide/Aerozine-50 Commonality Stage Performance. The N_2O_4/A-50 Commonality Stage gave substantially lower payloads than the alternate propellants, and could not perform the Jupiter Orbiter mission without incorporating excessive amounts of insulation (5 to 6 inches) or actively heating the propellants to prevent them from freezing. Optimized characteristics for each mission are presented in Table 26.

1.9.2 System Sensitivity

The greatest sensitivity to mission variations was shown by F_2/H_2 and N_2O_4/A-50. The space storables, $FLOX/CH_4$ and OF_2/B_2H_6, showed the greatest flexibility in performing the various missions.

1.9.2.1 Fluorine/Hydrogen Stage Sensitivity. In the F_2/H_2 systems the fluorine tank operating pressure was very low for all missions except the Venus orbiter for which the total tank pressure capability was utilized. For hydrogen there was more variation, but only for the Jupiter orbiter and the Mars orbiter w/Titan IIID was the tank design pressure exceeded. The variation in hydrogen ullage is primarily an indicator of expansion requirements during the mission. This is somewhat attenuated for the ascent burn cases, but these have less propellant to serve as a heat sink for the remainder of the mission. The cumulative heating, primarily by conduction, during the long duration Jupiter orbiter mission provided the severest test to the commonality stage. In allowing the optimization to search freely a tank pressure of 166 psi was obtained. For the Mars orbiter w/Titan IIID mission, the optimization was run with and without the tank pressure constraints. The results are shown in Table 27.

Table 25

OF_2/B_2H_6 COMMONALITY STAGE CHARACTERISTICS

Mission	Payload (lb)	Impulse Propellant (lb)	Meteoroid Shield (lb)	Pressurization System (lb)	Tank Operating Pressure (psi)		Insulation Thickness (in.)		Initial Ullage (%)	
					OF_2	B_2H_6	OF_2	B_2H_6	OF_2	B_2H_6
Mars Orbiter	4608	4085	63	59	241	155	1/2	1/2	8	2
Venus Orbiter	4352	4088	59	58	228	155	1/2	1/2	8.5	2
Lunar Cargo	2995	4083	28	58	166	166	1/2	1/2	8.7	2
Jupiter Orbiter	1479	4094	104	64	226	165	5/8	1/2	7	2
Mars Orbiter – W/Titan IIID	1945	4091	63	57	216	66	1/2	1/2	8.1	2

Table 26

N_2O_4/A-50 COMMONALITY STAGE CHARACTERISTICS

Mission	Payload (lb)	Impulse Propellant (lb)	Meteoroid Shield (lb)	Pressurization System (lb)	Tank Operating Pressure (psi)		Insulation Thickness (in.)		Initial Ullage (%)	
					N_2O_4	A-50	N_2O_4	A-50	N_2O_4	A-50
Mars Orbiter	3797	4868	57	80	100	155	1/2	5/8	2	2
Venus Orbiter	3555	4869	53	68	174	155	1/2	1/2	2	2
Lunar Cargo	2317	4868	26	69	158	167	1/2	1/2	2	2
Jupiter Orbiter	784	5007*	94	79	219	190	5	4	2	6.1
Mars Orbiter – W/Titan IIID	1328	4874	57	79	166	166	1/2	1/2	2	2

*Propellant loaded at 500 °R.

Table 27

EFFECTS OF PRESSURE CONSTRAINTS ON MARS ORBITER W/TITAN IIID

Item	With Pressure Constraint	Without Pressure Constraint
Tank Pressure, psi	96	127
Insulation Thickness, inches	7-3/8	2-3/8
Tank Weight, lb	109	129
Insulation Weight, lb	136	44
Residual Vapor, lb	43	60
Pressure Sensitive Weight Total, lb	288	233

This shows that, by violating the tank pressure constraint, approximately 55 more pounds of payload on the Mars orbiter w/Titan IIID could be delivered, but a 20-pound tank penalty would be imposed on all the other commonality stage missions which do not test the tank pressure constraint. Alternate design approaches, discussed earlier under Section 1.9.1.1, might prove more attractive.

1.9.2.2 FLOX/Methane Stage Sensitivity. The FLOX/CH_4 design provides a very good commonality stage that is relatively insensitive to mission variations.

1.9.2.3 Oxygen Difluoride/Diborane Stage Sensitivity. The OF_2/B_2H_6 design also provides a very good commonality stage that is relatively insensitive to the mission variations.

1.9.2.4 Nitrogen Titroxide/Aerozine-50 Stage Sensitivity. Although one might expect N_2O_4/A-50 to provide an insensitive commonality stage, propellant freezing was a problem for the Jupiter mission. This difficulty was overcome by loading the propellant subcooled at 500 °F, increasing the α/ϵ to 2.2, and letting the optimization program solve for insulation thickness which becomes rather large. Active propellant heating would probably prove a more attractive solution than adding insulation. For the Mars orbiter w/Titan IIID mission, increasing the α/ϵ to 0.71 from 0.66 prevented the propellant from freezing.

1.9.3 Commonality Concept Conclusions

Both FLOX/CH_4 and OF_2/B_2H_6 provide great flexibility for a commonality stage which can be flown on all selected missions without modification to the baseline Mars orbiter stage.

F_2/H_2 systems continue to provide the best performance for all applications, but both the Mars Orbiter w/Titan IIID mission and the Jupiter mission have tank pressure requirements which exceed those for the baseline Mars orbiter and impose either a 20 to 45 lb tank weight penalty on the Mars orbiter stage concept or an alternate design approach.

The N_2O_4/A-50 propellant system has a broad range of applicability except for the Jupiter mission. For this mission the payload is too low to be of great interest and special protection from propellant freezing must be considered.

2.0 COMMONALTY STAGE APPLICATIONS SUMMARY

The comprehensive analyses that have been conducted on high energy propulsion stages for space applications during the past two years has paved the way for a low cost approach to a broad range of missions. Table 28 summarizes the missions and payloads which can be supported by the propulsion stages described in the report, when launched by Titan IIID/Centaur or Titan IIID. Additional missions, not analyzed, will also be attractive with this booster and stage combination. These include flybys of Saturn, of Mercury by way of Venus, of comets and of asteroids.

Table 28

COMMONALITY STAGE PAYLOAD CAPABILITIES SUMMARY

Mission		Payloads (Pounds)			
		F_2/H_2	$FLOX/CH_4$	OF_2/B_2H_6	N_2O_4/A-50
Mars Orbiter	-1973	5055	4788	4608	3802
Venus Orbiter	-1976	4779	4505	4352	3555
Lunar Cargo	-1975	3434	3170	2995	2317
Jupiter Orbiter	-1980	1769	1661	1479	784
Mars Orbiter W/Titan IIID	-1973	2255	2120	1945	1328
Mars Orbiter – W_p = 6000 lb	-1973	5370	5040	–	–
Solar Probe	-1973	1830	1590	1460	890
Jupiter Flyby	-1975	4700	4360	4140	3420
Grand Tour of Jovian Planets 1976, 1977, 1978, 1979		4090	3770	3570	2880

Section 3
ATTITUDE CONTROL SYSTEM REQUIREMENTS DEFINITION

3.1 TASK OBJECTIVES

The objectives of Task VI, Attitude Control System Requirements Definition, were to:

a. Define, for spacecraft and missions matched to the capability of Titan III D/Centaur, the following attitude control system requirements and effects:

   Thruster location
   Thrust level or levels
   Duty cycle
   Minimum impulse bit
   Total impulse
   Modes of operation

b. Compare, in a preliminary manner, methods for mechanizing attitude control systems to meet requirements. Consider cold gas, monopropellant, and bipropellant (separate and integrated with secondary or primary propulsion system) and estimate system weights. Recommend a concept that is attractive for analysis in greater depth.

3.2 GROUND RULES AND CONSTRAINTS

The task was performed under the following assumptions and constraints:

- The spacecraft is sized to the launch capability of Titan III D/Centaur
- Main Propulsion System alternatives are:
  - 5000 lb thrust, F_2/H_2, Pump Fed
  - 5000 lb thrust, FLOX/CH_4, Pump Fed
  - 3500 lb thrust, OF_2/B_2H_6, Pressure Fed
  - 3500 lb thrust, N_2O_4/A-50, Pressure Fed

- Midcourse and orbit trim corrections are accomplished by the throttled main engine
- The main engine is gimbaled for thrust vector control
- Impulse accuracy of the throttled main engine (throttled to 500 lbf) is:

  3σ impulse repeatability $\simeq$ 40 lb-sec

  minimum impulse bit $\simeq$ 400 lb-sec
- Desired $\Delta V_{min} = 1 \pm 0.1$ meter/sec
- The vehicle is stabilized full time about three axes, with one axis pointed toward the sun

3.3 VEHICLE CONFIGURATIONS AND INERTIAS

Missions and spacecraft compatible with the launch capabilities of Titan III D/Centaur were assumed. These included:

Mars Orbiter – Orbit insertion stage
Jupiter Orbiter – Earth escape and orbit insertion stage
Venus Orbiter – Orbit insertion stage
Jupiter Flyby – Earth escape stage
Solor Probe – Earth escape stage
Lunar Cargo Delivery – Orbiter/Lander stage

Moments of inertia were computed about three axes for each of these configurations as a function of mission time. Assumptions made to simplify the calculations were:

1. Propellant is a spherical mass located at the center of the tank during coast
2. Configurations can be defined by blocks of time
3. For Jupiter missions, radioisotope thermal generators are flush mounted in a symmetrical pattern in the hex sides of the spacecraft bus
4. Inertias for the Venus Orbiter could be assumed similar to the Mars Orbiter for analysis purposes

Details of spacecraft configurations and changes as propellant is consumed and payloads separated are presented in the following figure and tables.

Figure 38 illustrates a typical spacecraft configuration at the start of interplanetary cruise and defines the reference axes used for describing mass locations. The spacecraft consists of a propulsion stage, spacecraft bus or equipment module (shown with solar panels deployed), and a lander capsule. The configuration, masses, and inertias change with mission time as propellant is consumed and the lander capsule is separated.

Descriptions of the mass distribution for each configuration are shown in Tables 29 through 31 except for the lunar lander configuration which was lifted from Lockheed's Intermediate Size Lunar Landing Spacecraft proposal, LMSC-A946971. Calculated moments of inertia are summarized in Table 32.

3.4 GUIDANCE ERROR EFFECTS

The relationships between guidance pointing errors allowable, attitude reorientation time, and attitude hold angle limits should be known when determining requirements for an attitude control system. A guidance error analysis was therefore performed for Mars, Venus, Jupiter, and Lunar missions utilizing the Titan III/Centaur booster. Two midcourse corrections were simulated for each mission, with the first occurring in the earth departure phase and the second during the target body approach phase. The sensitivities of the midcourse velocity and residual miss at the target to midcourse pointing and impulse errors were determined based on a particular booster injection covariance matrix. Effects of tracking errors were not considered.

The guidance law followed for both maneuvers was to correct deviations in miss vector components, $\overline{B}\cdot\overline{T}$ and $\overline{B}\cdot\overline{R}$, and linearized time of arrival, t_A. The variables of interest for each mission were the rms values of the magnitudes of the velocity corrections and the residual miss due to the midcourse errors. The analysis utilized a matched-conics trajectory computer program to generate nominal mission trajectories and to build matrices of partial deviatives for mapping parameter variations along the trajectory.

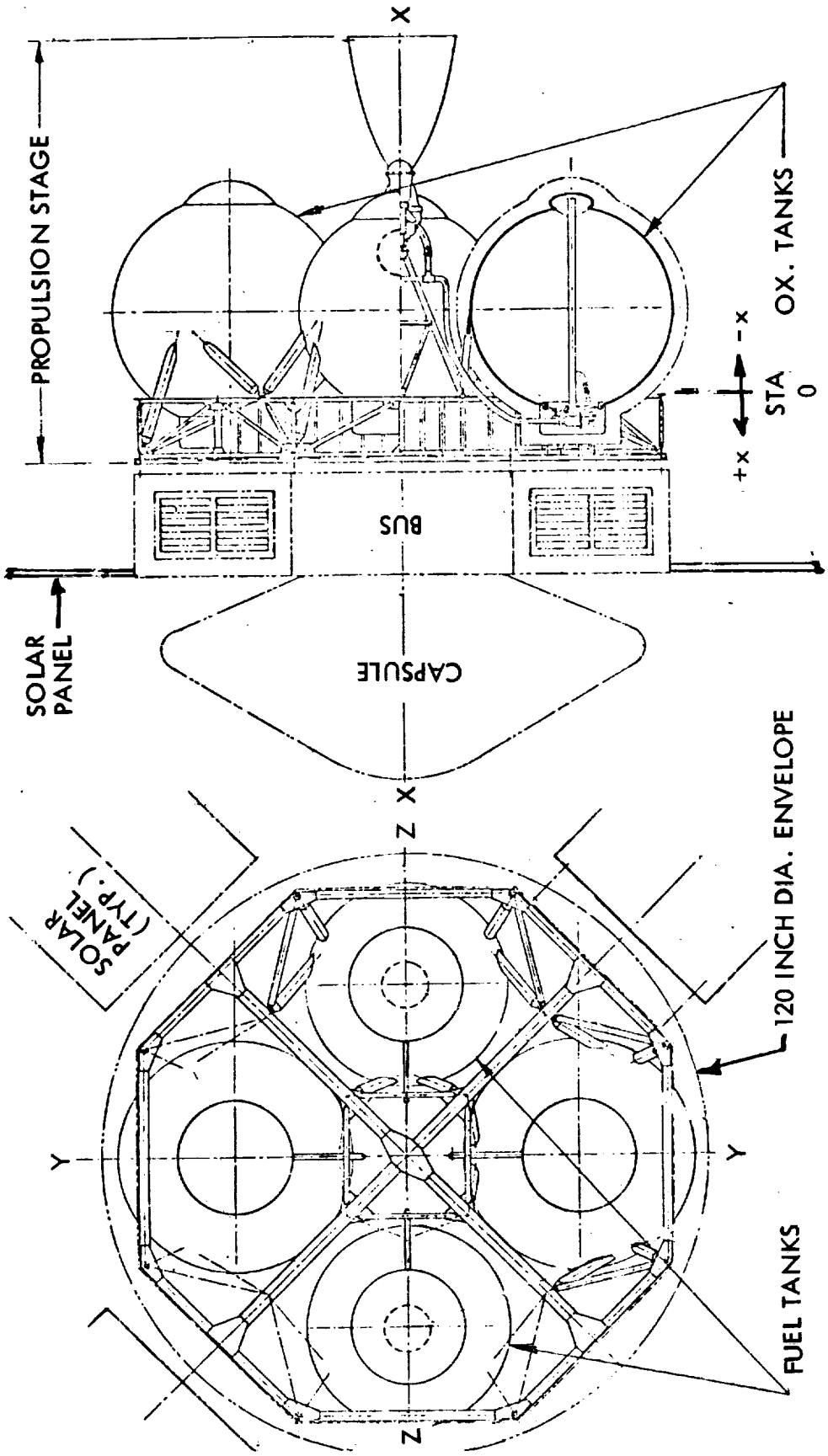


Fig. 38 Typical Spacecraft at Start of Cruise

Table 29

MARS (OR VENUS) CONFIGURATIONS

Item No. and Name	Mass (lb)	X (in.)	Radius From X-X (in.)	Assumed Mass Distribution
CONFIGURATION MARS 1				
1. Engine	98	-50	0	Point mass
2. Engine Support	10	-10	0	Point mass
3. Stage Frame	43	6	51 to 53	12 in. thick ring perpendicular to XX axis
4. Beams	34	6	0 to 52	Cruciform with two 2 × 12 × 104 in. beams ⊥ to XX
5. Insulation	18	13	0 to 52	Disc of uniform mass ⊥ to XX axis
6. Bus	3981	27	15 to 52	24 in. thick disc of uniform mass ⊥ to XX
7. Solar Panels	100	39	68 to 155	Four 25 lb panels, 42 × 87 in., from r = 68 to r = 155
8. Fuel System	270	-17	34	Two 135 lb point masses in opposite quadrants
9. Oxid. Syst.	346	-17	34	Two 173 lb point masses in opposite quadrants
10. Fuel (usable)	640	-17	34	In two spherical masses of 34 in. dia., centered on ZZ
11. Oxid. (usable)	3360	-17	34	In two spherical masses of 40 in. dia., centered on YY
12. Capsule	800	64	0 to 48	36 in. thick disc of uniform mass ⊥ to XX
CONFIGURATION MARS 2				
1. through 9.	4900	(Same as Mars 1)		
10. Fuel (usable)	30	-17	34	In two point masses of 15 lb ea.
11. Oxid. (usable)	158	-17	34	In two point masses of 79 lb ea.
12. Capsule	(Same as Mars 1)			
CONFIGURATION MARS 3				
1. through 9.	4900	(Same as Mars 1)		
10. Fuel (usable)	0	–	–	
11. Oxid. (usable	0	–	–	
12. Capsule	0	–	–	

Table 30

JUPITER ORBITER CONFIGURATIONS

Item No. and Name	Mass (lb)	X (in.)	Radius From X-X (in.)	Assumed Mass Distribution
CONFIGURATION JUPITER 1				
1. through 5.	203	(Same as Mars 1)		
6. Bus	1600	27	15 to 52	Disc of uniform mass ⊥ to XX
7. Fuel System	270	-17	34	Two 135 lb point masses in opposite quadrants
8. Oxid. Syst.	346	-17	34	Two 173 lb point masses in opposite quadrants
9. Fuel (usable)	640	-17	34	In two spherical masses of 34 in. diameter
10. Oxid. (usable)	3360	-17	34	In two spherical masses of 40 in. diameter
CONFIGURATION JUPITER 2				
1. through 8.	2419	(Same as Jupiter 1)		
9. Fuel (usable)	290	-26	34	In two point masses of 145 lb ea.
10. Oxid. (usable)	1490	-26	34	In two point masses of 745 lb ea.
CONFIGURATION JUPITER 3				
1. through 8.	2419	(Same as Jupiter 1)		
9. Fuel (usable)	0	–	–	
10. Oxid. (usable)		–	–	

Table 31

JUPITER FLYBY AND SOLAR PROBE CONFIGURATIONS

Item No. and Name	Mass (lb)	X (in.)	Radius From X-X (in.)	Assumed Mass Distribution
CONFIGURATION JUPITER FB 1				
1. through 5.	203	(Same as Mars 1)		24 in. thick disc of uniform mass ⊥ to XX
6. Bus	4330	27	0 to 52	
7. through 10.	4616	(Same as Jupiter 1)		
CONFIGURATION JUPITER FB 2				
1. through 5.	(discarded)			24 in. thick disc of uniform mass ⊥ to XX
6. Bus	4330	–	0 to 52	
7. through 10.	(discarded or consumed)			
CONFIGURATION SOLAR PROBE 1				
1. through 5.	203	(Same as Mars 1)		24 in. thick disc of uniform mass ⊥ to XX
6. Bus	1530	27	0 to 52	
7. through 10.	4616	(Same as Jupiter 1)		
11. Solar Panels	50	55 to 97	52	Four 12.5 lb panels, 42 × 42 in. extending fwd in XX direction
CONFIGURATION SOLAR PROBE 2				
1. through 5.	(discarded)			24 in. thick disc of uniform mass ⊥ to XX
6. Bus	1530	27	0 to 52	
7. through 10.	(discarded or consumed)			
11. Solar Panels	50	39	68 to 110	Four 12.5 lb panels, 42 × 42 in. extending ⊥ to XX

Table 32

SUMMARY OF MOMENTS OF INERTIA

Item	Configuration					
	Mars 1	Mars 2	Mars 3	Jupiter 1	Jupiter 2	Jupiter 3
Weight (lb)	9700	5880	4900	6419	4199	2419
I_{xx} (slug-ft^2)	3079	1999	1753	1823	1138	696
I_{yy} (slug-ft^2)	2885	1739	1250	1188	1015	628
I_{zz} (slug-ft^2)	3580	1789	1268	1883	1333	646

Item	Configuration					
	Jup. FB 1	Jup. FB 2	Solar 1	Solar 2	Lunar 1	Lunar 2
Weight (lb)	9149	4330	6399	1580	12307	9051
I_{xx} (slug-ft^2)	2685	1264	1852	535	3890	2624
I_{yy} (slug-ft^2)	2102	677	1207	285	3980	3109
I_{zz} (slug-ft^2)	2798	677	1902	285	4119	3248

The characteristics of the various missions are summarized below:

MISSION	LAUNCH DATE (day/month/year)	TRIP TIME	NOMINAL MIDCOURSE TIME 1st	2nd
Mars	8/10/73	195 days	7 days	180 days
Venus	11/23/76	175 days	7 days	160 days
Jupiter	6/02/74	900 days	7 days	850 days
Moon	9/12/73	84 hr	17 hr	74 hr

METHOD

The error models for the midcourse maneuvers were as follows: The velocity errors due to an error in pointing and in impulse, $\delta\overline{V}_p$ and $\delta\overline{V}_I$, respectively, were presented by

$$\delta\overline{V}_P = \bar{e}_p \times \overline{V}$$

$$\delta\overline{V}_I = e_I \left(\frac{g}{W}\right)\left(\frac{\overline{V}}{V_{rms}}\right)$$

where $\overline{V}$ is the velocity correction vector, V_{rms} is its rms magnitude, which is given by

$$V_{rms} = \left\{E\left[V_1^2 + V_2^2 + V_3^2\right]\right\}^{1/2} = \left\{Tr\left[cov\,(\overline{V})\right]\right\}^{1/2}$$

W is the spacecraft weight, g the earth gravitational acceleration, and pointing error sources e_{p_i} (i = 1, 2, 3) and impulse error e_I follow independent Gaussian distributions with zero means and variances, $var\,(e_{p_i}) = \sigma_p^2$ and $var\,(e_I) = \sigma_I^2$. The covariance matrix of midcourse errors is then found as

$$cov\,(\delta\overline{V}) = cov\,(\delta\overline{V}_P) + cov\,(\delta\overline{V}_I)$$

where

$$\mathrm{cov}(\delta\overline{V}_P) = \sigma_P^2 \left[V_{rms}^2 \cdot I - \mathrm{cov}(\overline{V}) \right]$$

$$\mathrm{cov}(\delta\overline{V}_I) = \sigma_I^2 \left(\frac{g}{W}\right)^2 \left(\frac{1}{V_{rms}^2}\right) \mathrm{cov}(\overline{V})$$

and I is a 3×3 identity matrix. The midcourse covariance matrix, $\mathrm{cov}\ (\overline{V})$, is obtained by a linear mapping of either the injection covariance matrix, or the previous midcourse error covariance matrix, to the midcourse point, with the mapping function defined by the guidance law simulated.

After each midcourse, a residual miss covariance matrix, $\mathrm{cov}\ (\delta\overline{m}, \delta t_A)$, where $\delta\overline{m} = \delta(\overline{B} \cdot \overline{T}, \overline{B} \cdot \overline{R})$ and δt_A is arrival time error, is found by mapping the midcourse error covariance matrix into the terminal miss coordinate system. The rms miss magnitude error corresponding to the midcourse errors is then given by

$$\delta m_{rms} = \left\{ T_r\ \mathrm{cov}(\delta\overline{m}) \right\}^{1/2}$$

Based on the above model, it can be shown that the rms values of midcourse velocity and miss for each maneuver can be written as functions of the system errors in the following form:

$$V_{1_{rms}} = a_1$$

$$\sigma m_{1_{rms}} = \left[a_2 \sigma_P^2 + a_3 \sigma_I^2 \right]^{1/2}$$

$$V_{2_{rms}} = \left[b_1 \sigma_P^{\ 2} + b_2 \sigma_I^{\ 2}\right]^{1/2}$$

$$\sigma m_{2_{rms}} = \left[b_3 \sigma_P^{\ 4} + b_4 \sigma_P^{\ 2} \sigma_I^{\ 2} + b_5 \sigma_I^{\ 2}\right]^{1/2}$$

where, for a given injection covariance matrix, the sensitivity coefficients $a_i = a_i(t_1)$ and $b_i = b_i(t_1, t_2)$ are merely functions of the mission trajectory and the midcourse execution times, t_1 and t_2.

For each of the missions, the sensitivity coefficients were determined for various combinations of midcourse times by making runs with the matched-conics trajectory program. Values of rms midcourse velocity and miss for each midcourse were then calculated for a range of pointing errors, σ_P, from 0 to 2 degrees, and normalized impulse errors, σ_I, from 0 to 80 lb-sec/W, where W, the spacecraft weight, was taken as nominally 10,000 lbs for all missions except the one to Jupiter. For the Jupiter Mission, W = 5000 lbs. Note that a σ_I of 41 lb-sec/10,000 lbs corresponds to a velocity error of about .04 m/s.

RESULTS

The results of this study are presented in figs. 39 through 49. These results can be considered at the 3σ probability level. In fig. 39 is presented the rms velocity magnitude for the first midcourse as a function of execution time t_1 and type of mission. The Lunar mission is seen to require the largest velocity correction, and the Jupiter mission the smallest.

The rms velocity for the second correction is shown in figs. 40 to 43 for each mission as functions of the first midcourse pointing and impulse errors, and the execution time t_2 for the second maneuver. It is seen that the second midcourse

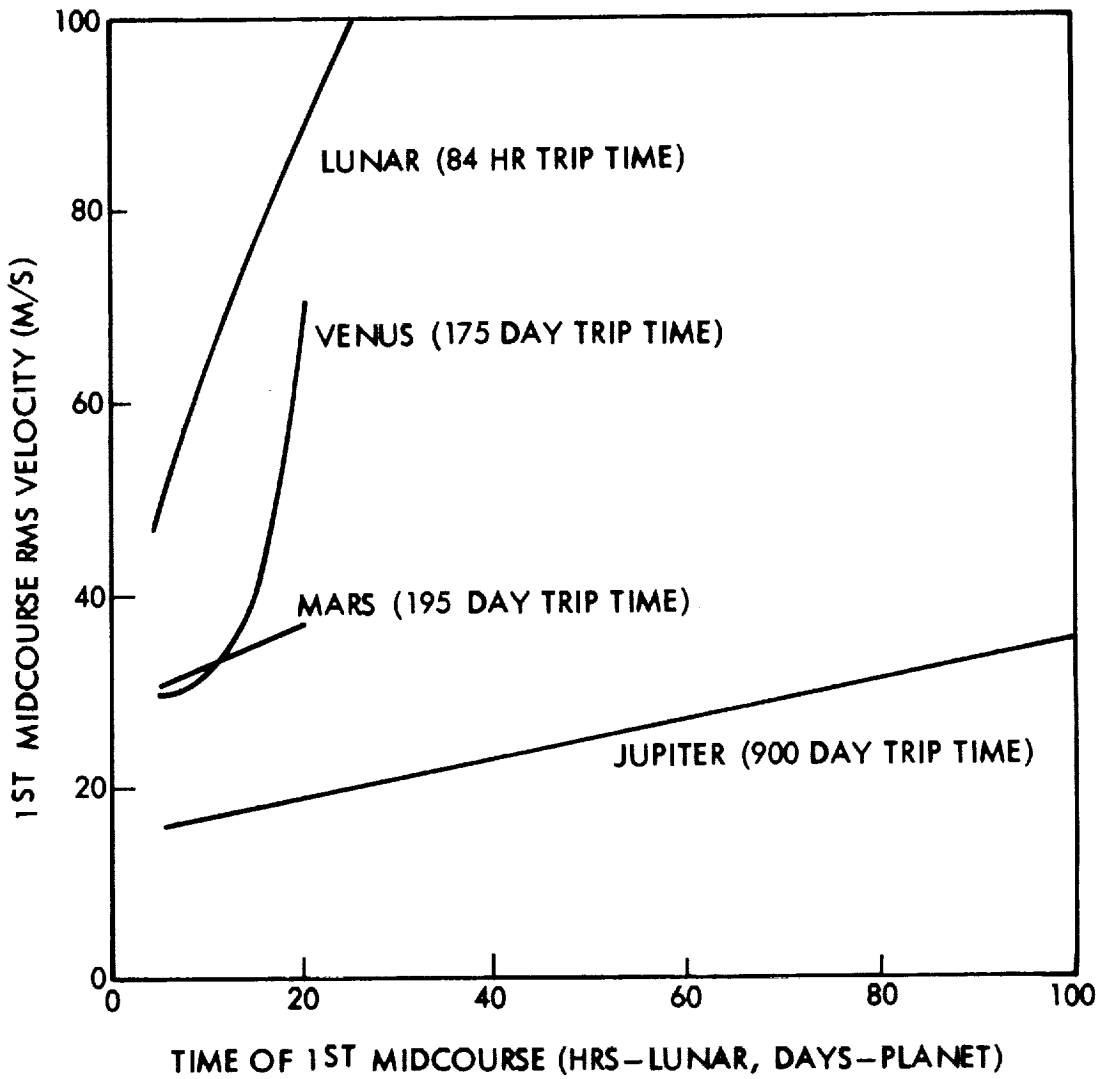


Fig. 39 Sensitivity Of First Midcourse Δ Velocity To Mission And Execution Time

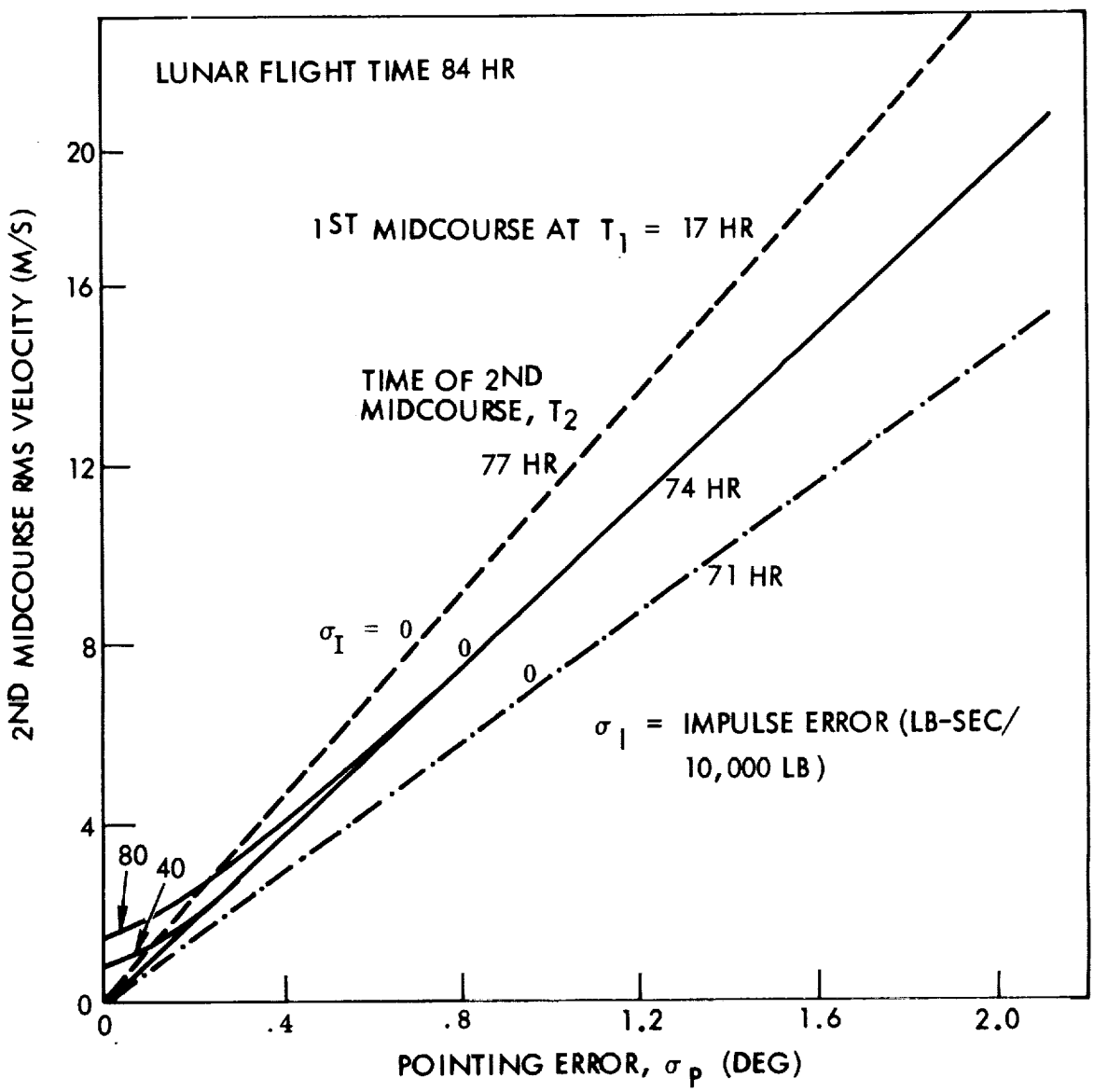


Fig. 40 Effects of Execution Time and First Midcourse Errors On Second Midcourse Δ Velocity For Lunar Mission

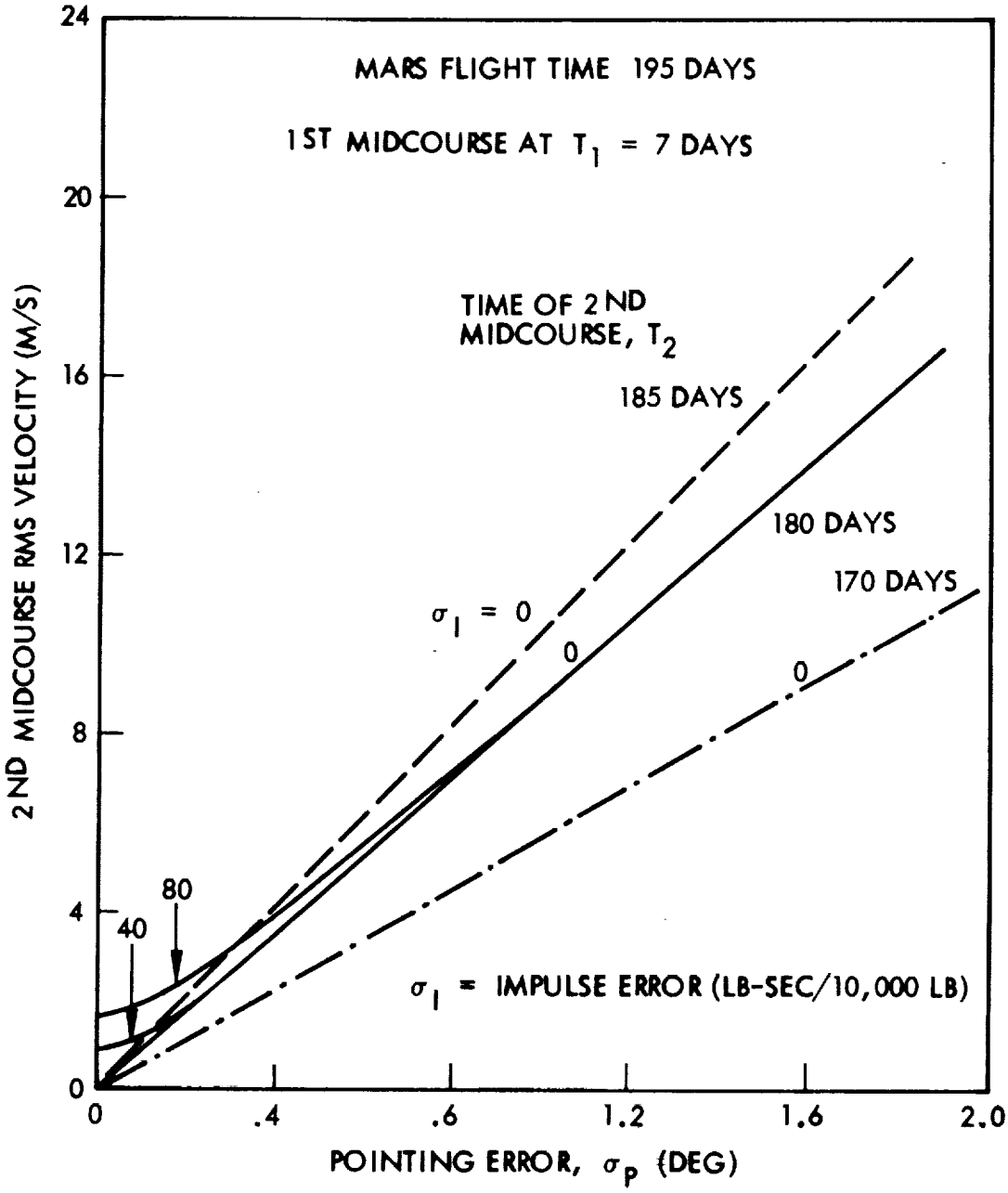


Fig. 41 Effects of Execution Time and First Midcourse Errors On Second Midcourse Δ Velocity For Mars Mission

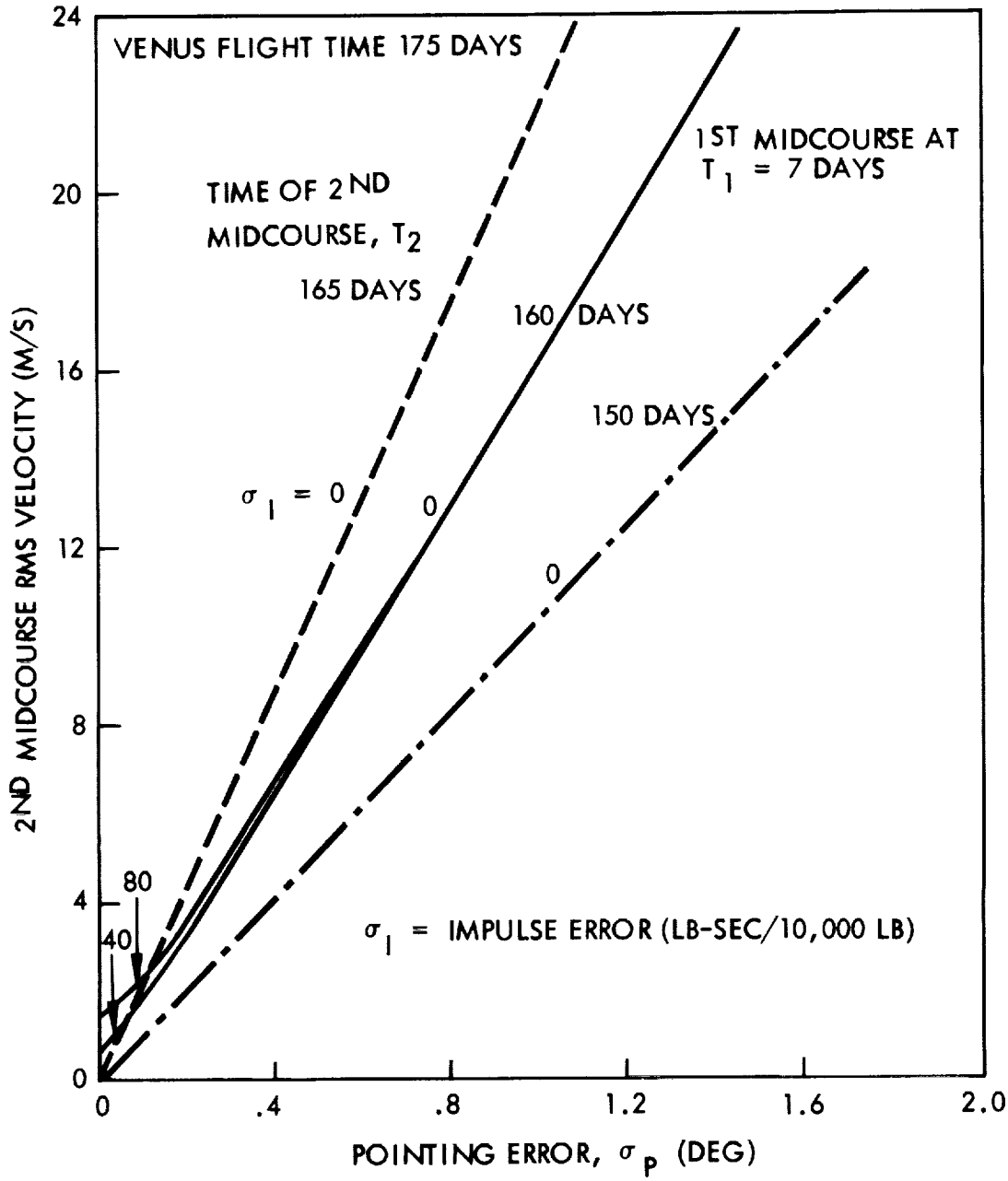


Fig. 42 Effects Of Execution Time and First Midcourse Errors On Second Midcourse Δ Velocity for Venus Mission

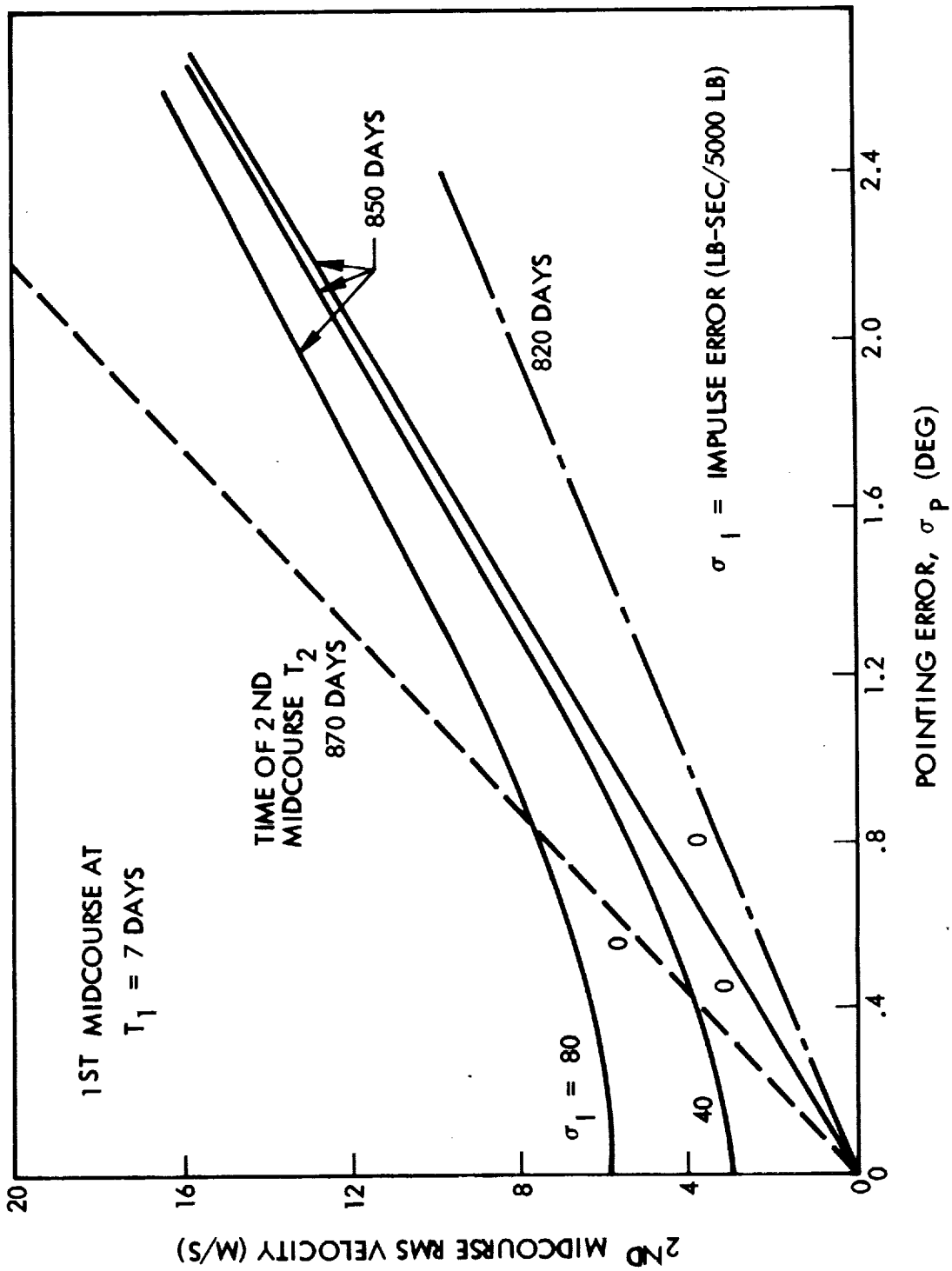


Fig. 43 Effects Of Execution Time and First Midcourse Errors On Second Midcourse Velocity for Jupiter Mission

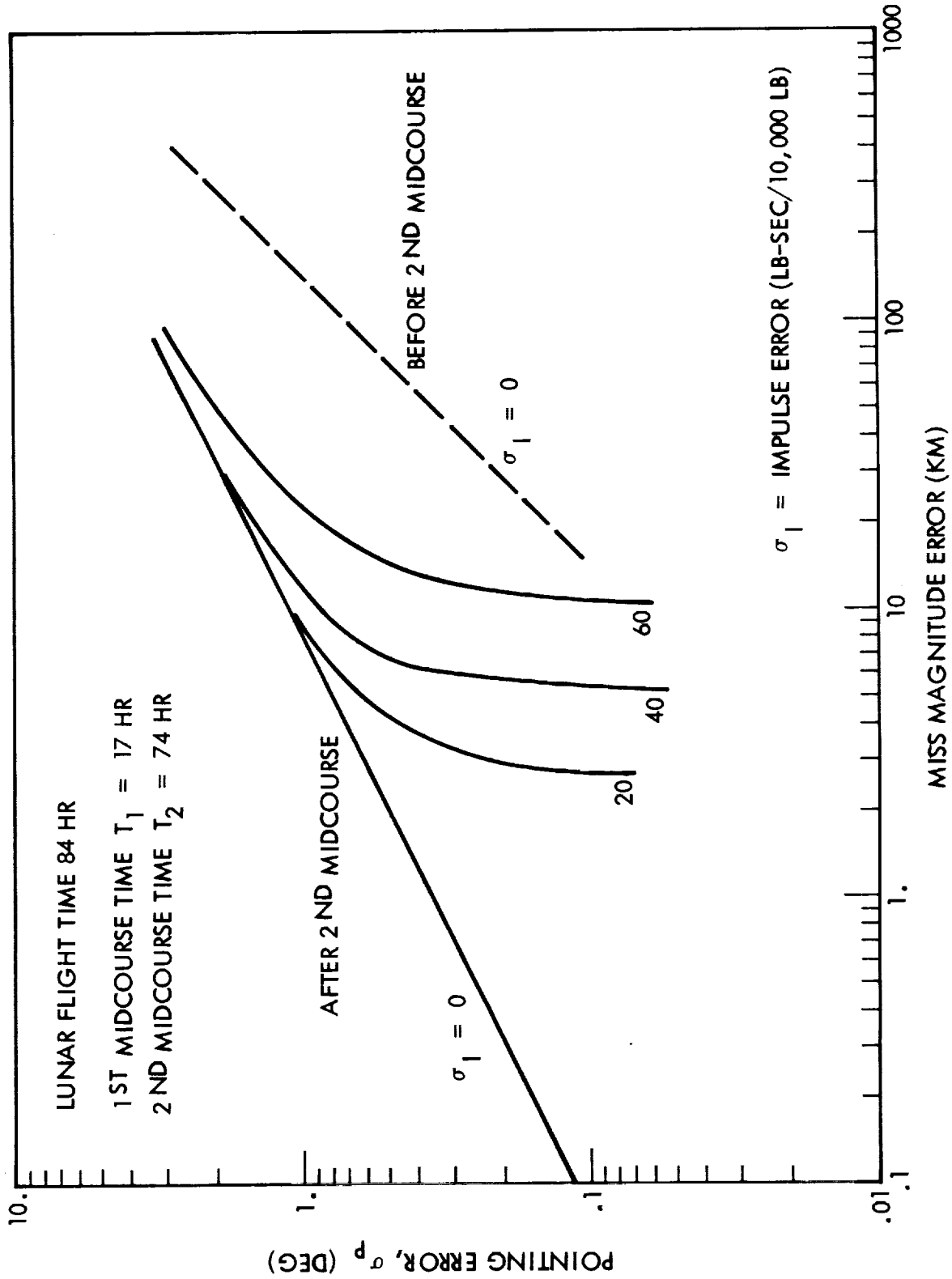


Fig. 44 Sensitivity Of Residual Miss To Midcourse Errors For Lunar Mission

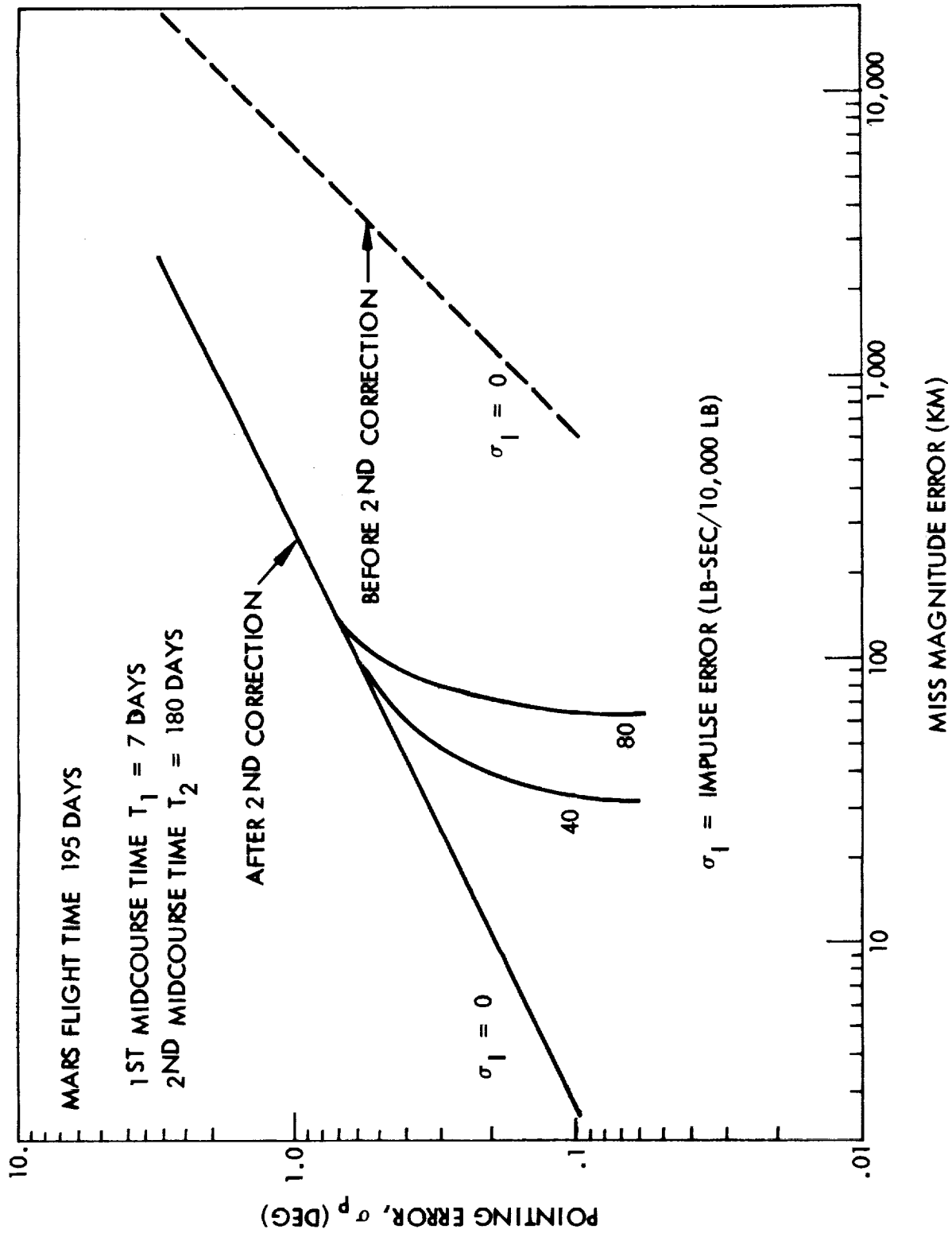


Fig. 45 Sensitivity of Residual Miss To Midcourse Errors For Mars Mission

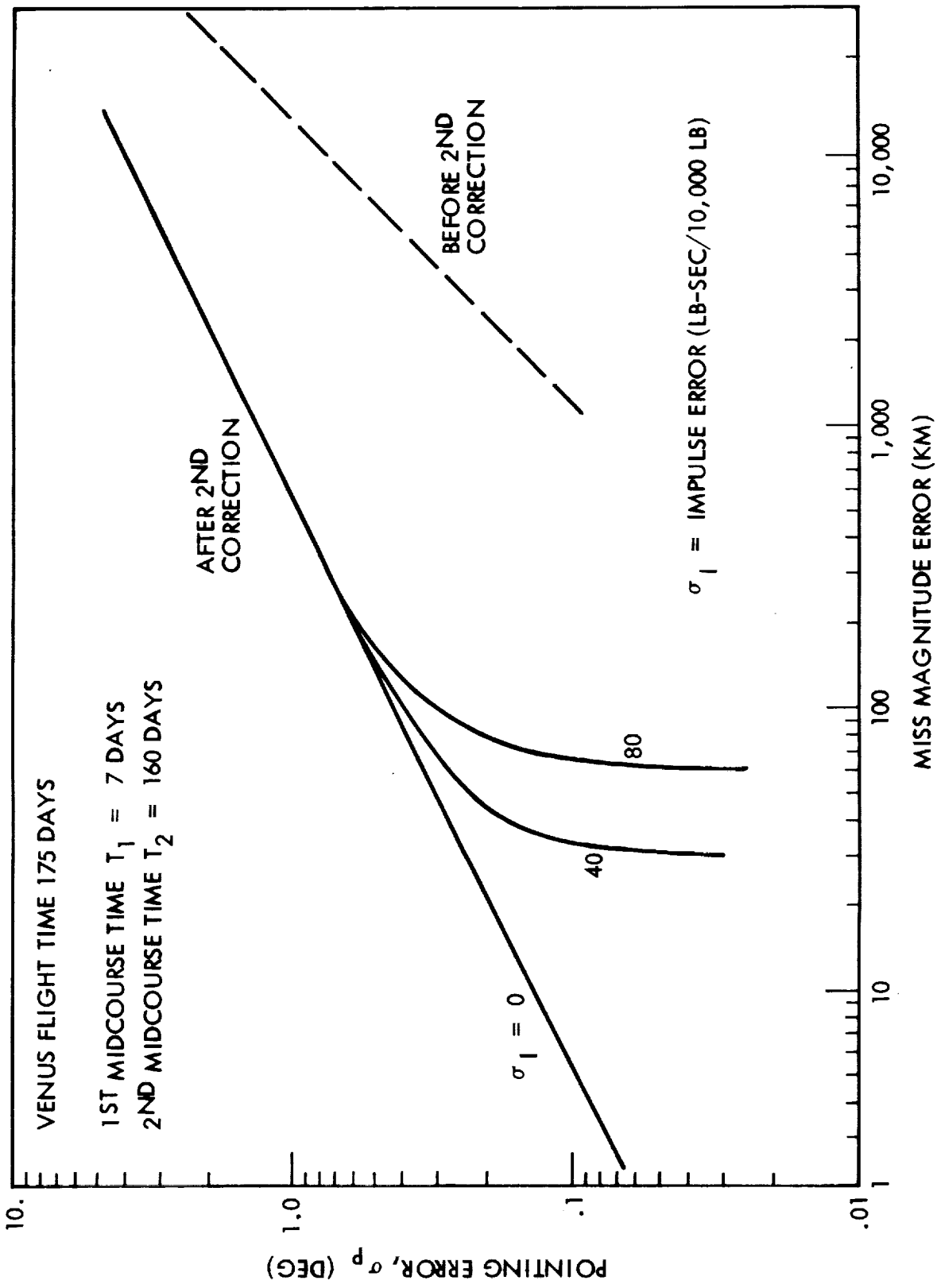


Fig. 46 Sensitivity of Residual Miss To Midcourse Errors For Venus Mission

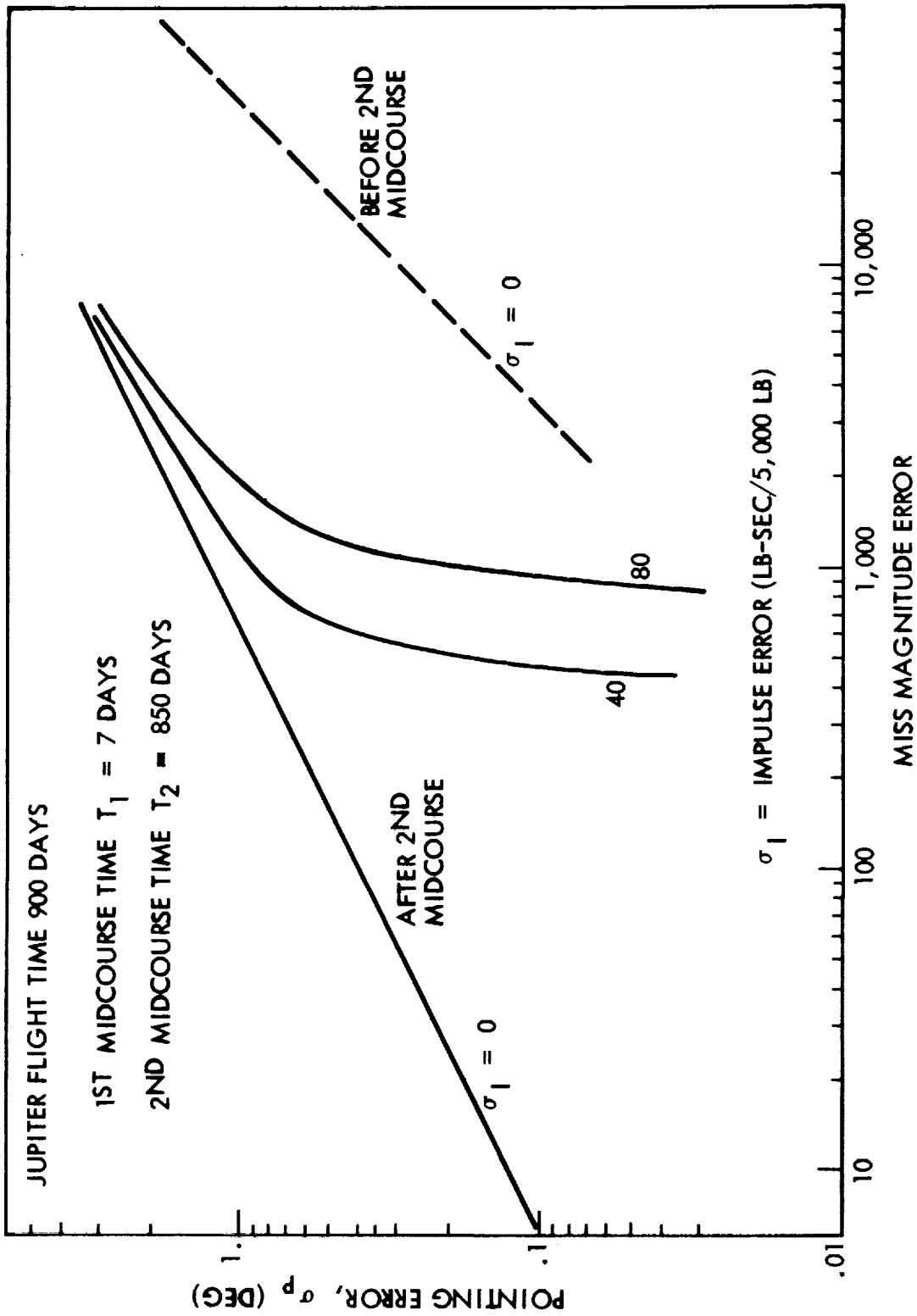


Fig. 47 Sensitivity of Residual Miss To Midcourse Errors for Jupiter Mission

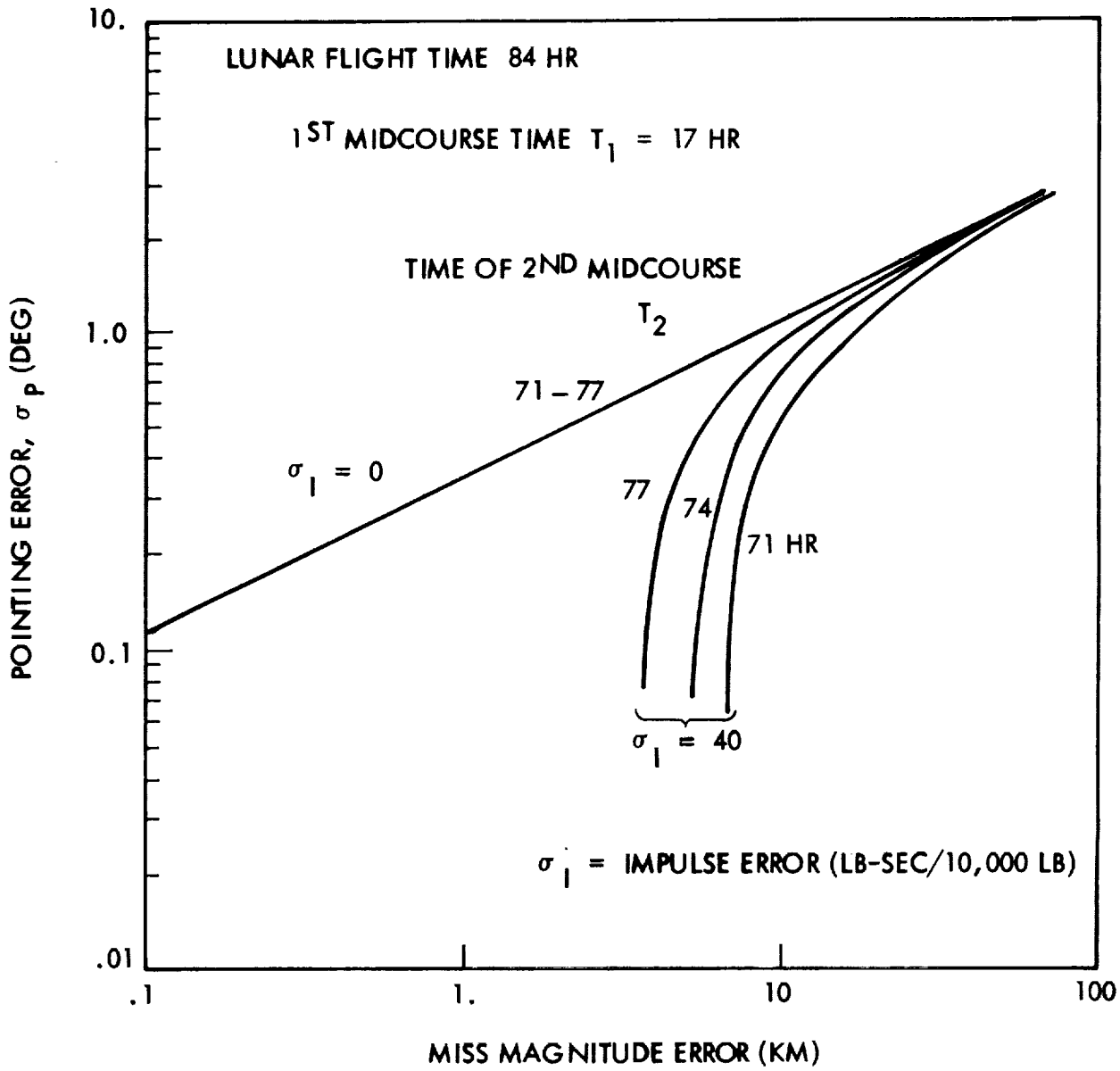


Fig. 48 Effects Of Second Midcourse Execution Time and Errors On Residual Miss For Lunar Mission

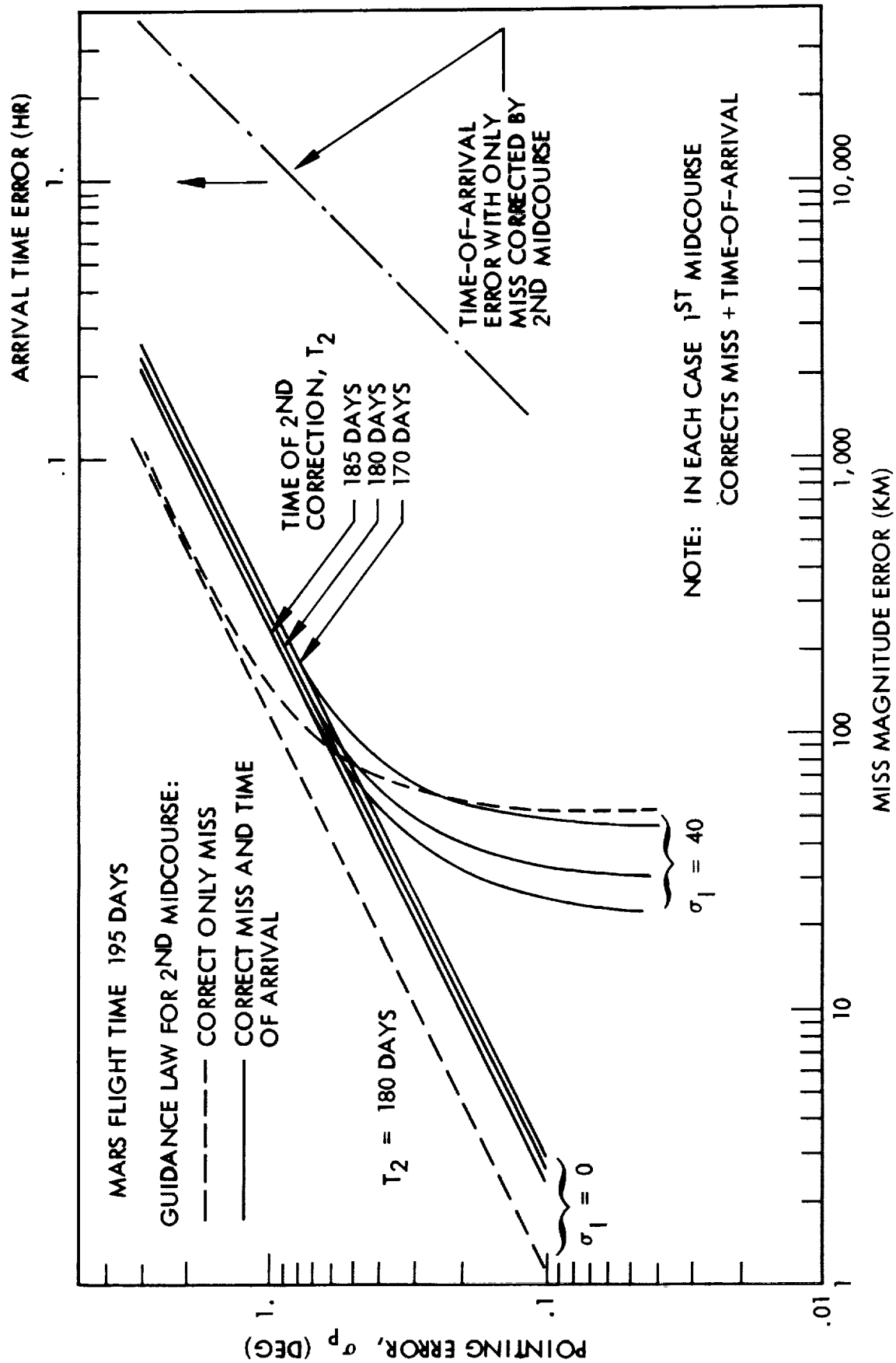


Fig. 49 Effects of Second Midcourse Execution Time, Midcourse Errors, And Guidance Law On Residual Miss And Arrival Time For Mars Mission

velocity for the Jupiter mission is considerably more sensitive to impulse errors than either the Lunar, Venus or Mars missions. The Venus mission shows the large effects of pointing error on second midcourse velocity. The second midcourse execution time is also seen to be of comparable importance to that of the error sources. Increasing the execution time has the same effects as increasing the magnitude of the errors. All of the data shown assume the first midcourse occurs at its nominal time. The effects on the second midcourse of varying the execution time of the first midcourse were generally insignificant.

The rms miss remaining after performing the second midcourse is presented in figs. 44 to 47 for each mission. Also shown for comparison is the miss existing prior to the second midcourse. Note that the residual miss after the second correction is sensitive to the errors in both the first and second corrections. In all cases, it is assumed that the error source deviations σ_P and σ_I take on the same values for both corrections.

It is seen from figs. 44 to 47 that the effects on residual miss of impulse errors relative to pointing errors is generally largest for the Jupiter mission and smallest for the Mars missions. For example, with a pointing error of about 0.5 deg, adding an impulse error of 40 units (see figures) increases the miss by about 12 percent for the Venus mission, 21 percent for the Mars mission, a factor of 2.4 for the Lunar mission, and a factor of 3.3 for the Jupiter mission. The actual values of miss in these cases is 145 KM, 80 KM, 6.8 KM, and 640 KM for the Venus, Mars, Lunar, and Jupiter missions, respectively.

The effects of varying the time of the second midcourse are shown in figs. 48 and 49 for the Lunar and Mars missions. In general, delaying the time of occurrence of the second midcourse was comparable to decreasing the magnitudes of the error sources, primarily the value of the impulse error. Fig. 49 also shows the effects of changing the guidance law of the second midcourse for the Mars mission. Data were obtained for the case where the second midcourse corrected only the miss

components and did not correct for arrival time error. It is seen that with this guidance law change, the pointing error effects are reduced considerably, however, the impulse error effects are magnified. The net advantages of this change becomes significant only when the pointing error is larger than about 0.5 deg. Associated with this change in guidance law is an increase in arrival time error. With arrival time corrected during the second midcourse, the residual arrival time error is the order of a few minutes. Without an arrival time correction in the second midcourse, the residual arrival time error will be larger than 0.6 hour for pointing errors larger than 0.5 deg.

The effects on residual miss of varying the first midcourse time were of no consequence.

3.5 SPACECRAFT/CONTROLLER DYNAMICS

Appropriate consideration is given to spacecraft dynamics through use of simplified rigid body modeling for decoupled pitch, roll, and yaw control channels. Conventionally oriented thruster assemblies for each channel are driven by separate controllers that operate on several attitude error signals. The closed loop dynamic behavior thus generated is discussed in this section and in Appendices A and B.

The assumption of inter-channel decoupling is justified by choosing principal axes for the three control axes and assuming that the motions and rates are either small (so that superposition applies) or can be made to occur sequentially.

3.5.1 Thruster Configuration

The requirements analysis task is based on a conventional thruster layout utilizing 12 nozzles. To avoid coupling between pitch, yaw, and roll control channels and sensitivity to spacecraft center of mass location, two thruster pairs separated by a distance L are used to generate pure control couples of $\pm FL$ ft-lb where F is the thrust in lb. In addition, the arrangement provides an attractive reliability advantage

through the incorporation of independent propellant feed systems with excess capacity for each set of three pairs.

Without serious loss of generality, the assumption of common F and L values for all three channels was made to make the study tractable. It is to be understood, however, that interplanetary missions may involve dual thrust levels which may or may not be generated by the same nozzles (as in the case of an all cold gas system with dual pressure levels).

The nominal thrusters configuration is assumed to be symmetrical pairs about the X-X (roll) axis, Y-Y (pitch) axis and Z-Z (yaw) axis.

3.5.2 Digital Pseudo Rate Controller

Dynamic response characteristics of a spacecraft/controller combination were established by digital simulation using a computer model of one control channel and a promising controller channel. The selected controller is a sampled data version of the standard pseudo rate controller. This class of controller is particularly well adapted to implementation on a time-shared onboard digital computer, a consideration appropriate to the spacecraft generation of interest. A detailed discussion of controller operation, parameter evaluation, and typical system transient behavior may be found in Appendix B.

The simulation effort described in Appendix B indicates that suitable controller parameter values can be found to produce satisfactory dynamic performance for all other system parameter combinations of interest. Consequently, changes in thruster force level, minimum on-time, spacecraft moment of inertia, etc., can be compensated by

suitable adjustment of controller parameter values. Of equal value was the demonstration that well defined maneuvers can actually be performed with propellant expenditures that exceed the ideal value by less than 25 percent.

3.5.3 External Torque Assumptions

The primary continuous disturbing torques of interest in computing attitude control system requirements are solar torques. Precise evaluation of solar torques is both difficult and strongly dependent on spacecraft design details. In addition, torque history during cruise mode depends to some extent on spacecraft orientation history in the characteristic limit cycle motion. In view of these practical difficulties for a generalized study, the usual recourse of designing to a constant level large enough to serve as a practical bound is employed. Based on Lockheed experience the following values are estimated for earth-moon space.

$$M_x = 0.2 \times 10^{-5} \text{ ft-lb}$$
$$M_y = 1.0 \times 10^{-5} \text{ ft-lb}$$
$$M_z = 1.0 \times 10^{-5} \text{ ft-lb}$$

In the case of Mars and Jupiter missions, application of the above numbers is additionally conservative since solar pressure depends on the inverse square of distance from the sun. On the other hand, the solar probe mission terminates at a distance of 0.2 astronomical units from the sun where solar pressure is a factor of 25 higher than in earth-moon space. Accordingly, an average torque that gives the same time integral over the mission as a parabolically increasing torque (based on a conservative constant speed approach to the sun) must be 9 times the actual initial torque. Therefore, the following values were used for the solar probe mission.

$$M_x = 1.8 \times 10^{-5} \text{ ft-lb}$$
$$M_y = 9.0 \times 10^{-5} \text{ ft-lb}$$
$$M_z = 9.0 \times 10^{-5} \text{ ft-lb}$$

Solar torques may be augmented in orbital cruise modes by gravity gradient torques. Gravity gradient torques depend on the spacecraft moment of inertia tensor, together with spacecraft orientation relative to the local vertical. Since the spacecraft remains sun/Canopus oriented while cruising in orbit, the relative orientation has large magnitude, periodic variations so that many of the cyclical motions induced by these torques may be within deadbands and thus not induce control response. Nevertheless, the combined solar and gravity gradient torque envelopes were increased for orbital cruise to account for an estimated secular effect.

$$M_x = 1.0 \times 10^{-5} \text{ ft-lb}$$
$$M_y = 5.0 \times 10^{-5} \text{ ft-lb}$$
$$M_z = 5.0 \times 10^{-5} \text{ ft-lb}$$

Disturbing torques generally exist about all body axes during periods of main engine burn. A ground rule specified in paragraph 3.2 is that the main engine has a thrust vector control system to compensate for pitch and yaw disturbance torques. It is further assumed that the pitch and yaw attitude control thrusters are normally disabled during main engine burns to prevent channel response with attendant expenditure of attitude control propellant. The roll channel of the attitude control system must counteract engine-generated roll disturbance torques, however. Values for these swirl torques are based on Mariner data scaled by engine thrust level, resulting in 3.0 ft-lb for a 5,000 lb thrust engine.

3.5.4 Limit Cycle Behavior

Appendix A is entitled "Attitude Control Limit Cycle Expenditure in Presence of Constant Applied External Torque." The following expression for cruise mode control impulse expenditure in a single channel is developed in this appendix.

$$\dot{Q} \; \frac{\text{lbf-sec)}}{\text{sec}} = \max. \; (2.25 \; \dot{Q}_o \, , \; 2 \, M/L)$$

from which

$$Q\ (\text{lbf-sec}) = \text{Max.}\ \frac{202.5\ (F\Delta)^2 L}{\pi I D},\ \frac{2M}{L}\ t$$

where

F = thruster average force level during a single pulse, lbf*

Δ = thruster minimum on time (pulse width), sec

L = thruster separation distance, ft

I = spacecraft moment of inertia, slug-ft^2

D = control deadband, ±deg

M = external torque, ft-lb

t = cruise duration, sec.

The second function argument dominates (and is therefore used) for large M where a one-sided single pulse thruster mode must exist to counteract the external torque. The first argument is a threshold large enough to include all peak expenditure rates for the many one/two-sided pulse patterns that may occur for smaller M. It is 2.25 times the level corresponding to a symmetric, two-sided, single pulse mode for $M = O$. Analysis details may be found in Appendix A.

The foregoing equation is useful in establishing the range of attitude control system parameters of interest for long interplanetary missions. First, we note that the threshold is parabolically related to F and Δ so that it is very expensive to let these parameters become much larger than the values which correspond to equality of the

*This value is not to be confused with the somewhat higher thrust level for a given nozzle which would occur in steady state operation. The correct interpretation is associated with the fact that $F\Delta \triangleq$ "minimum impulse bit."

two arguments. For the same reason, it is desirable that deadband be no smaller than is consistent with other mission requirements such as acceptable antenna gain loss. Secondly, it is instructive to equate the two arguments and solve for the resulting thrust level, denoted F_L. When $F > F_L$ the threshold term dominates cruise mode control expenditure; otherwise, the external torque dominates. We find

$$F_L = \frac{0.1761\sqrt{DIM}}{\Delta L} .$$

A tabular F_L evaluation for D, I, and L ranges of interest in this study is contained in Table 33. The torque bound given in paragraph 3.5.3, $M = 1.0 \times 10^{-5}$ ft-lb, and the smallest assumed minimum on time, $\Delta = 0.025$ sec, are used in the calculation. An immediate observation is that the indicated thrust levels are all small, the largest value in the table bring 0.668 lbf. More typically, the Mars missions, where $I = 3{,}500$ slug-ft^2, $D = \pm 0.5$ deg, and $L = 10$ ft are representative values, should have cruise thruster levels of 0.093 lbf or less. If $L = 25$ (e.g., thrusters on outer edges of solar panels) to reduce control expenditure the desirable thrust level drops to 0.037 lbf.

For reference purposes, the theoretical minimum weight of cold nitrogen gas required for a 200 day cruise is also shown in Table 33. The value is based on common M and I for all three control channels, and a specific impulse of 65 sec, i.e.,

$$W_{theo.} = \frac{6M}{L}\frac{t}{I_{sp}}$$

$$W_{theo.,\ 200\ days} = \frac{6M\ (1.728 \times 10^7)}{65L}$$

When combined with other requirements, the data in Table 33 can be used to establish the need for dual thruster levels. In the roll control channel, for example, there is a need to counteract a 3 ft-lb external torque with a single thruster operative (for reliability considerations). With $L = 10$ ft this implies $F \geq 0.6$ lbf. If $F = 0.6$ is

Table 33

THRUST, F_L, FOR $M = 1 \times 10^{-5}$ FT-LB, $\Delta = 0.025$ SEC

Thruster Separation, L, (ft)		5	10	15	20	25
$W_{theoretical}$, 200 Days Cold N_2 Gas, (lb)		3.190	1.595	1.063	0.798	0.638
D (Deadband) (± deg)	I (slug-ft^2)	F_L (lbf)	F_L (lbf)	F_L (lbf)	F_L (lbf)	F_L (lbf)
0.5	500	0.071	0.035	0.024	0.018	0.014
	1,500	0.122	0.061	0.041	0.031	0.024
	2,500	0.158	0.079	0.053	0.039	0.032
	3,500	0.186	0.093	0.062	0.047	0.037
	4,500	0.211	0.106	0.071	0.053	0.042
1.0	500	0.100	0.050	0.033	0.025	0.020
	1,500	0.173	0.086	0.058	0.043	0.035
	2,500	0.223	0.111	0.074	0.056	0.045
	3,500	0.264	0.132	0.088	0.066	0.053
	4,500	0.300	0.150	0.100	0.075	0.060
5.0	500	0.223	0.111	0.074	0.056	0.045
	1,500	0.386	0.193	0.129	0.097	0.077
	2,500	0.498	0.249	0.166	0.125	0.100
	3,500	0.590	0.295	0.197	0.147	0.118
	4,500	0.668	0.334	0.223	0.167	0.134

used for a cruise mode to Mars, however, the roll channel expenditure will exceed the necessary minimum amount by a factor of over 40. In the pitch and yaw channels, the attitude control system has a requirement to remove initial tipoff rates of the order of 1.5 deg/sec. Other space programs have required that this rate be brought to zero before an excursion of 10 degrees from the initial orientation has occurred. If two

thrusters are turned on continuously to perform this task it can be shown that the minimum F is given by

$$F = \frac{\pi I \dot{\theta}^2}{360 \theta L}$$

where

$\dot{\theta}$ = initial tipoff rate, deg/sec

θ = permissable excursion, deg

Again, using $I = 3,500$, $L = 10$ we find $F = 0.687$ lbf. If this F value is used for a cruise mode to Mars, however, the pitch and yaw channel expenditures will exceed the necessary minimum amount by a factor of over 50.

Finally, one other point of view is illuminating. Consider the spacecraft with $F_L = 0.093$ and requiring a theoretical minimum of 1.595 lb of cold nitrogen for a 200 day cruise. If $F = 2.0$ lbf (a reasonable level for maneuvering and attitude control during main engine burns) the spacecraft could not even cruise for 1/2 day on the theoretical minimum amount of gas using the high thrust level.

Based on the preceding analysis, it is concluded that dual thruster levels are required on all missions with the probable exception of the lunar lander. In this latter case, the cruise duration is short enough that added weight associated with $F > F_L$ is offset by the weight and reliability disadvantages of an added system.

3.6 ATTITUDE CONTROL SYSTEM REQUIREMENTS AND MODES OF OPERATION

This section of the report is devoted primarily to the presentation of results from the attitude control system requirements studies of five missions. The first three paragraphs contain brief reviews of essential background material. A computer program used to generate the main body of output data is described in the fourth paragraph. The next five paragraphs describe individual mission results and are followed by a comparative summary discussion.

3.6.1 Operating Mode Review

The attitude control system operating modes which significantly affect propellant expenditure are listed below.

(1) Tipoff Rate Removal.
The removal of spacecraft tipoff rates about all three control axes following separation from the launch vehicle.

(2) Acquisition Transients and Searches.
The starting and stopping of spacecraft roll, pitch, and yaw rates in sun/Canopus acquisition processes.

(3) Commanded Turns.
The maneuvers generated by sequential turn commands at prescribed rates. A roll-pitch-roll sequence was taken as the standard with unwinding accomplished in reverse order.

(4) Transit Cruise
Limit cycle expenditures based on the most likely maximum solar torque levels about each axis.

(5) Orbital Cruise
Limit cycle expenditures based on the most likely maximum sum of solar and gravity gradient torque levels about each axis.

(6) Main Engine Burns
The counteracting of main engine swirl (roll) torques during escape, midcourse, orbit insertion, and orbit trim engine burns.

3.6.2 ACS Propulsion Data

A review of thrusts and propellant performance capabilities was made for thruster sizes from less than 0.1 lbf to 100 lbf and for cold N_2 gas, N_2H_4, N_2O_4/MMH, FLOX/CH_4, and F_2/H_2 propellants.

It was concluded that in the thrust range of interest for limit cycle operation (less than 0.1 lbf) only the cold N_2 gas system and a cold N_2H_4 gas system were attractive. Performance of a system in this low-level thrust range could be improved by heating

the gas, but this alternative was not analyzed. Specific impulse of the cold N_2 gas was assumed to be a constant 65 lbf-sec/lb m for all thrust levels and thruster on-times of interest. Specific impulse of cold N_2H_4 gas was assumed to be a constant 100 lbf-sec/lb m.

Operation of high-level thrusters was assumed to occur only in the steady state during initial orientation, reorientation, or maneuvers. Steady state specific inpulse was assumed to vary as shown on Fig. 50 for the propellants considered. Specific impulse for a high-level thrust cold N_2 gas system was assumed constant at 65 lbf-sec/lbm, identical to that for the low-level system.

In the analysis of ACS propellant requirements, a cold N_2 gas low-level system was combined with each candidate high-level system for alternate combinations of mission thrust level, thruster moment arm, low-level thruster on-time, and limit cycle dead-band. Propellant requirements are presented later in this section.

3.6.3 Basic Assumptions

The assumptions itemized below complete the necessary data base for the numerical study to follow. Certain items have been stated elsewhere but are included in the interest of completeness. Others actually evolved after preliminary iterations of the requirements study.

(1) A common thruster separation distance, L, is used for all three control channels.

(2) A common thruster force, F, is used for all three control channels except as noted in items 3 and 4.

(3) Two separate thruster force levels are employed for all missions except the intermediate size lunar lander where a single level is used.

(4) The pitch and yaw control channels are disabled during main engine burn periods while an independent thrust vector control system controls pitch and yaw attitude.

(5) The ratio of high level to low level thrust is 20 for all missions.

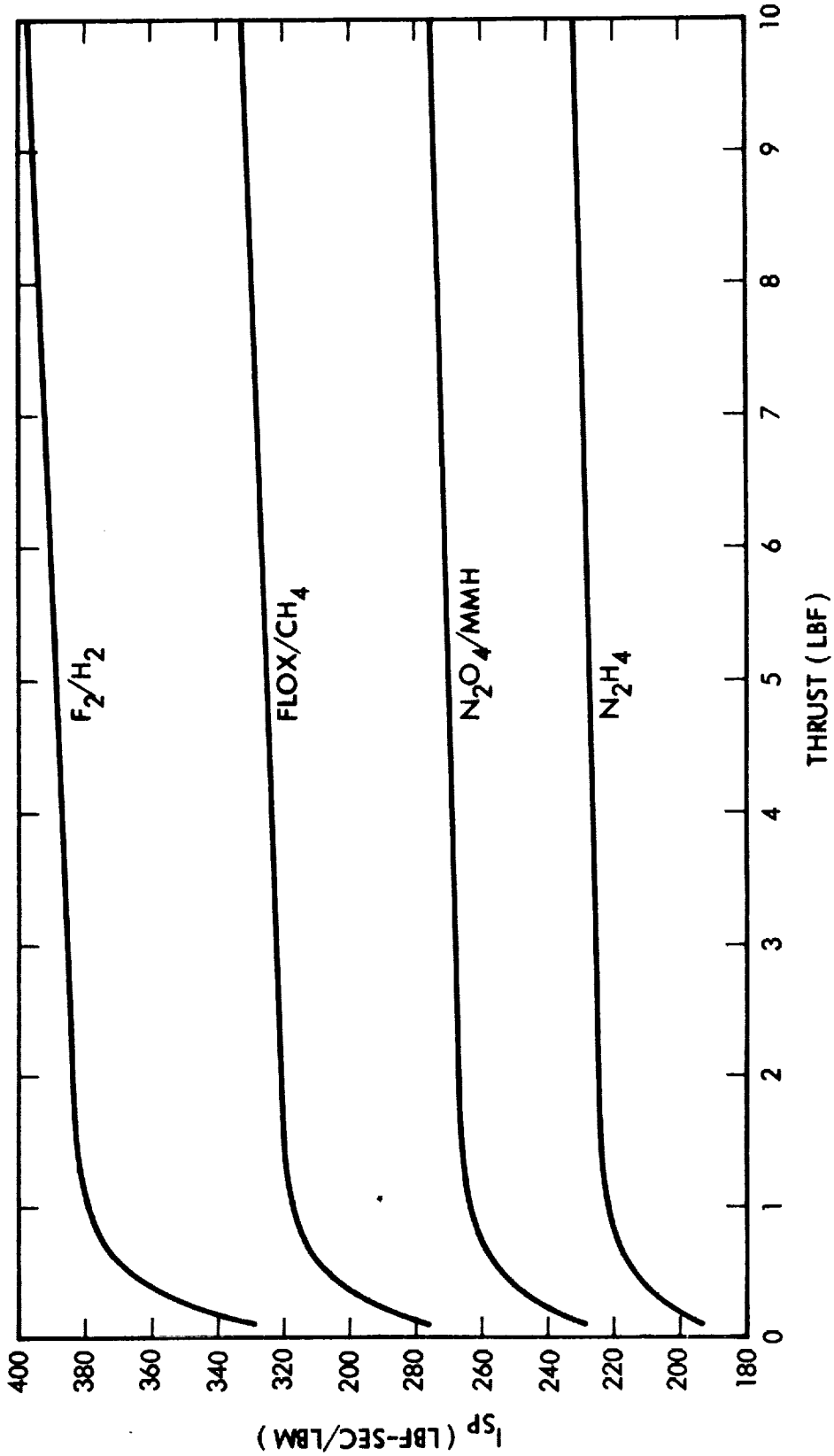


Fig. 50 Thruster Steady State Performance

(6) All extended duration cruise modes are performed with low-level thrusters using cold nitrogen gas with a specific impulse of 65 sec.

(7) All non-cruise modes are performed with high-level thrusters.

(8) The maximum attitude control system torque capability of interest is 20 ft-lb.

(9) Computer outputs do not include an allowance for leakage or any reserve to handle malfunctions.

(10) Three representative thruster separation distances of interest are L = 5, 10, 25 ft.

(11) Three representative minimum on-times of interest (applies only to low-force-level nitrogen thrusters) are Δ = 0.025, 0.050, 0.075 sec.

(12) Three cruise mode deadband levels of interest are D = ±0.5, 1.0, 5.0 deg. An exception is the intermediate size lunar lander spacecraft where one half of these values are used for orbital cruise.

3.6.4 Description of Requirements Analysis Program

A Univac 1108 program in Fortran V was modified to perform the requirements calculations for each mission and to prepare SC-4020 plotter tapes for graphical solution display.

3.6.4.1 Mission Elements and Spacecraft Configurations. Individual missions were constructed by appropriate combination of mission elements where each element could be designated as one of four types. The four types are closely related to the six operational modes listed in paragraph 3.6.1.

Type 1: Tipoff Rate Removal
Same as item (1) of paragraph 3.6.1

Type 2: Commanded Turns
Handles both acquisitions and commanded turns, i.e., items (2) and (3) of paragraph 3.6.1

Type 3: Cruise with Constant External Torques
Handles items (4) and (5) of paragraph 3.6.1.

Type 4: Large External Torques
Handles main engine burns, i.e., item (6) of paragraph 3.6.1.

The several spacecraft moment of inertia tensors, (i.e., sets of I_x, I_y, I_z presented in section 3.3) of interest in each mission are stored in the computer and indexed by a" configuration number" so that individual mission elements refer to appropriate values. Exceptions are mission elements of Type 4. since control expenditures during main engine burns are independent (essentially of spacecraft moments of inertia.

3.6.4.2 Propellant Performance Tables. Specific impulse functions of thrust for each of four propellants are shown in section 3.6.2. These functions are represented in the computer program by conventional linear interpolation between specific points. A constant value of 65 sec (i.e., independent of thrust level) is used for cold nitrogen gas.

3.6.4.3 Parameter Variation Technique. The purpose of the program is to generate graphic displays of mission control requirements that show the effects of variations in the following four parameters:

(1) Thruster force level (both high and low levels constrained by a fixed 20:1 ratio),
(2) Thruster minimum on-time,
(3) Thruster separation distance.

Based on the selected plot layout, the first two parameters above are automatically varied over a specified range during a single computer run, while fixed values are input for the last two. Accordingly, each plot corresponds to a specific combination of deadband and thruster separation distance; a set of several computer runs is required to display the effect of variation in these parameters. The selected set of nine runs per mission is described in paragraph 3.6.5.

3.6.4.4 Mission Element Input Data, Calculations. Input data required for each type of mission and associated calculations is discussed below.

Type 1 Tipoff Rate Removal

Input:
(1) Configuration number
(2) Dynamic factor, K_F
(3) Initial roll rate, ω_x, deg/sec
(4) Initial pitch rate, ω_y, deg/sec
(5) Initial yaw rate, ω_z, deg/sec

Using the configuration number to select the appropriate set of moment of inertia (I) values and the dynamic factor (always 2.0 for type 1 elements in this study) to scale ideal requirements, the following equation for determining impulse is evaluated

$$Q = \frac{K_F \pi}{90L} \left(\omega_x I_x + \omega_y I_y + \omega_z I_z\right) \text{ lbf-sec}$$

This maneuver is performed by the high level thrusters, so associated weights for various propellants are computed for later summation using the general form

$$W = Q/I_{SP} \text{ lb}, \; I_{SP} = f(F)$$

with F = high level thruster force level.

Type 2 Commanded Turns

Input:
(1) Configuration number
(2) Dynamic factor, K_F
(3) Unwind factor, K_U
(4) Number of maneuvers factor, K_R
(5) First axis designator, i
(6) First rate, ω_i, deg/sec
(7) Second axis designator, j

(8) Second rate , ω_j , deg/sec

(9) Third axis designator , k

(10) Third rate , ω_k , deg/sec

The configuration number is again used to select the proper moment of inertia set. The three factors are self explanatory. The designators i , j , k can each be x , y , or z to indicate roll, pitch, or yaw turns, respectively.

$$Q = \frac{K_F K_R \pi (1 + K_U)}{45\,L} \left(\omega_i I_i + \omega_j I_j + \omega_k I_k\right) \text{ lbf-sec}$$

Observe that the type 2 format is actually a generalized form of type 1, except that both starting and stopping of the turn is assumed. Provision for multiple maneuvers and then unwinds is also made.

Type 2 maneuvers are performed by the high level thrusters so propellant weight calculations are identical to those discussed under type 1.

<u>Type 3</u> Cruise with Constant External Torque

Input: (1) Configuration number

(2) Duration , t , days

(3) Roll deadband factor , K_{DX}

(4) Roll torque , M_x , ft-lb

(5) Pitch deadband factor , K_{DY}

(6) Pitch torque , M_y , ft-lb

(7) Yaw deadband factor , K_{DZ}

(8) Yaw torque , M_z , ft-lb

The configuration number has the same meaning as before. The deadband factors, all normally 1.0, are applied to nominal deadband. In the intermediate size lunar landing spacecraft $K_{DX} = K_{DY} = K_{DZ} = 0.5$ for the orbital cruise element, only.

Cruise requirements are based on expressions in Appendix A.

$$\dot{Q}_{oi} = \frac{90\,(0.05\,F\Delta)^2\,L}{\pi\,I_i\,K_{Di}\,D} \quad , i = x, y, z$$

$$Q = 86400\,t\left[\mathrm{Max}\left(2.25\,\dot{Q}_{ox}, \frac{2\,M_x}{L}\right) + \mathrm{Max}\left(2.25\,\dot{Q}_{oy}, \frac{2\,M_y}{L}\right) + \mathrm{Max}\left(2.25\,\dot{Q}_{oz}, \frac{2\,M_z}{L}\right)\right] \text{ lbf-sec}$$

With the exception of the lunar mission, cruise mode operation is performed by the low level thrusters using cold nitrogen gas. The associated gas weight is therefore simply

$$W = Q/65 \text{ lb} .$$

<u>Type 4</u> Large External Torques

Input:
(1) Dynamic Factor, K_D
(2) Duration, t, sec
(3) Roll torque, M_x, ft-lb
(4) Pitch torque, M_y, ft-lb
(5) Yaw torque, M_z, ft-lb

This type of element provides attitude control torques to counteract external torques applied for a specified duration. A dynamic factor (2.0 used in this study) is included to handle transient effects such as noise, non-zero frequency content, etc., in the actual torque.

$$Q = \frac{2 K_D t}{L} (M_x + M_y + M_z)$$

Note that $M_y = M_z = 0$ in this portion of the study.

Since the control torques are supplied by the high level thrusters, propellant weight calculations are identical to those described for type 1.

3.6.4.5 Summary. Separate summation of low and high level control impulse and propellant weight for the several mission elements is accomplished in a straightforward manner. No attempt is made to present results for individual mission elements. Output plots are presented and discussed in the following sections.

3.6.5 Mars (or Venus) Orbiter Mission

Three spacecraft configurations are of interest for the Mars Orbiter mission as described in Section 3.3. The moments of inertia for this configuration are shown below:

Moment of Inertia	Configuration		
	Mars 1	Mars 2	Mars 3
I_x slug-ft^2	3079	1999	1753
I_y slug-ft^2	2885	1739	1250
I_z slug-ft^2	3580	1789	1268

Twelve mission elements are used to construct the mission as follows.

1. Type 1 with configuration Mars 1.
   Tipoff rates of 0.5, 1.5, and 1.5 deg/sec, about the X, Y, and Z axes respectively, are removed using a dynamic factor of 2.0.
2. Type 2 with configuration Mars 2.
   Four acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.

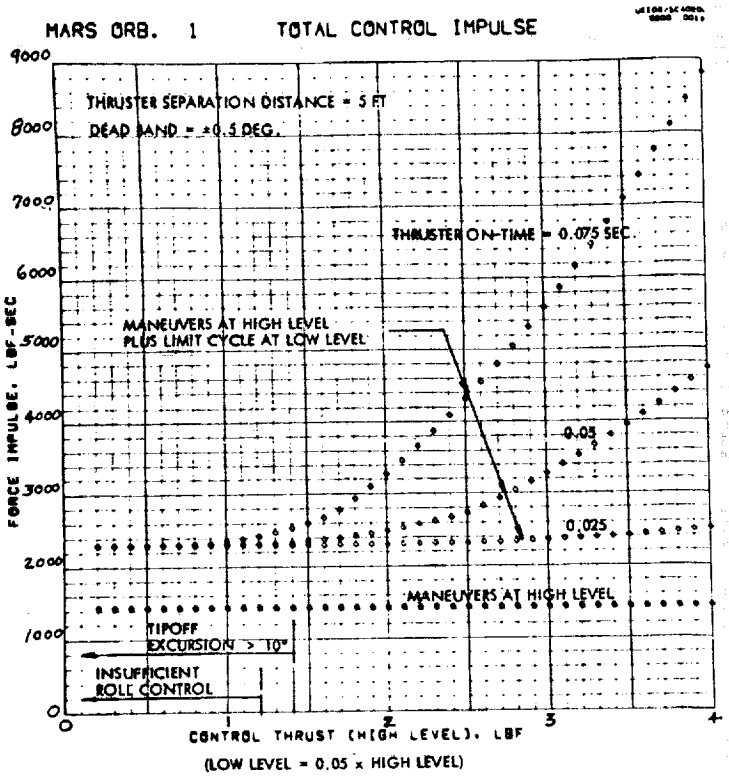
3. Type 3 with configuration Mars 1.
   A 195 day transit cruise with unity deadband factors and solar torques of (0.2, 1.0, and 1.0) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.
4. Type 2 with configuration Mars 1.
   Commanded roll-pitch-roll turns of 0.2 deg/sec with dynamic factor = 1.25. To simulate two turns with unwinds and one without set $K_u = 1$, $K_R = 2.5$.
5. Type 2 with configuration Mars 1.
   Commanded roll-pitch-roll unwind of 0.2 deg/sec with dynamic factor = 1.25. Set $K_u = 1$, $K_R = 0.5$.
6. Type 4 with dynamic factor = 2.
   Roll torque of 0.3 ft-lb for 155 sec, corresponding to 500 lbf main engine burns.
7. Type 4 with dynamic factor = 2.
   Roll torque of 3.0 ft-lb for 300 sec, corresponding to 5000 lbf main engine burn.
8. Type 3 with configuration Mars 2.
   A 10 day orbital cruise with unity deadband factors and solar/gravity gradient torques of (1, 5, and 5) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.
9. Type 2 with configuration Mars 2.
   A commanded roll-pitch-roll turn with unwind based on 0.2 deg/sec rates and a dynamic factor of 2.0. Capsule is separated during the maneuver.
10. Type 2 with configuration Mars 2.
    Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
11. Type 3 with configuration Mars 3.
    A 180 day orbital cruise with unity deadband factors and solar/gravity gradient torques of (1, 5, and 5) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.
12. Type 2 with configuration Mars 3.
    Four acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.

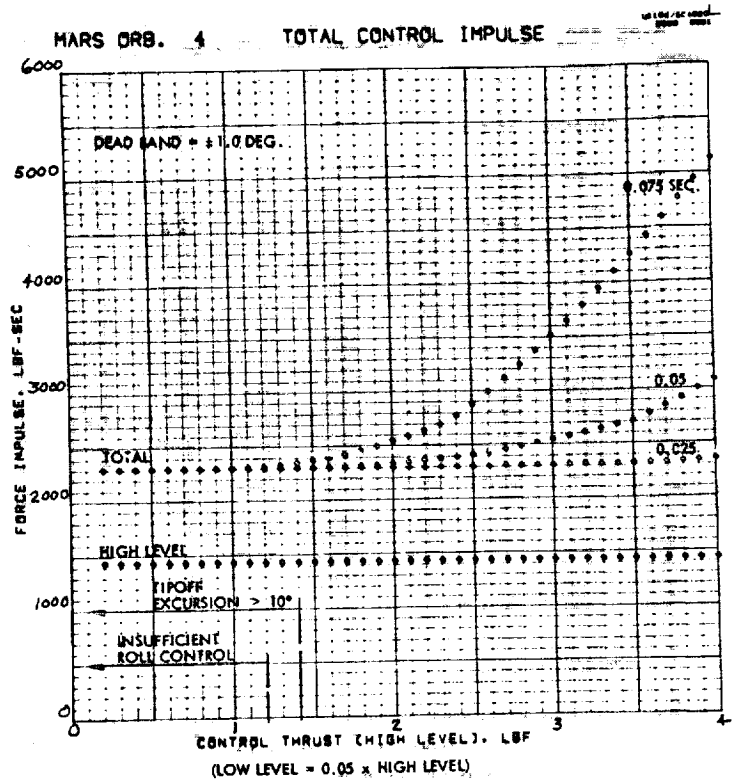
Nine combinations of thruster separation distance (L) and deadband (D) were used per the following table:

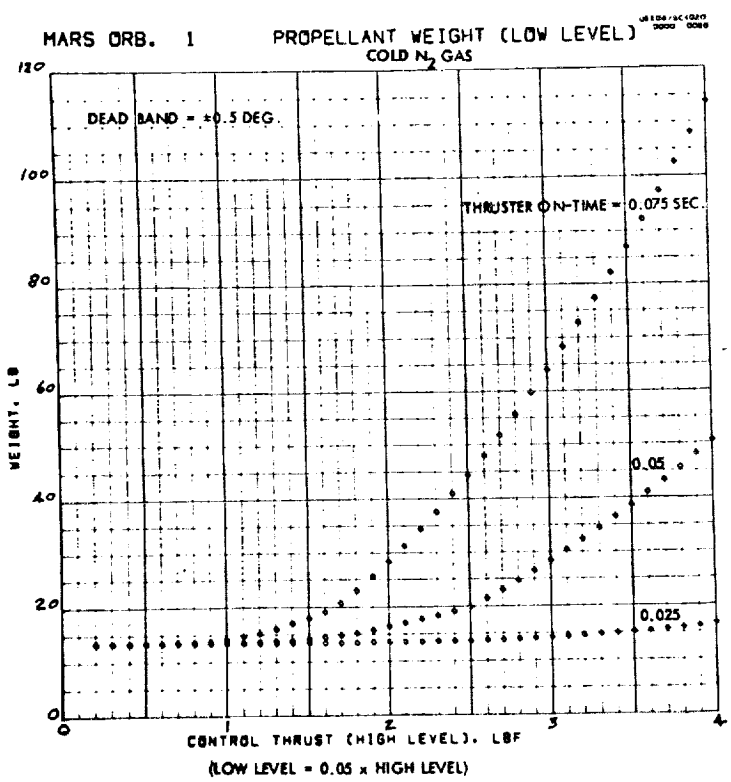
Run Designation	L (ft)	D (± deg)
Mars Orb. 1	5	0.5
Mars Orb. 2	10	0.5
Mars Orb. 3	25	0.5
Mars Orb. 4	5	1.0
Mars Orb. 5	10	1.0
Mars Orb. 6	25	1.0
Mars Orb. 7	5	5.0
Mars Orb. 8	10	5.0
Mars Orb. 9	25	5.0

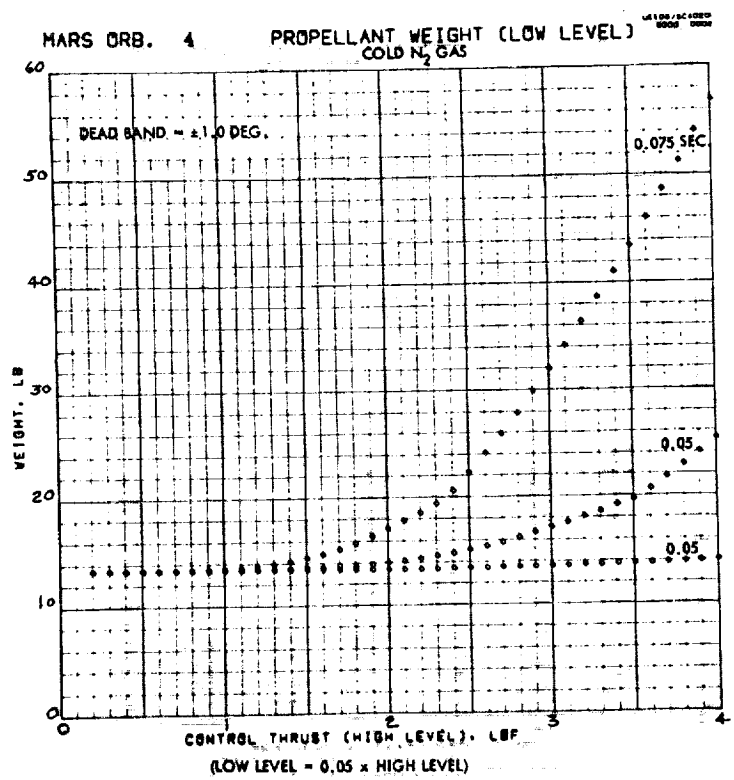
Computer output is displayed in three composite figures, each pertaining to a particular thruster separation distance. Figure 51 shows the results for $L = 5$ ft in seven plots. Three plots give total control impulse as a function of thruster force level and minimum on time. The portion of control impulse associated with maneuvers at high level is shown on each of these three plots – note that this portion is independent of F, Δ, and D as indicated in paragraph 3.6.4 equations. For $L = 10$ ft, we find $Q_{HIGH\ LEVEL} = 1433$ lb-sec. As F is increased the cruise mode expenditure departs from the fixed level associated with external torques (871 lb-sec). For a given deadband and thruster separation distance (i.e., D and L) this departure tends to occur at a fixed value of $F\Delta$, so low Δ implies departure at high F and conversely as can be seen in the figure. Another predictable trend is that larger deadbands correspond to departure at higher F values.

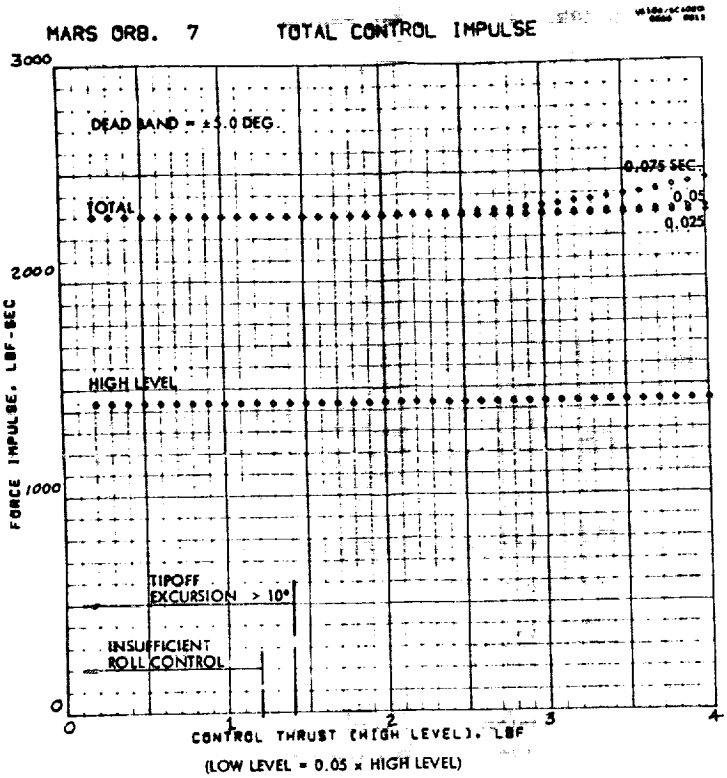
Three companion plots give the required weight of cold nitrogen gas for the low level thrusters. It can be seen that these plots are simply obtained by dividing the corresponding low level impulse requirement by 65 sec. The weight value associated with the threshold is therefore $871/65 = 13.39$ lb.

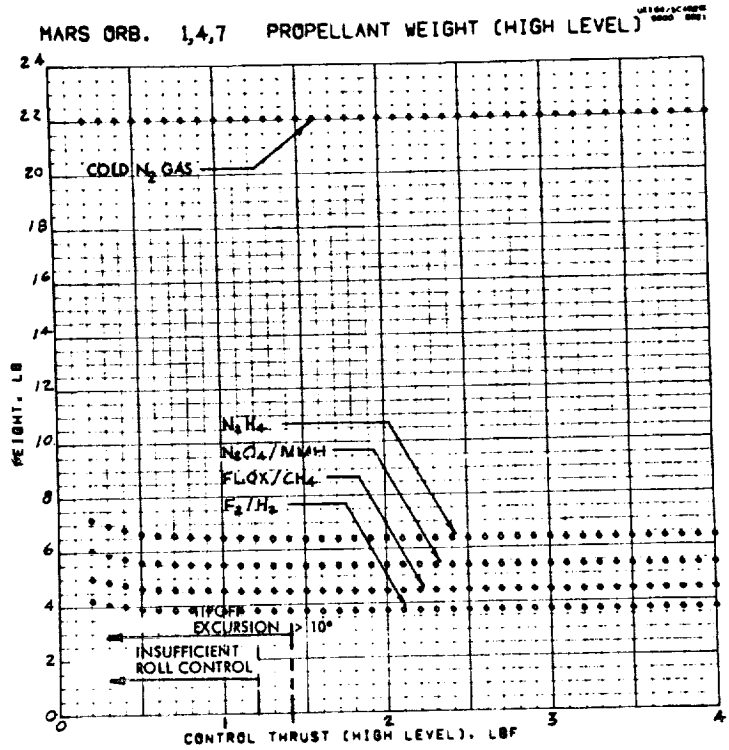

MARS ORB. 1 TOTAL CONTROL IMPULSE
THRUSTER SEPARATION DISTANCE = 5 FT
DEAD BAND = ±0.5 DEG.
THRUSTER ON-TIME = 0.075 SEC.
MANEUVERS AT HIGH LEVEL
PLUS LIMIT CYCLE AT LOW LEVEL
0.05
0.025
MANEUVERS AT HIGH LEVEL
TIPOFF
EXCURSION > 10°
INSUFFICIENT
ROLL CONTROL
FORCE IMPULSE, LBF-SEC
CONTROL THRUST (HIGH LEVEL), LBF
(LOW LEVEL = 0.05 x HIGH LEVEL)


MARS ORB. 4 TOTAL CONTROL IMPULSE
DEAD BAND = ±1.0 DEG.
0.075 SEC.
0.05
0.025
TOTAL
HIGH LEVEL
TIPOFF
EXCURSION > 10°
INSUFFICIENT
ROLL CONTROL
FORCE IMPULSE, LBF-SEC
CONTROL THRUST (HIGH LEVEL), LBF
(LOW LEVEL = 0.05 x HIGH LEVEL)


MARS ORB. 1 PROPELLANT WEIGHT (LOW LEVEL)
COLD N2 GAS
DEAD BAND = ±0.5 DEG.
THRUSTER ON-TIME = 0.075 SEC.
0.05
0.025
WEIGHT, LB
CONTROL THRUST (HIGH LEVEL), LBF
(LOW LEVEL = 0.05 x HIGH LEVEL)


MARS ORB. 4 PROPELLANT WEIGHT (LOW LEVEL)
COLD N2 GAS
DEAD BAND = ±1.0 DEG.
0.075 SEC.
0.05
0.05
WEIGHT, LB
CONTROL THRUST (HIGH LEVEL), LBF
(LOW LEVEL = 0.05 x HIGH LEVEL)





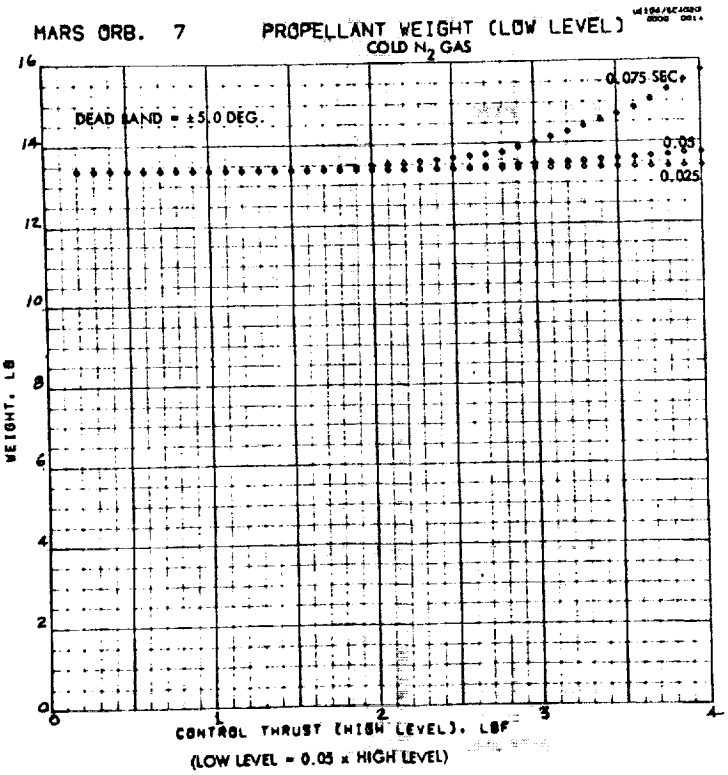


Fig. 51 Mars Orbiter ACS Requirements
Thruster Separation = 5 ft.

The seventh plot shows the required weight of high level propellant. Only one plot is required since this result is independent of deadband. A thrust-independent value of 1434/65 = 22.06 lb of cold nitrogen gas would be required for high level maneuvers. Lesser amounts of other propellants would be required with a slight dependence on F observable owing to I_{SP} dependencies.

Note the relative importance of selecting small, fast, low level thrusters. For D = ±0.5 deg we compare the combination $F_{LOW\ LEVEL} = 0.05$ lbf ($F_{HIGH\ LEVEL} = 1.0$ lbf), $\Delta = 0.025$ sec where the low level control impulse is 875 lb-sec to the combination $F_{LOW\ LEVEL} = 0.2$ ($F_{HIGH\ LEVEL} = 4.0$), $\Delta = 0.075$ sec where low level control impulse is 7398 lb-sec. This latter value represents an increase of 750% and exceeds the high level requirement by 415%.

In Fig. 52 are displayed a similar set of plots for L = 10 ft. The thrust range has been altered to maintain a maximum torque capability of 20 ft-lb. With this modification and a halving of the ordinate scales, the two sets of plotted points can essentially be overlaid.[1] This being true, all previous trend comments apply. This pattern extends to the third sets of plots for L = 25 shown in Fig. 53.

The above mentioned pattern provides a convenient method of estimating requirements for other thruster separation distances. For example, suppose we desire to find the total control impulse requirements for L = 12 ft, $\Delta = 0.050$ sec, and D = ±1.0 deg at $F_{HIGH\ LEVEL} = 1.5$, $F_{LOW\ LEVEL} = 0.075$ lbf. We use the plot for run 5 where L = 10. Since L = 12 the abscissa point marked 1.8 corresponds to $F_{HIGH\ LEVEL} = 1.5$ and is the point entered on the plot to read Q = 1390 lbf-sec. But this indicated value must also be scaled by the L ratio so that the desired result is 1390/1.2 ≅ 1160 lbf-sec.

Two identified constraints are indicated on each plot. One pertains to the roll control thrust level required for a single high level thruster to counteract the assumed 3.0 ft-lb

[1]Exceptions are weight requirements for propellants with thrust level dependent specific impulse.

main engine swirl torque. The other corresponds to the yaw excursion following a 1.5 deg/sec tipoff rate. For dual thrusters having force levels lower than the indicated value, spacecraft yaw excursion will exceed 10 deg before the rate is nulled.

A summary of Mars Orbiter mission control impulse requirements is presented in Table 34. The term "min low level" used in summary tables implies "subject to minimum force level constraints".

Table 34

MARS ORBITER-CONTROL IMPULSE SUMMARY

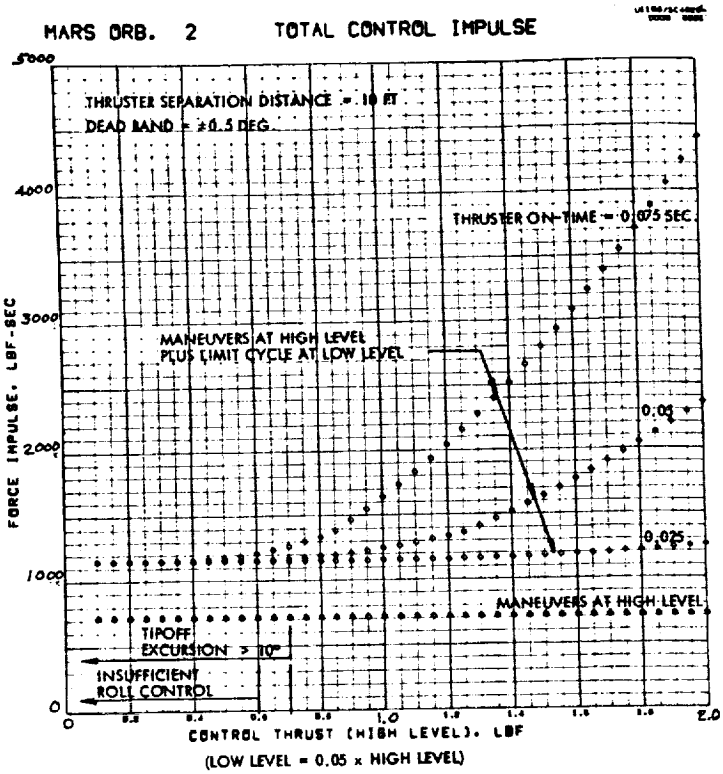
Thruster Separation L (ft)	5	10	25
High Level Impulse (lb-sec)	1433 ($F \geq 1.4$ lbf)	717 ($F \geq 0.7$ lbf)	287 ($F \geq 0.28$ lbf)
Min. Low Level Impulse (lb-sec)	871 ($F < 0.1$ lbf)	435 ($F < 0.062$ lbf)	174 ($F < 0.025$ lbf)
Min. Total Impulse (lb-sec)	2304	1152	461

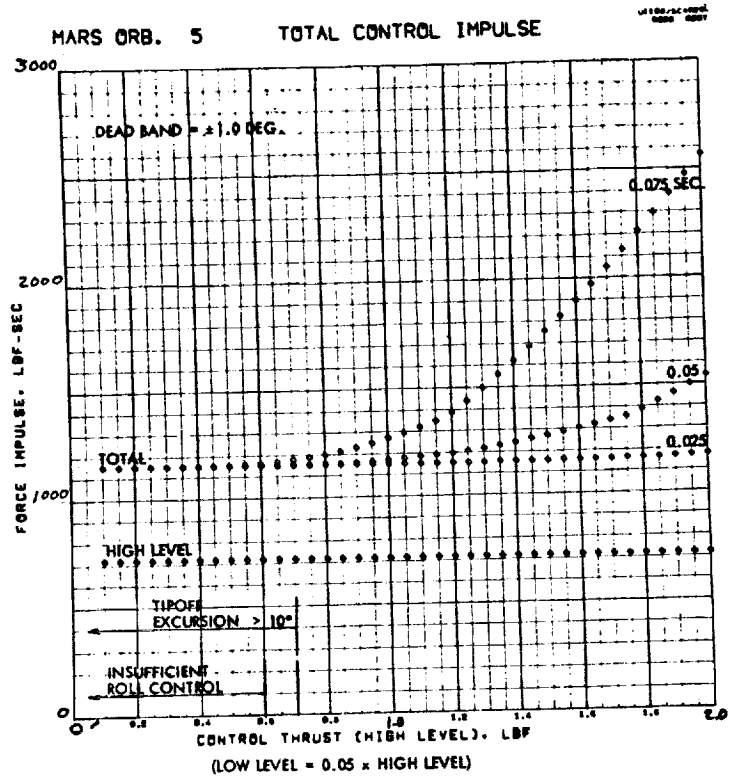
Note: Deadband = ± 0.5 deg

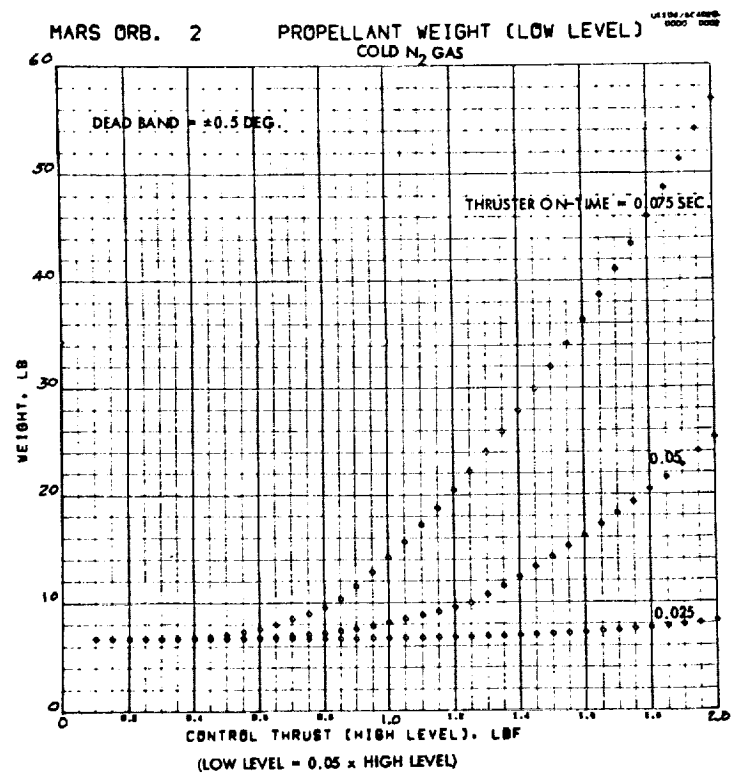
3.6.6 Solar Probe Mission

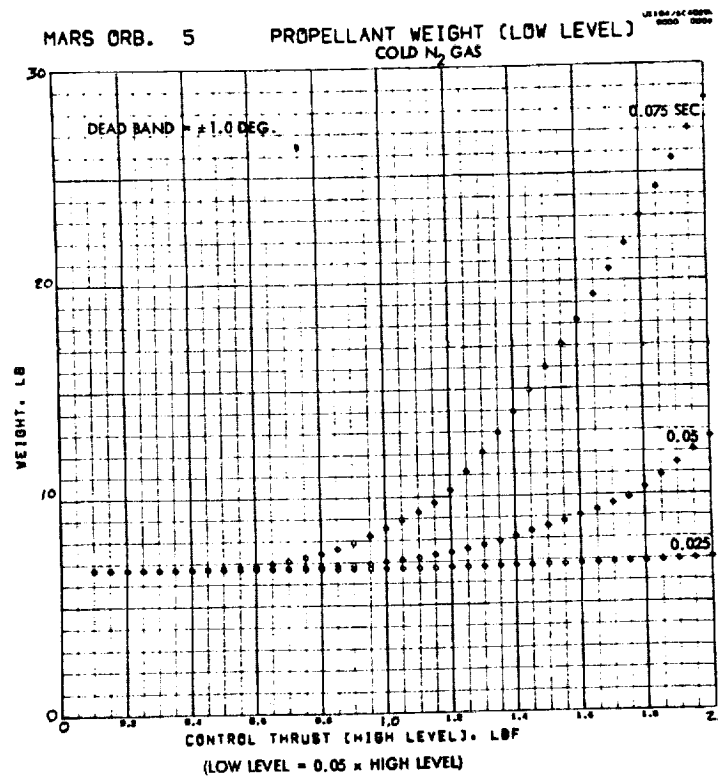
There are only two spacecraft configurations of interest for the Solar Probe mission; one with full tanks at tipoff, the other consisting only of the bus and solar panels as discussed in Section 3.3. The inertias for these two configurations are shown below.

Moment of Inertia	Configuration	
	Solar Probe 1	Solar Probe 2
I_x slug-ft^2	1852	535
I_y slug-ft^2	1207	285
I_z slug-ft^2	1902	285









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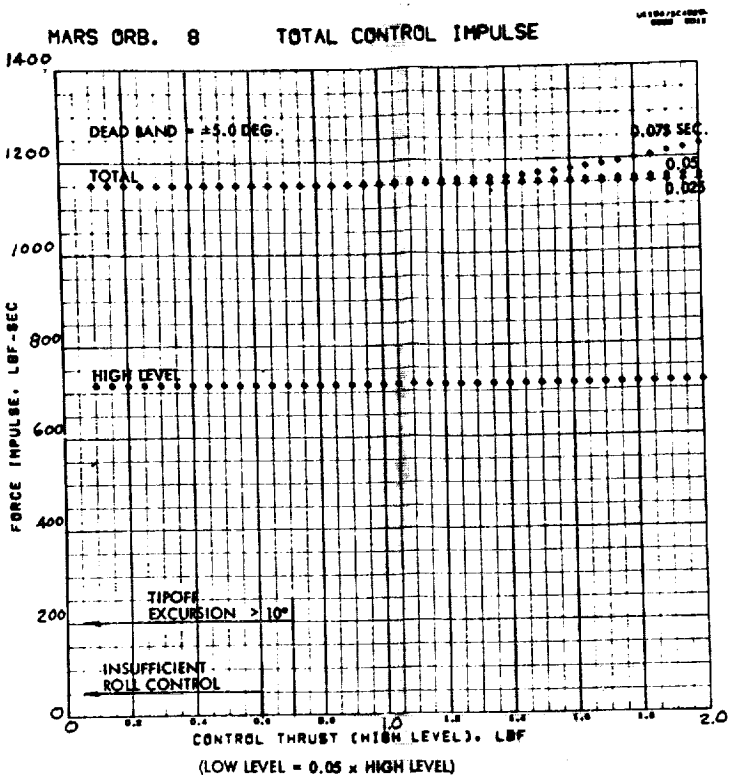

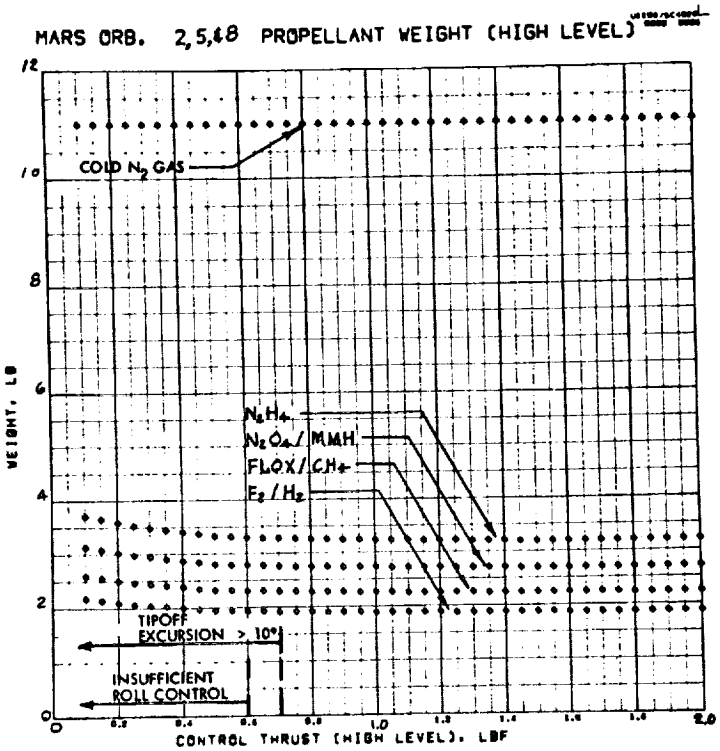

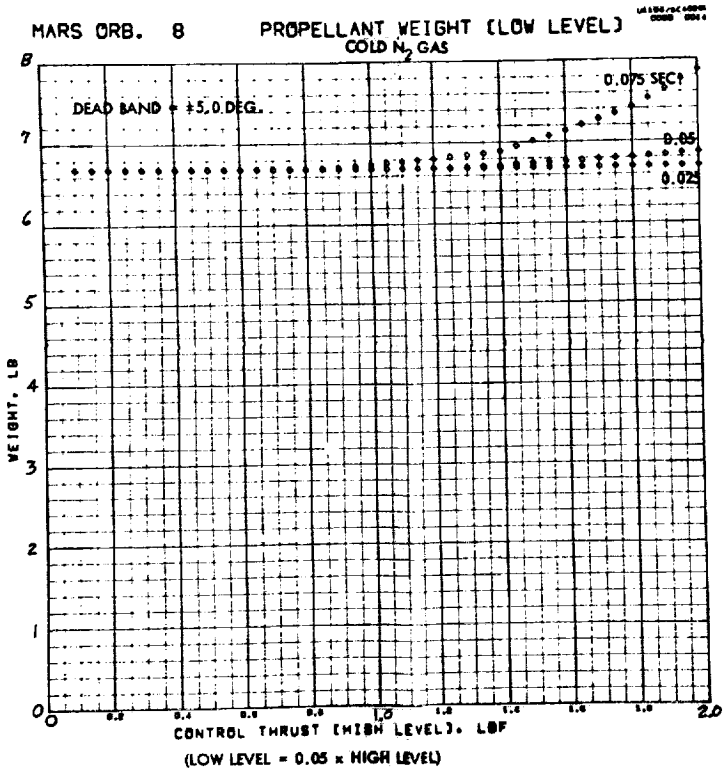

Fig. 52 Mars Orbiter ACS Requirements
Thruster Separation = 10 Ft.

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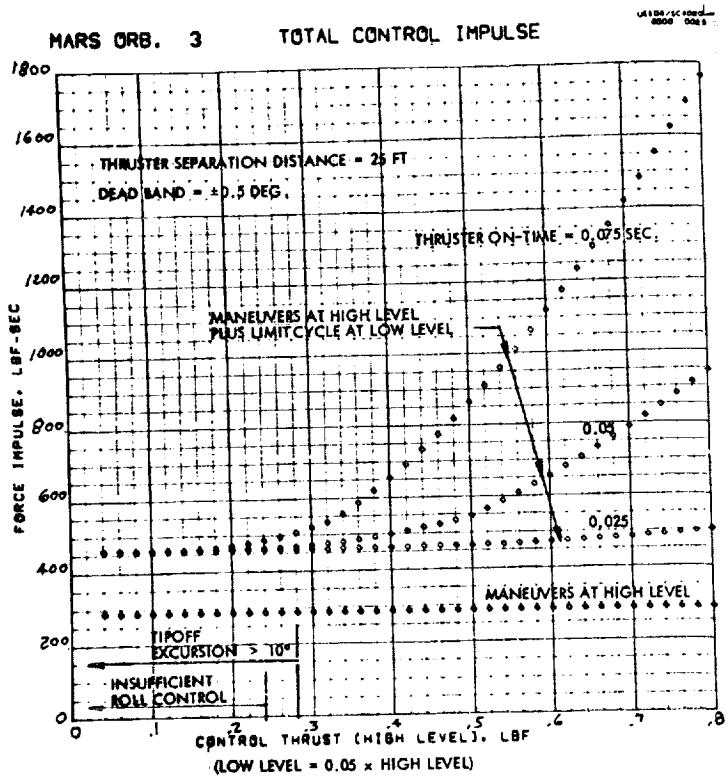

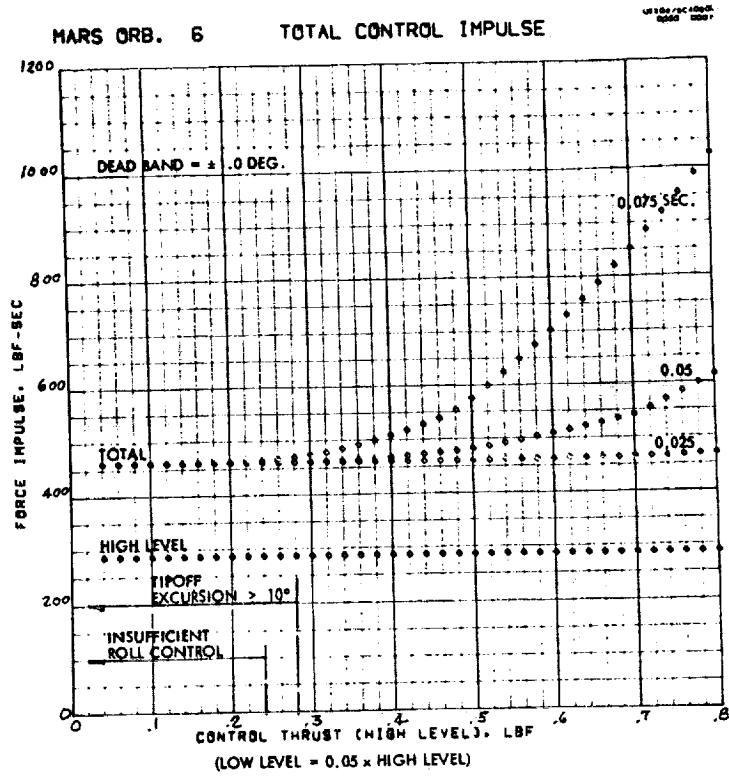

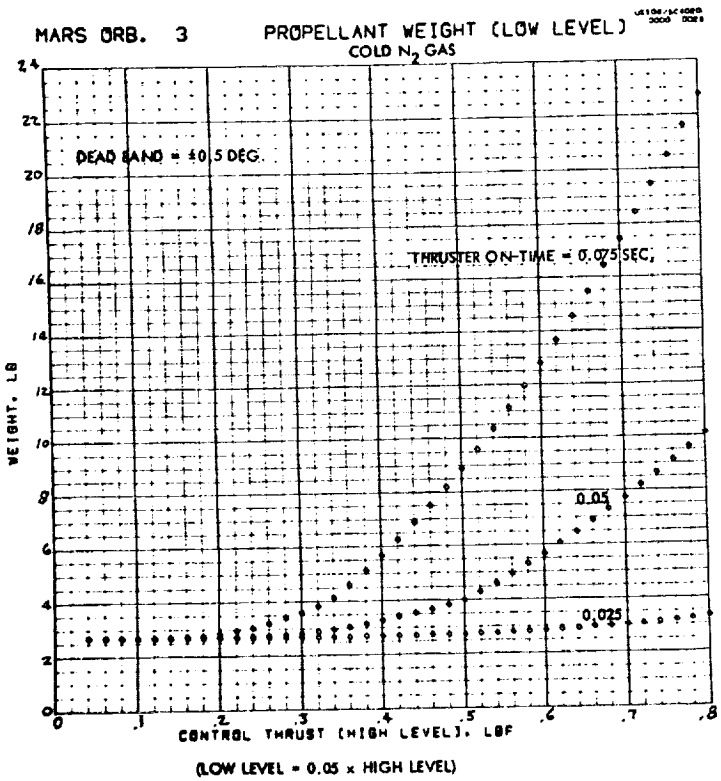

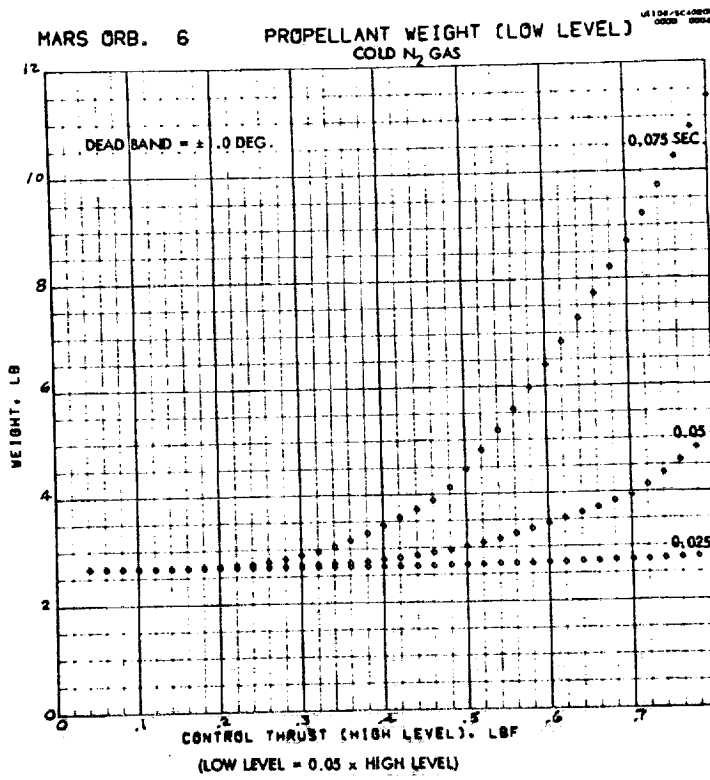

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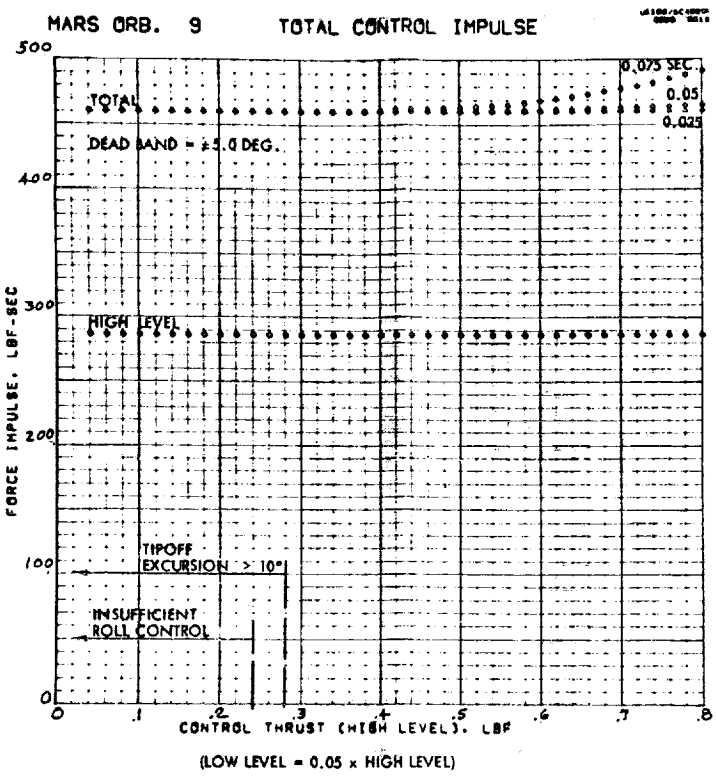

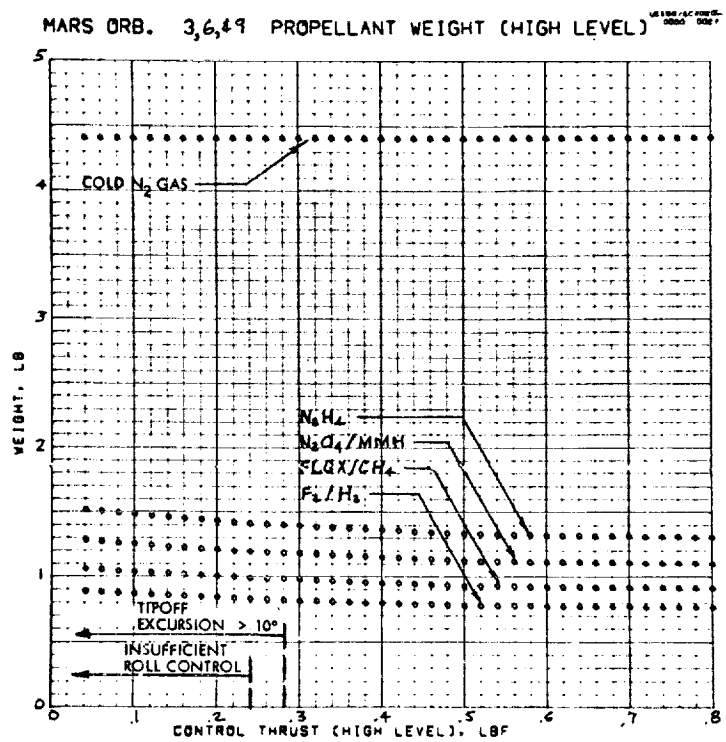

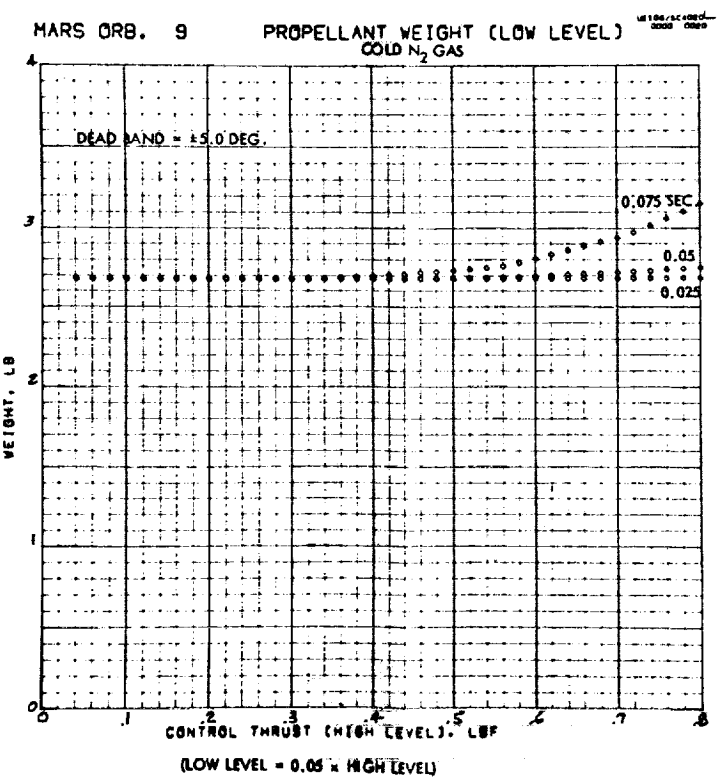

Fig. 53 Mars Orbiter ACS Requirements
Thruster Separation = 25 ft.

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Six mission elements are used to construct the mission.

1. Type 1 with Configuration Solar Probe 1.
   Tipoff rates of 0.5, 1.5, and 1.5 deg/sec, respectively, are removed using a dynamic factor of 2.0.
2. Type 2 with Configuration Solar Probe 1.
   Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
3. Type 4 with dynamic factor = 2.
   Roll torque of 3.0 ft-lb for 322 sec, corresponding to 5000 lbf main engine burn.
4. Type 2 with Configuration Solar Probe 2.
   Two commanded roll-pitch-roll turns of 0.2 deg/sec with dynamic factor = 1.25 unwinds.
5. Type 2 with Configuration Solar Probe 2.
   Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
6. Type 3 with Configuration Solar Probe 2.
   A 730 day transit cruise with unity deadband factors and solar torques of $(1.8, 9.0, 9.0) \times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.

Nine computer runs were made using the same parameter combinations and run numbering system described in the Mars Orbiter mission, paragraph 3.6.5. Computer results are also displayed in three figures each with seven plots. Qualitative comments previously made about trends and patterns continue to apply.

An immediate observation to be made from Figs. 54, 55, and 56 is that the low level requirements exceed the high level by a factor of at least 5. The high level requirements are only 64% of the comparable values for the Mars orbiter mission while the minimum low level requirements are 574% of the corresponding Mars orbiter mission value. Accordingly, the emphasis in the solar probe mission must be on the cruise mode performance and minimization of external torque levels.

One of the constraints indicated on each plot remains unchanged since the assumed 3.0 ft-lb main engine swirl torque exists as before. The thrust level pertaining to yaw excursion after tipoff is lower, however, since the solar probe spacecraft moments of inertia are appreciably lower.

A summary of Solar Probe mission control impulse requirements is presented in Table 35.

Table 35

SOLAR PROBE – CONTROL IMPULSE SUMMARY

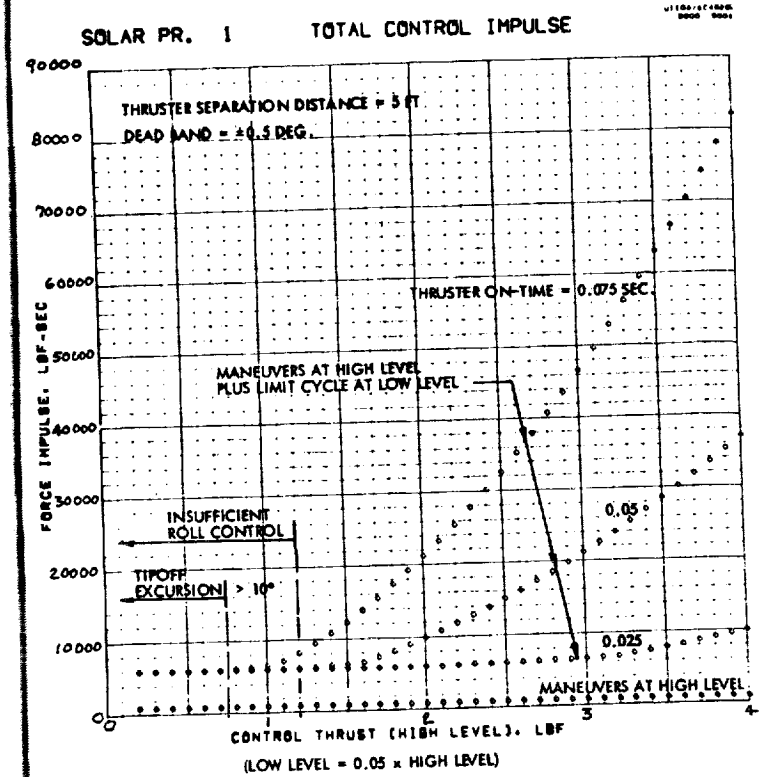
Thruster Separation L (ft)	5	10	25
High Level Impulse (lb-sec)	921 ($F \geq 1.2$ lbf)	461 ($F \geq 0.6$ lbf)	184 ($F \geq 0.24$ lbf)
Min. Low Level Impulse (lb-sec)	4995 ($F < 0.11$ lbf)	2498 ($F < 0.075$ lbf)	999 ($F < 0.023$ lbf)
Min. Total Impulse (lb-sec)	5916	2959	1183

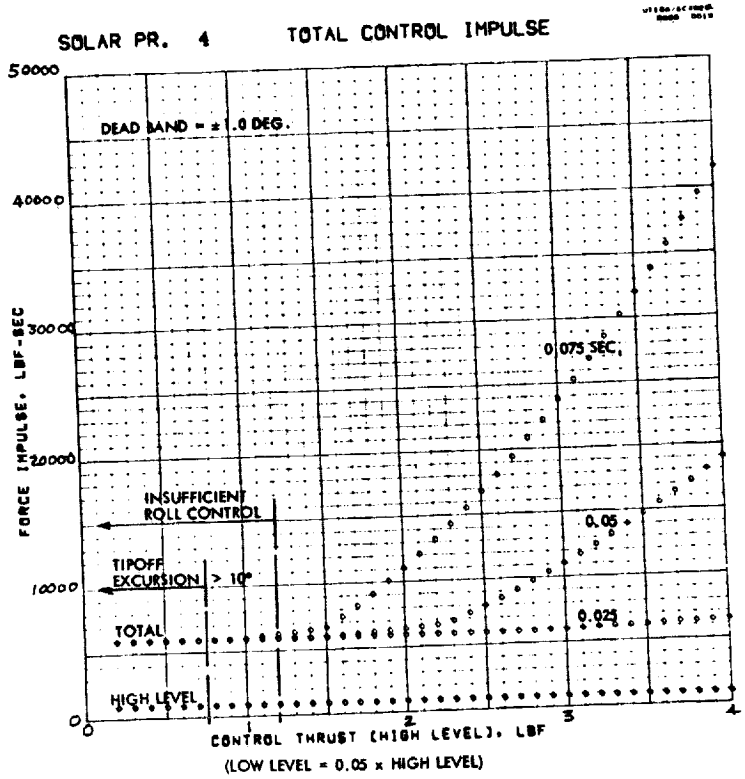
Note: Deadband = ±0.5 deg

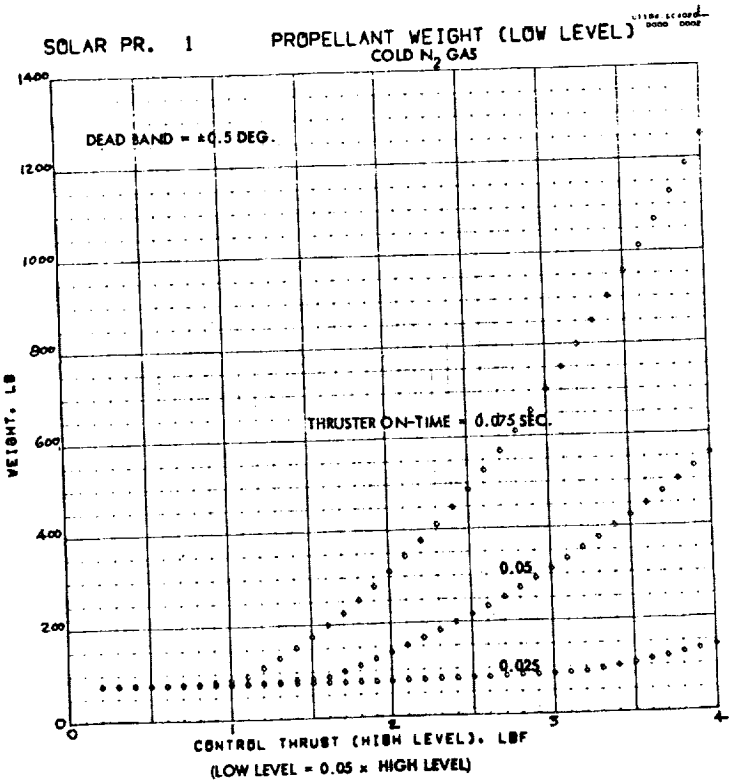
3.6.7 Jupiter Orbiter Mission

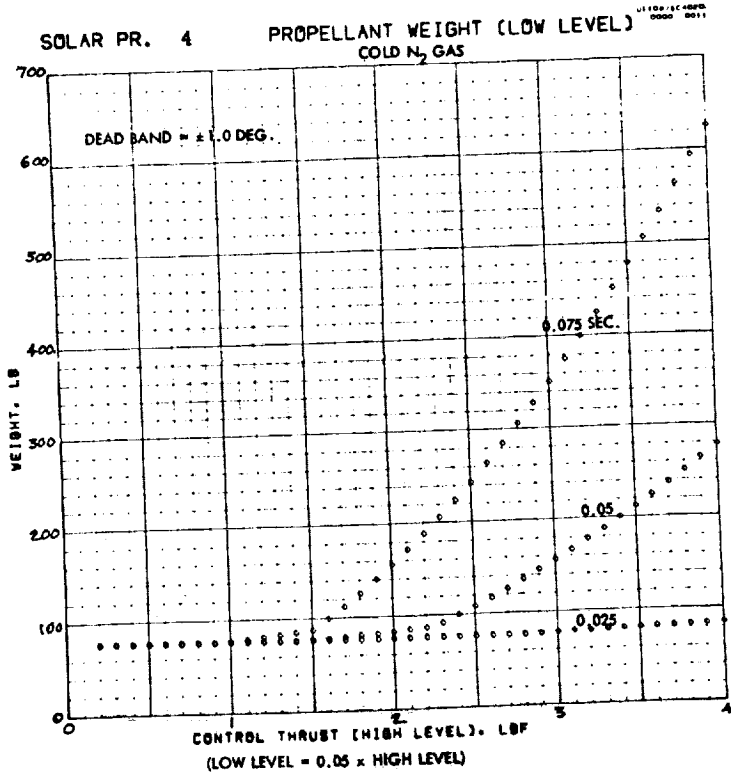
Three spacecraft configurations are of interest to the Jupiter Orbiter mission per discussion in Section 3.3. The inertias for this configuration are shown below:

Moments of Inertia	Configuration		
	Jupiter Orbiter 1	Jupiter Orbiter 2	Jupiter Orbiter 3
I_x slug-ft^2	1823	1138	696
I_y slug-ft^2	1188	1015	628
I_z slug-ft^2	1883	1333	646









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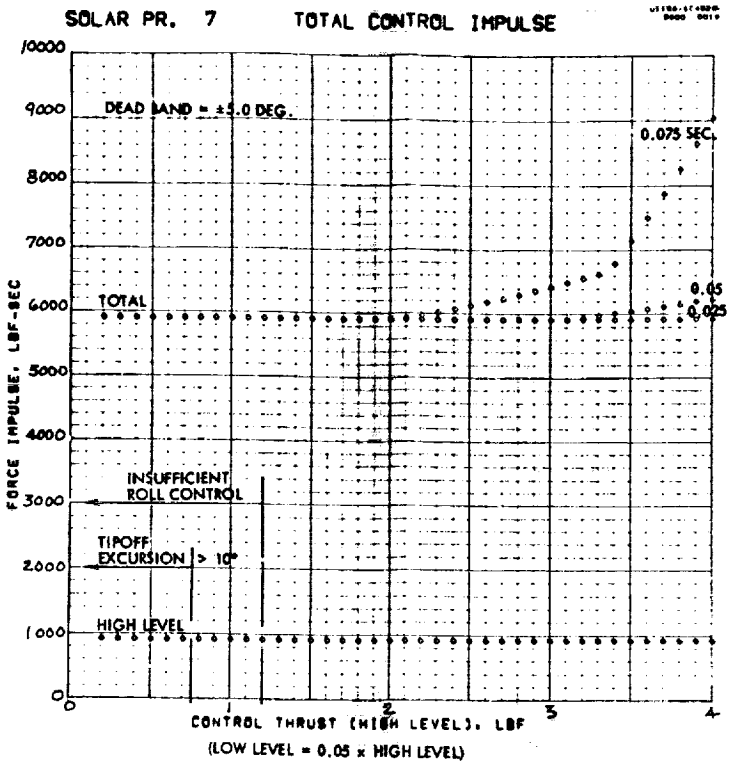

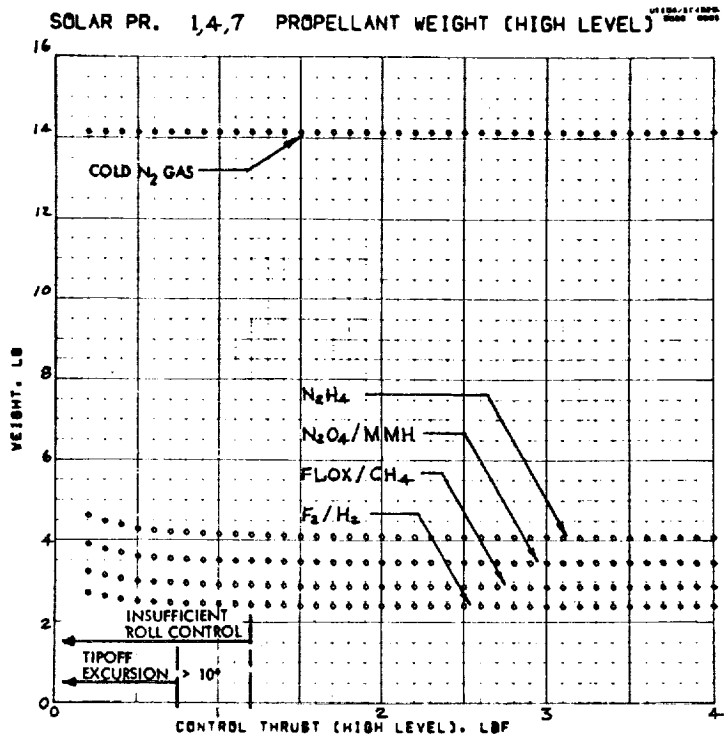

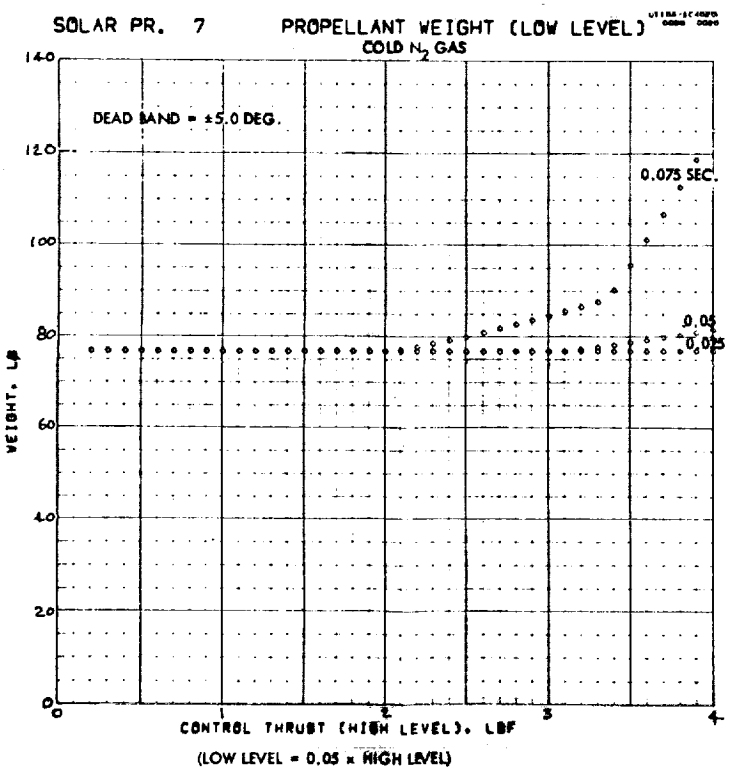

Fig. 54 Solar Probe ACS Requirements
Thruster Separation = 5 ft

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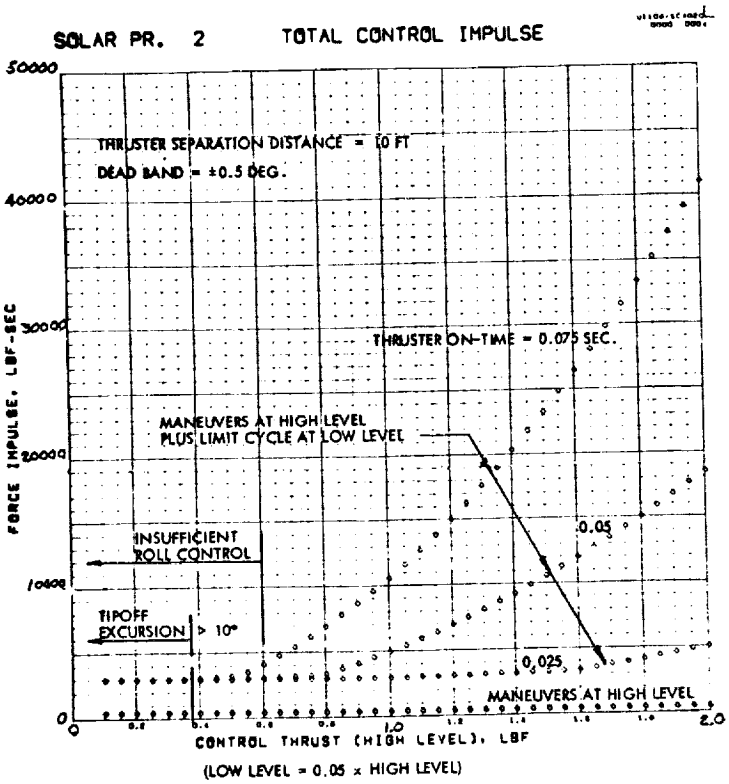

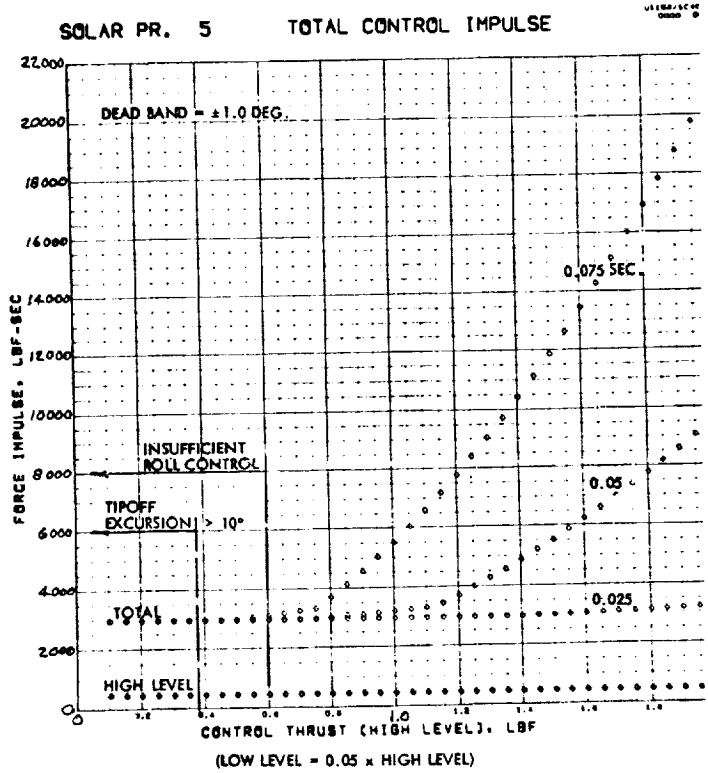

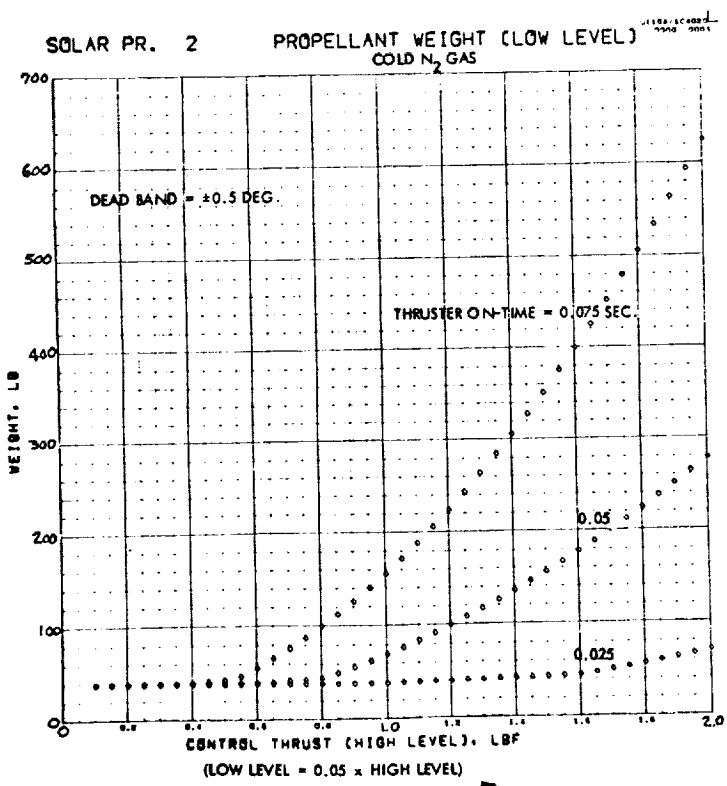

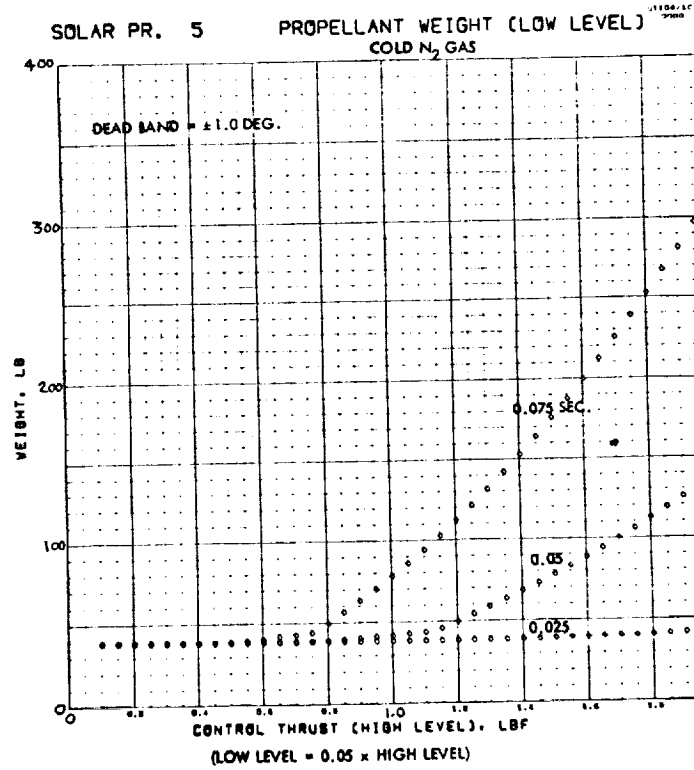

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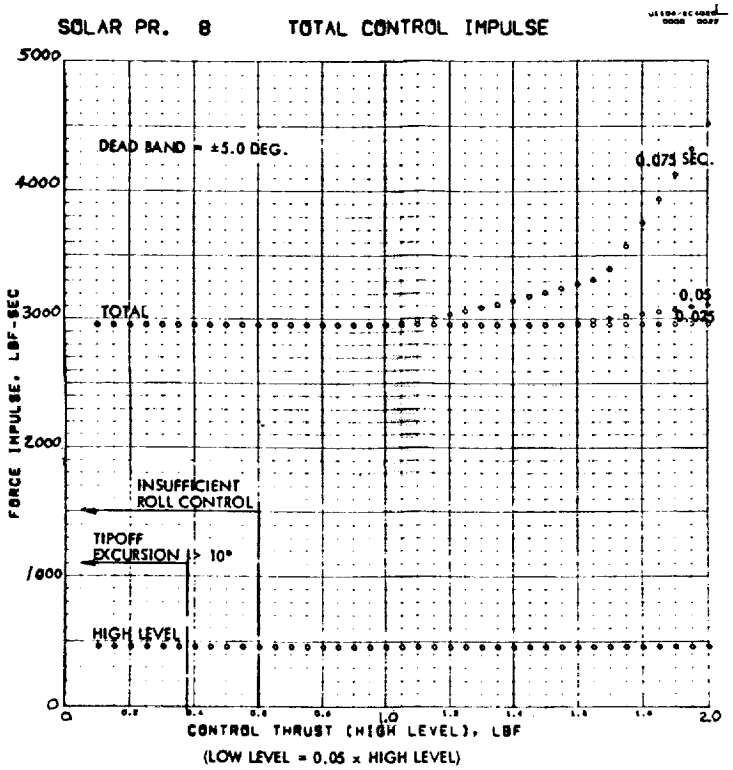

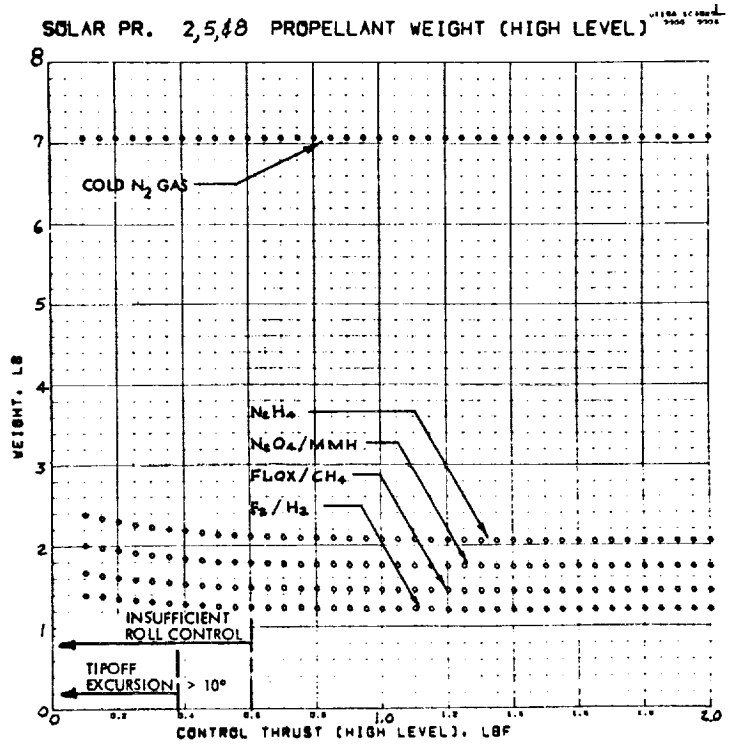

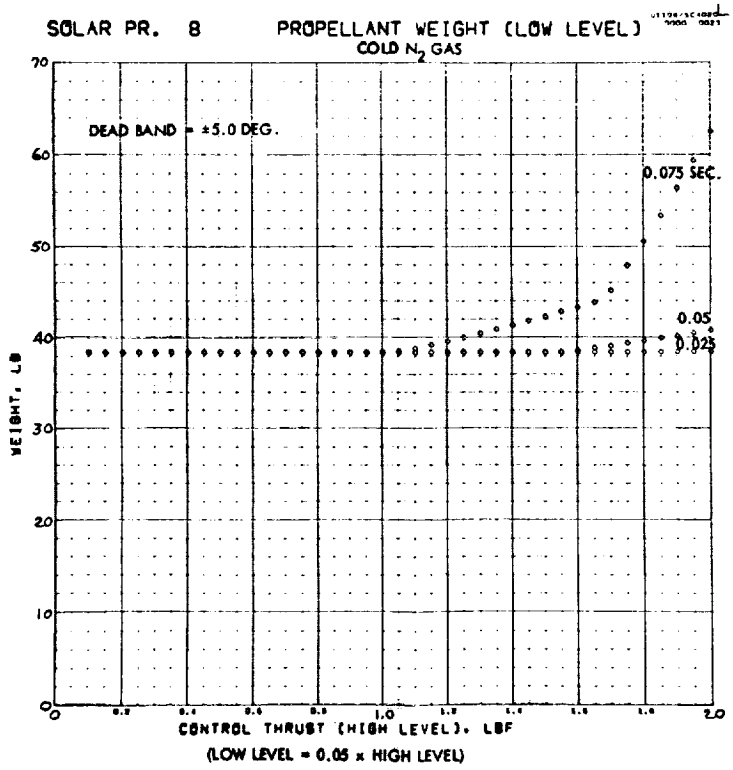

Fig. 55 Solar Probe ACS Requirements
Thruster Separation = 10 ft

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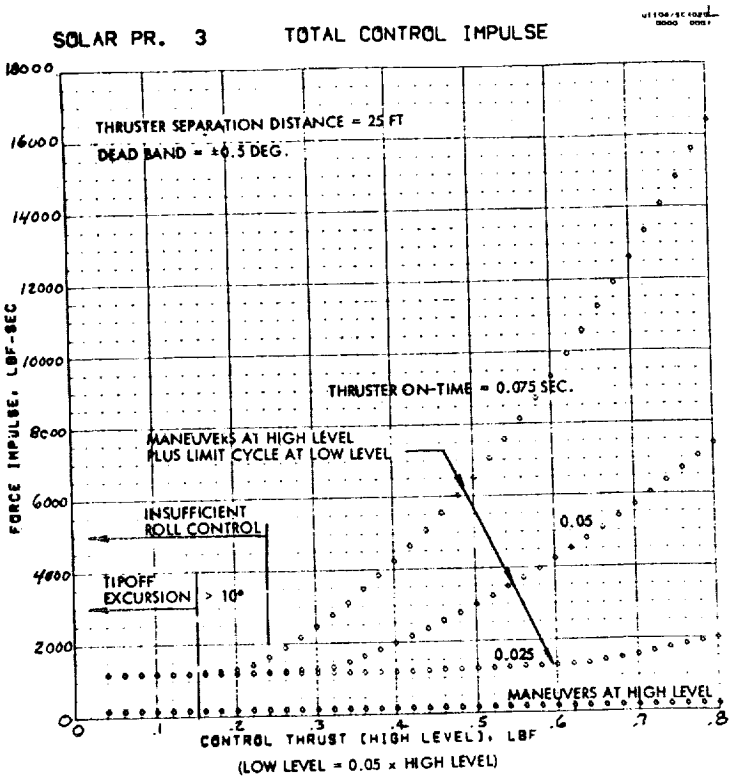

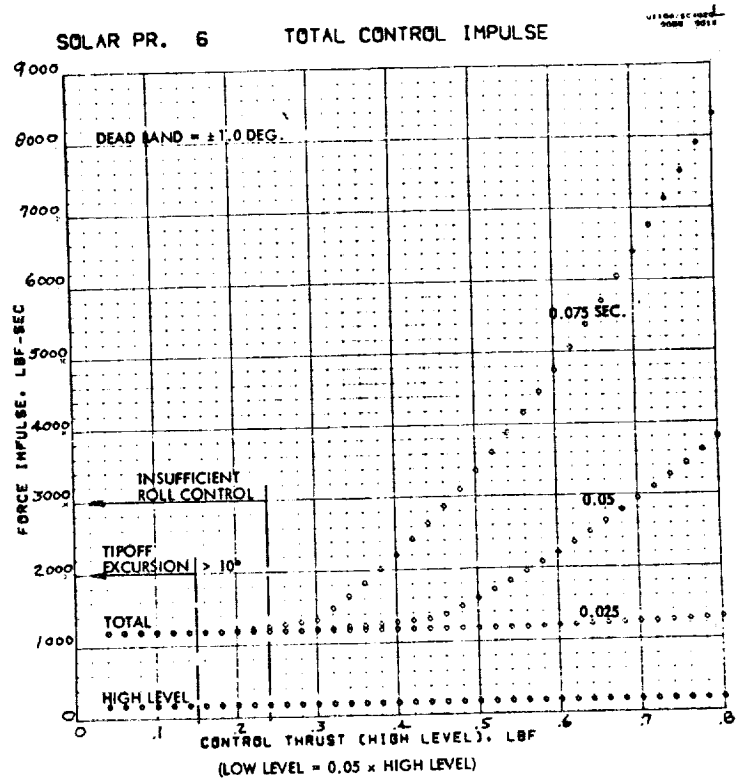

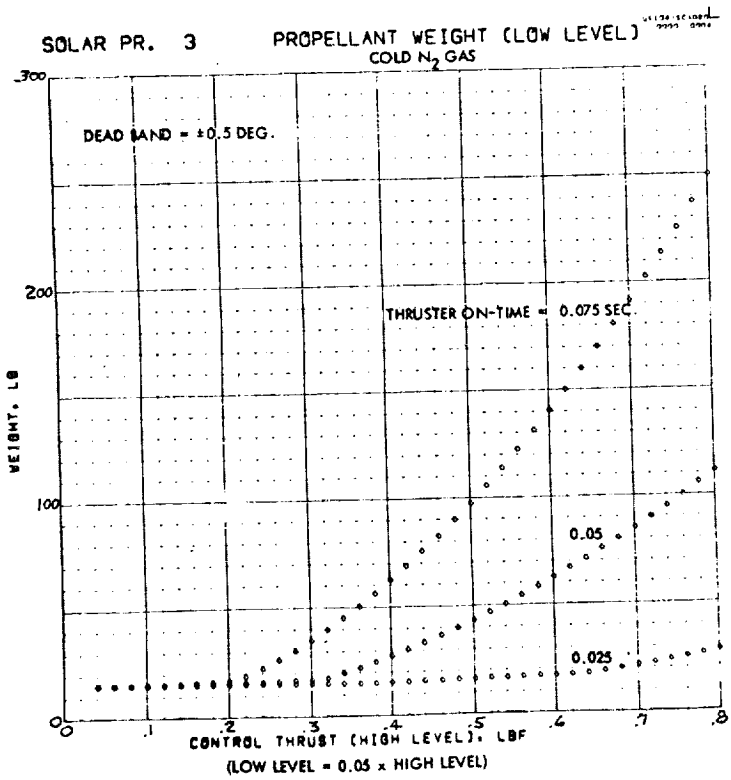

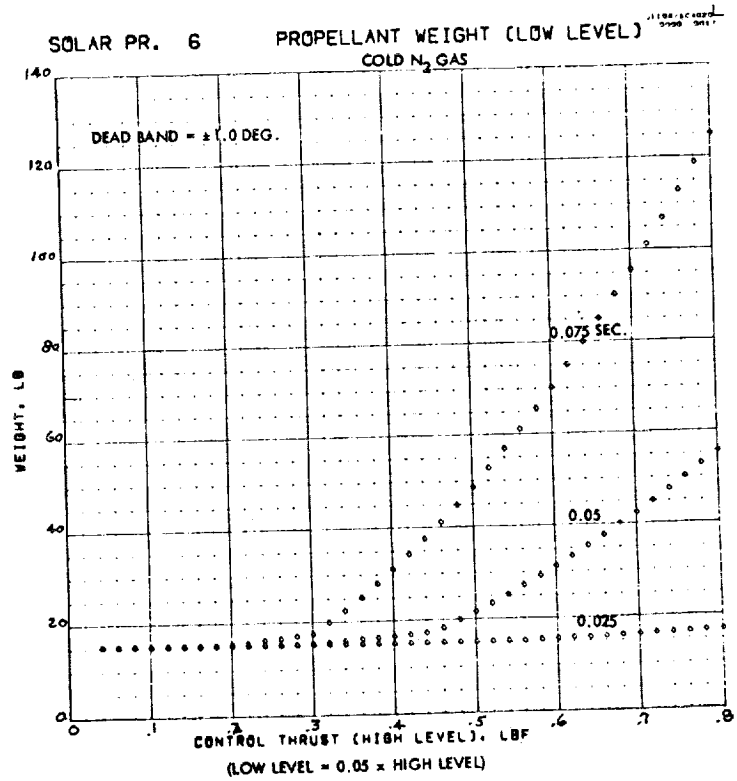

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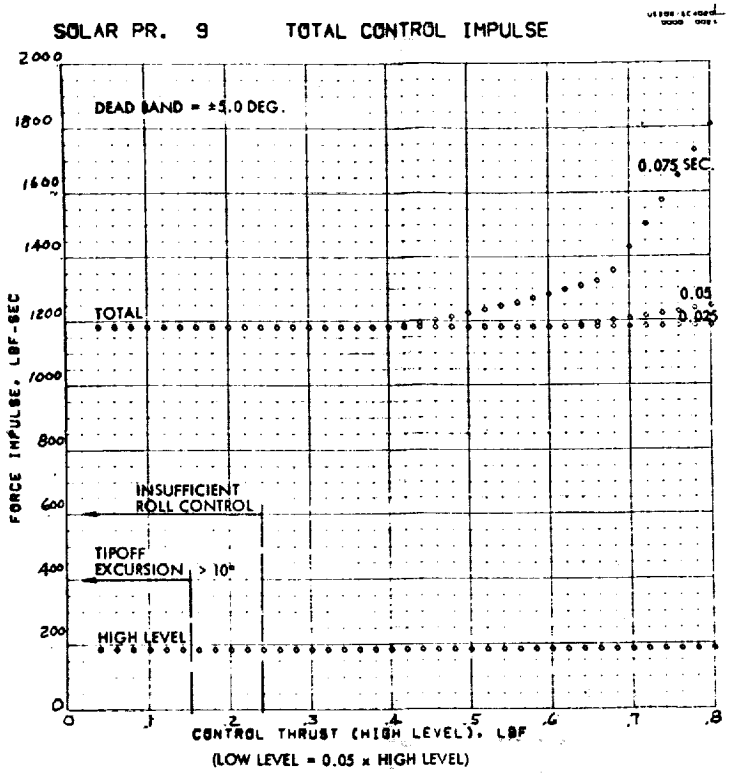
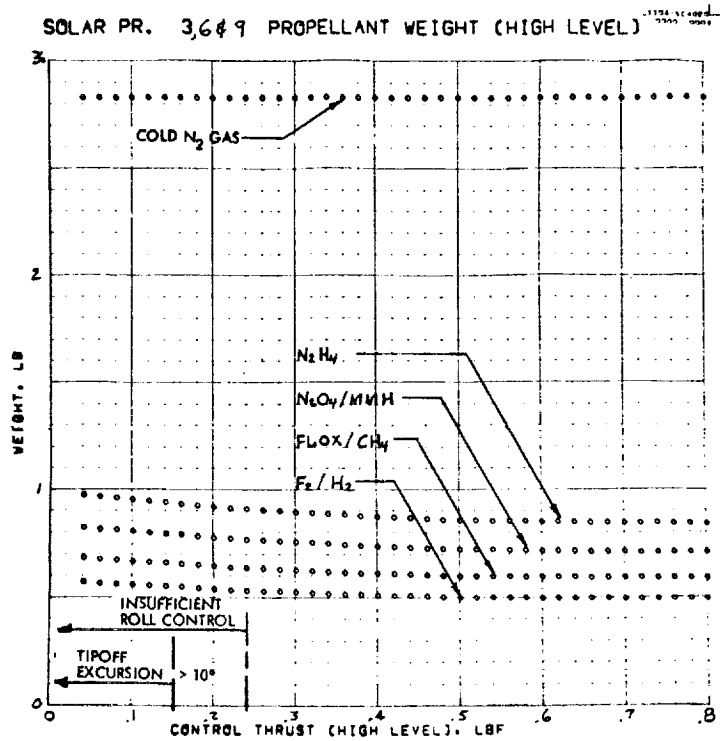




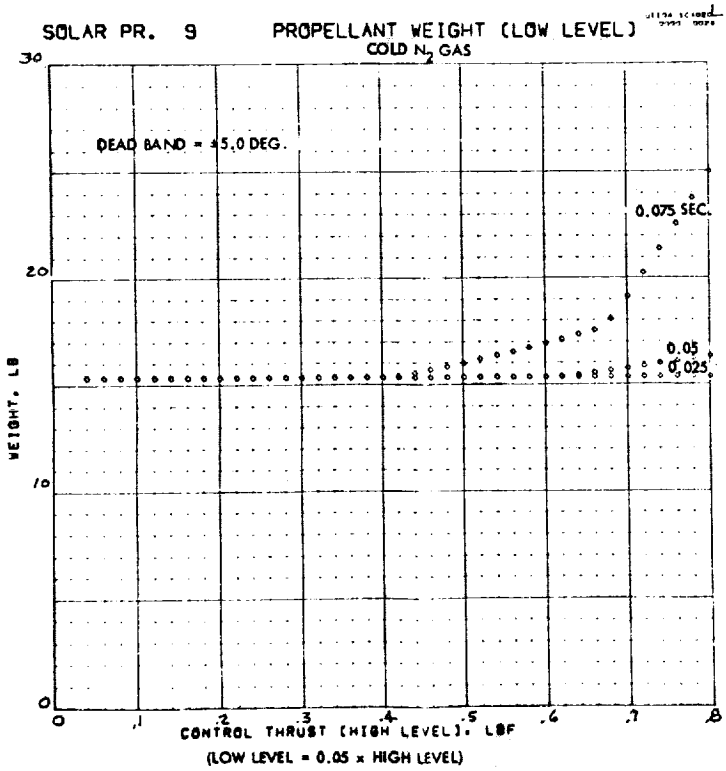


Fig. 56 Solar Probe ACS Requirements
Thruster Separation = 25 ft

FOLDOUT FRAME

Nine mission elements are used.

1. Type 1 with Configuration Jupiter Orbiter 1.
   Tipoff rates of 0.5, 1.5, and 1.5 deg/sec, respectively, are removed using a dynamic factor of 2.0.
2. Type 2 with Configuration Jupiter Orbiter 1.
   Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
3. Type 4 with dynamic factor = 2.
   Roll torque of 3.0 ft-lb for 304 sec, corresponding to the two 5000 lbf main engine burns.
4. Type 2 with Configuration Jupiter Orbiter 2.
   Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
5. Type 3 with Configuration Jupiter Orbiter 2.
   A 900 day transit cruise with unity deadband factors and solar torques of (0.2, 1.0, and 1.0) $\times 10^{-5}$ ft-lb about the X, Y and Z axes respectively.
6. Type 4 with dynamic factor = 2.
   Roll torque of 0.3 ft-lb for 23 sec, corresponding to the 500 lbf main engine burns.
7. Type 2 with Configuration Jupiter Orbiter 2.
   Four commanded roll-pitch-roll turns of 0.2 deg/sec with dynamic factor = 1.25, unwinds.
8. Type 2 with Configuration Jupiter Orbiter 3.
   Five acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
9. Type 3 with Configuration Jupiter Orbiter 3.
   A 180 day orbital cruise with unity deadband factors and solar/gravity gradient torques of (1, 5, and 5) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.

Nine computer runs were made using established parameter combinations, run numbering system, and display formats. It can be seen from Figs. 57, 58, and 59 that minimum total requirements are not significantly different than those for comparable Mars configuration. A larger proportion of the impulse requirements are derived from cruise mode operation, however.

Table 36 contains a summary of mission results.

The high level thrust constraint for yaw excursion subsequent to tipoff has been adjusted for moment of inertia.

Table 36

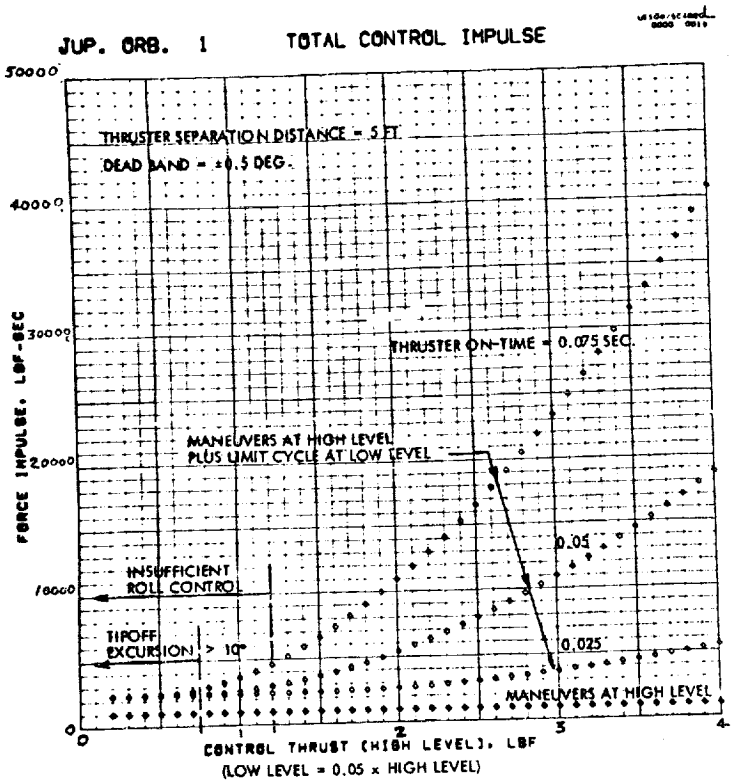
JUPITER ORBITER – CONTROL IMPULSE SUMMARY

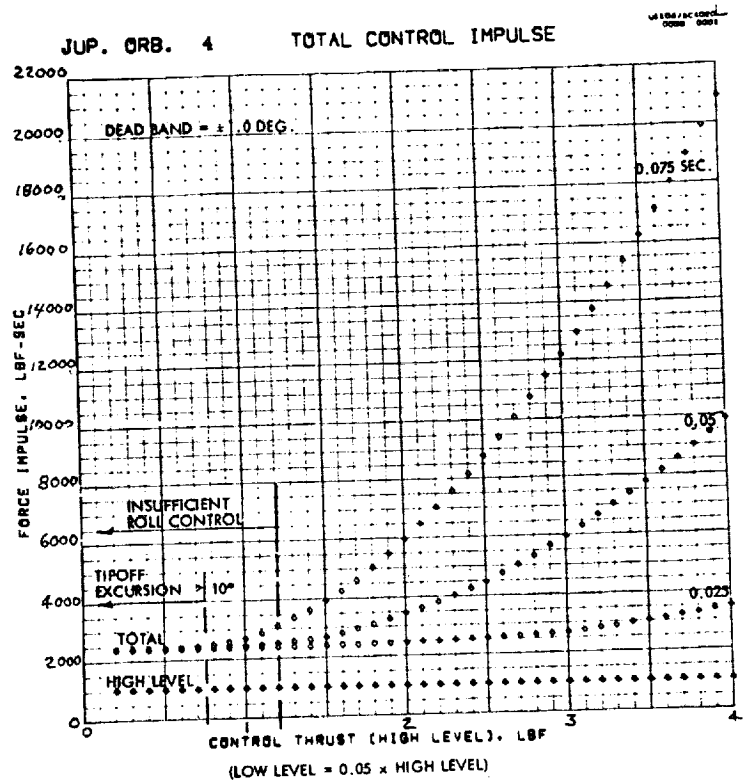
Thruster Separation L (ft)	5	10	25
High Level Impulse (lb-sec)	1015 ($F \geq 1.2$ lbf)	508 ($F \geq 0.6$ lbf)	203 ($F \geq 0.24$ lbf)
Min. Low Level Impulse (lb-sec)	1406 ($F < 0.085$ lbf)	703 ($F < 0.04$ lbf)	281 ($F < 0.017$ lbf)
Min. Total Impulse (lb-sec)	2421	1211	484

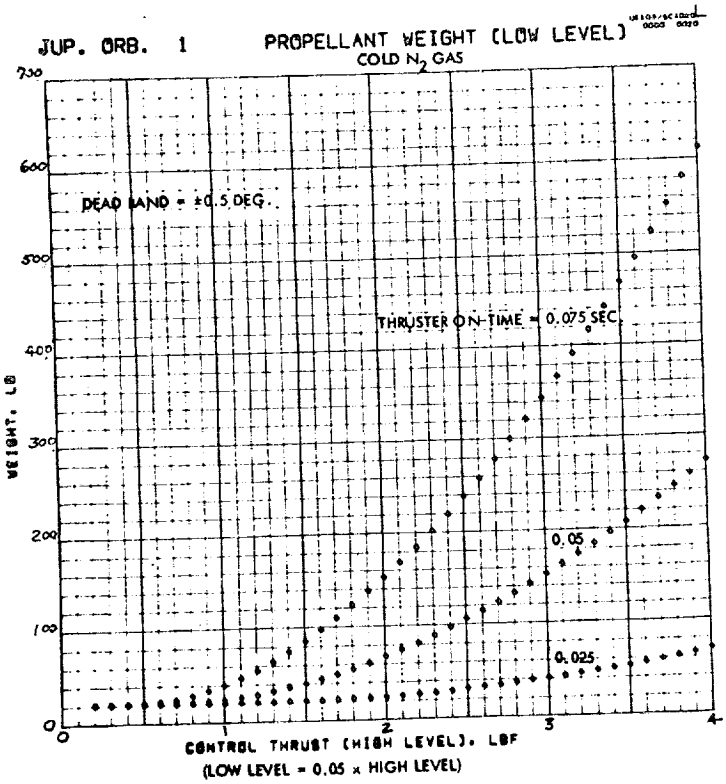
Note: Deadband = ±0.5 deg.

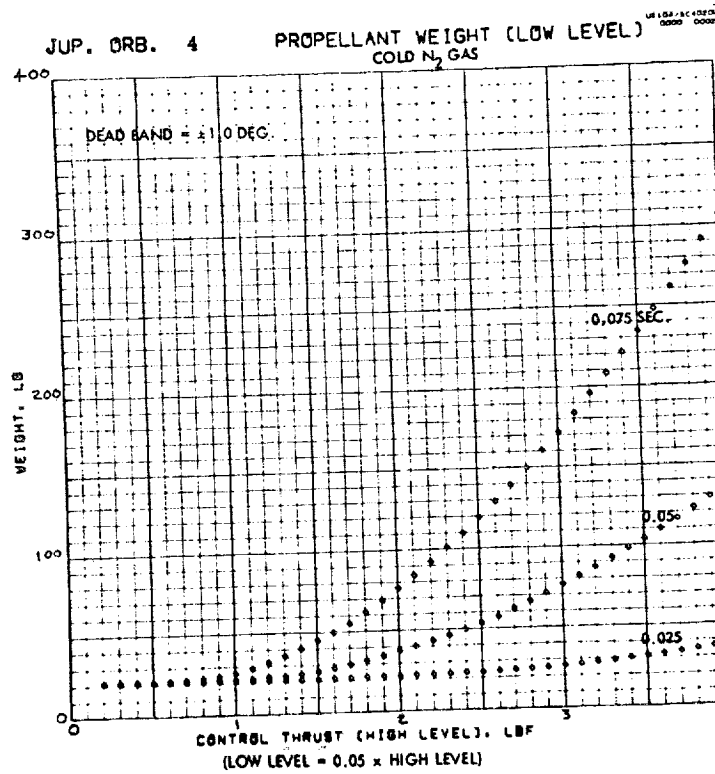
3.6.8 Jupiter Flyby Mission

Two spacecraft configurations are of interest to the Jupiter Flyby mission as noted in Section 3.3. The moments of inertia for this configuration are shown below.









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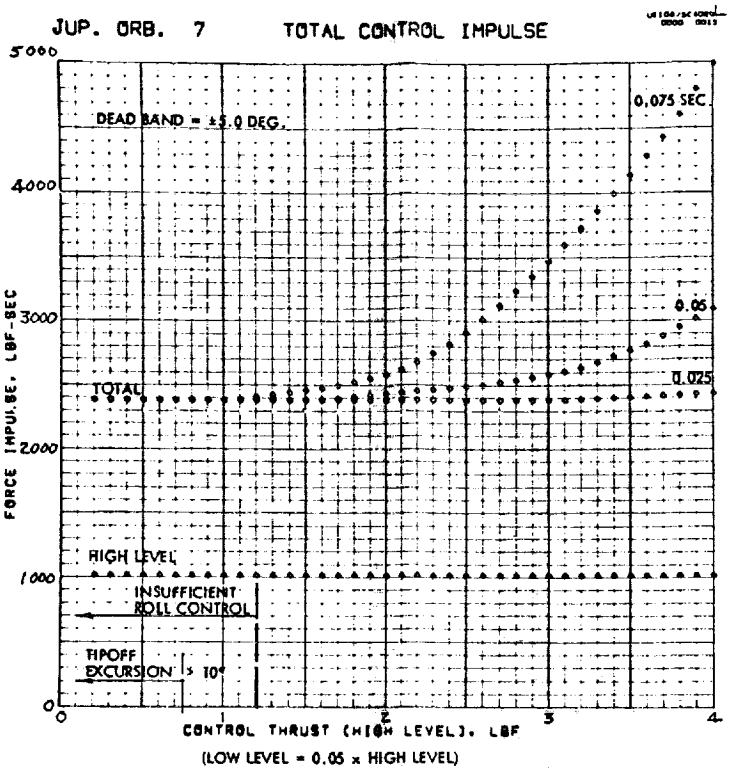
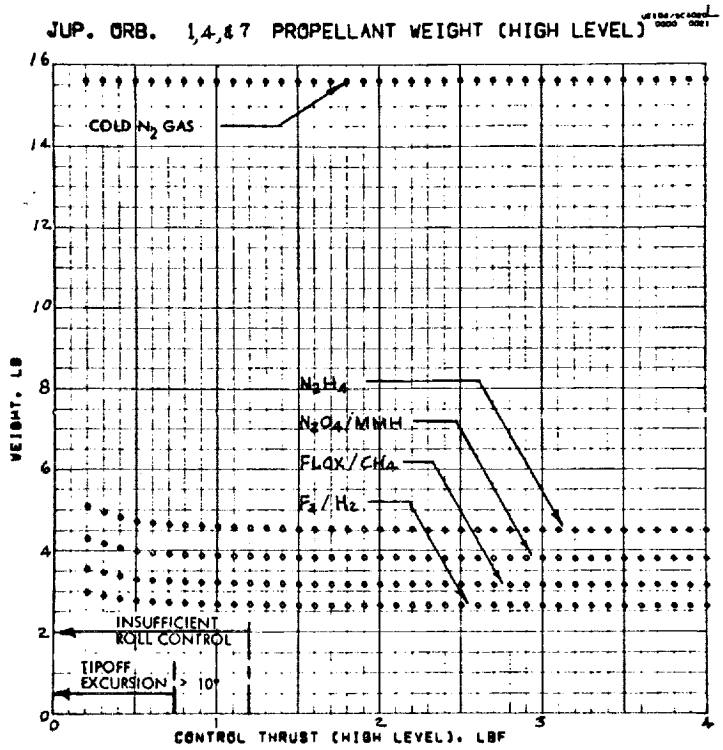




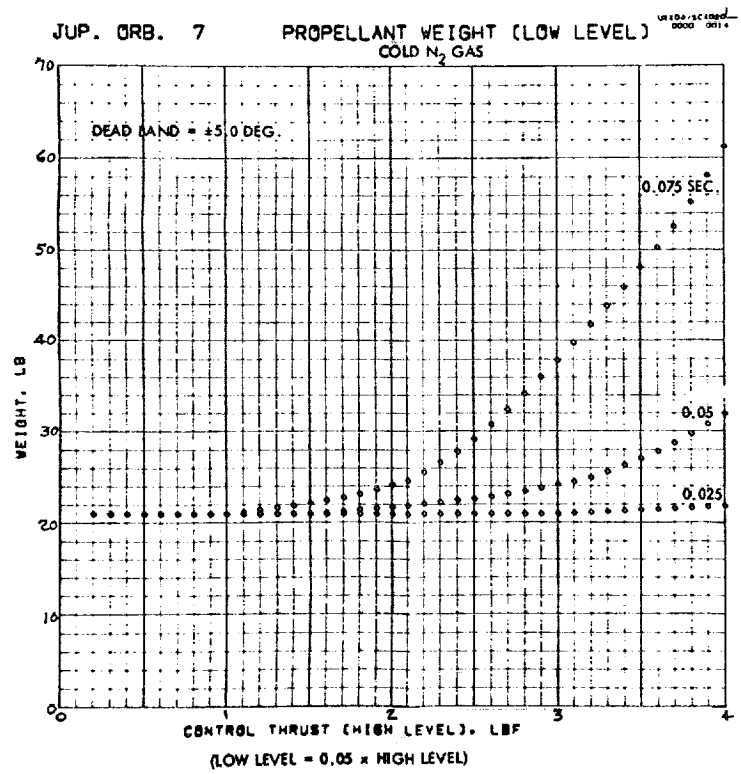
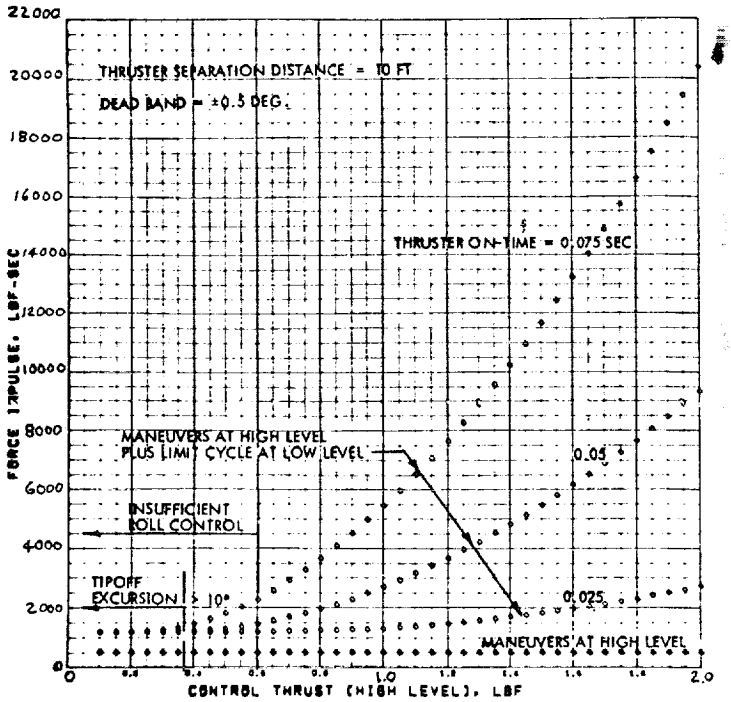


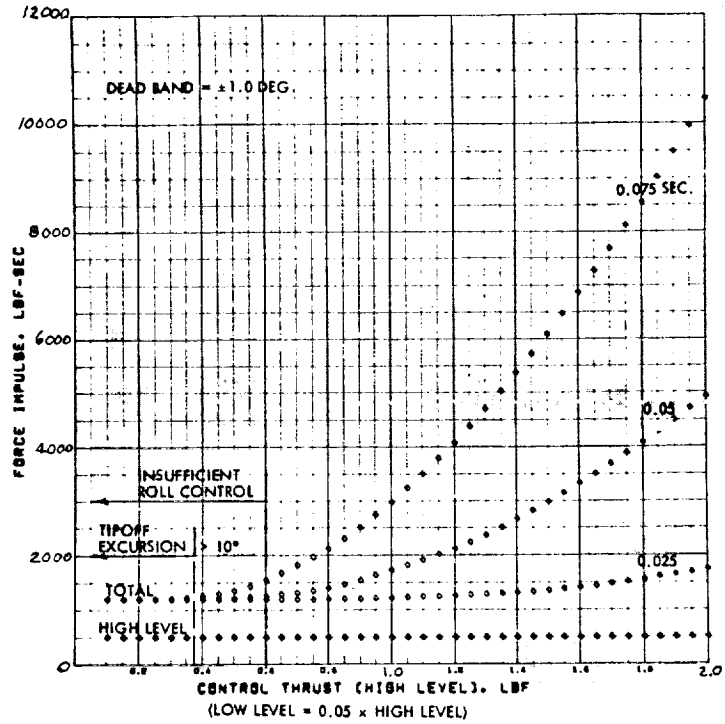
Fig. 57 Jupiter Orbiter ACS Requirements
Thruster Separation = 5 ft

FOLDOUT FRAME 2

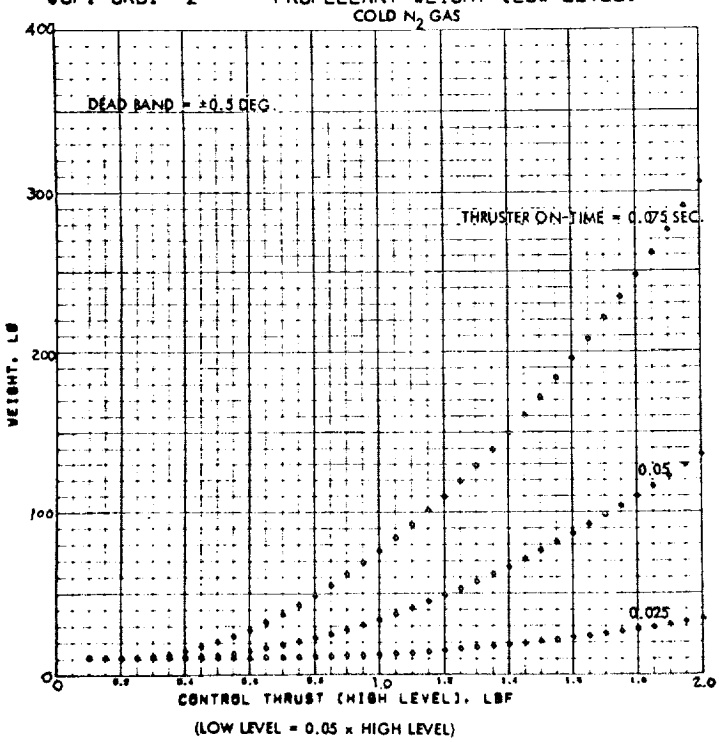
JUP. ORB. 2 TOTAL CONTROL IMPULSE



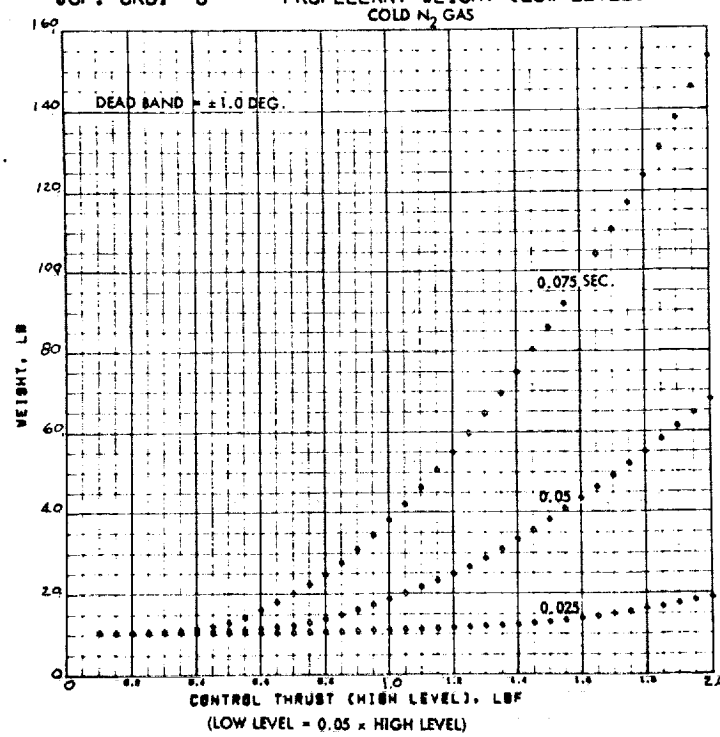
JUP. ORB. 5 TOTAL CONTROL IMPULSE



JUP. ORB. 2 PROPELLANT WEIGHT (LOW LEVEL)
COLD N_2 GAS



JUP. ORB. 5 PROPELLANT WEIGHT (LOW LEVEL)
COLD N_2 GAS



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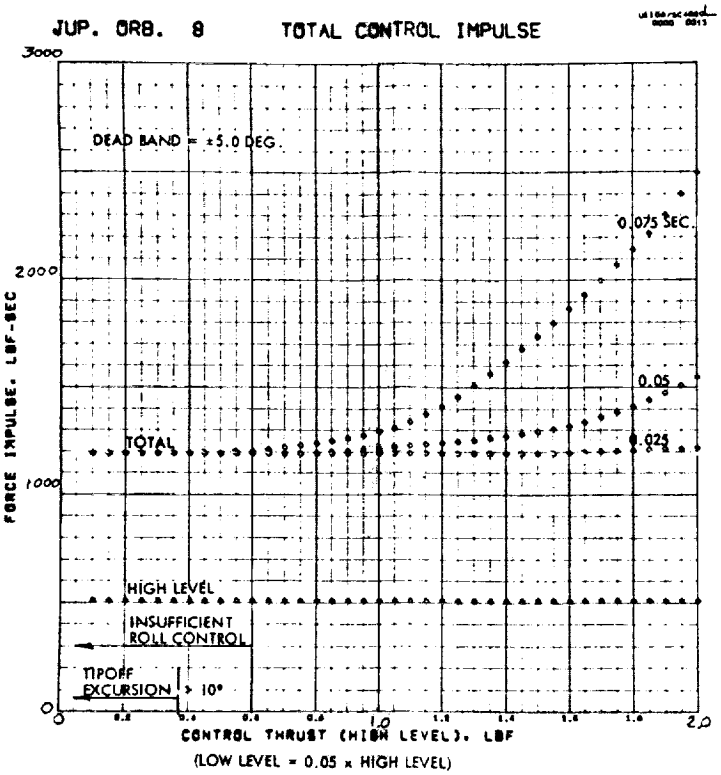
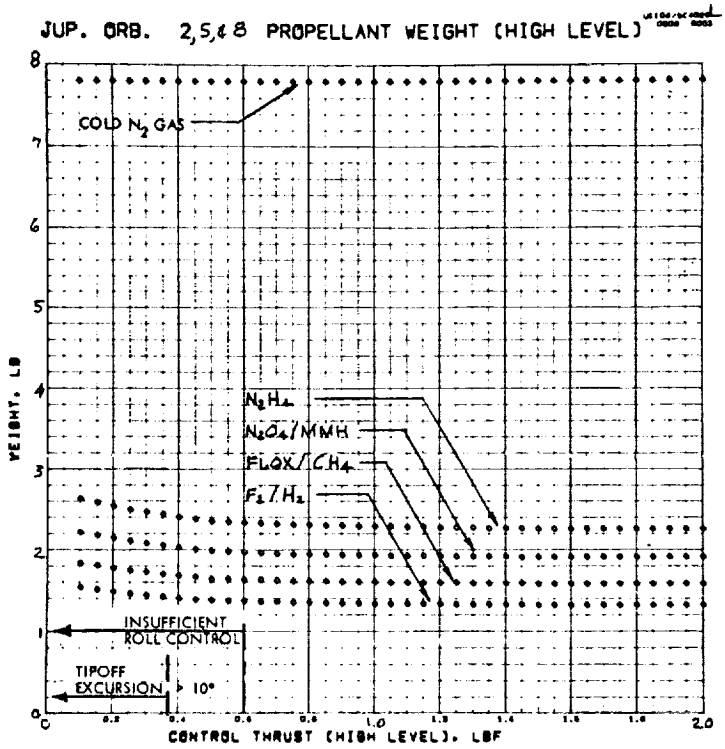




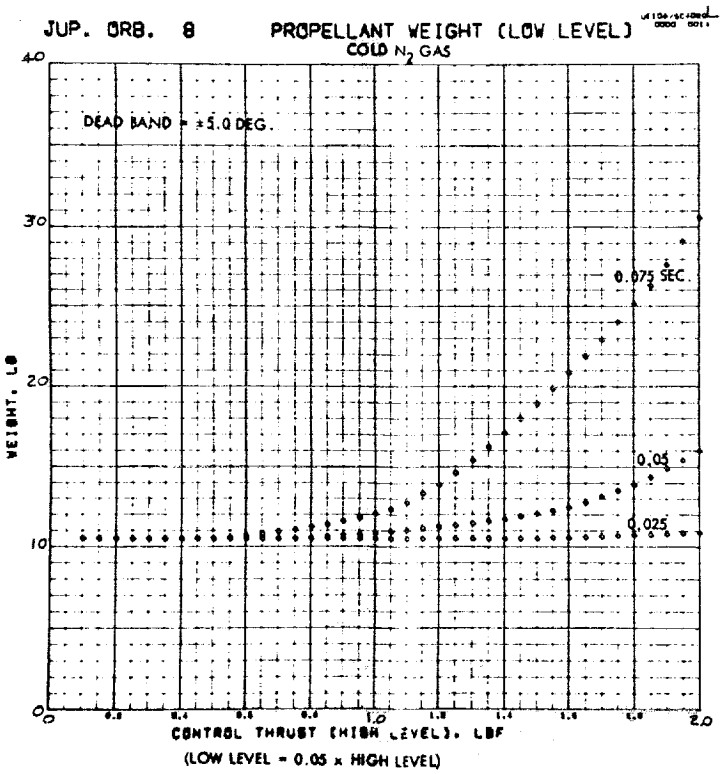
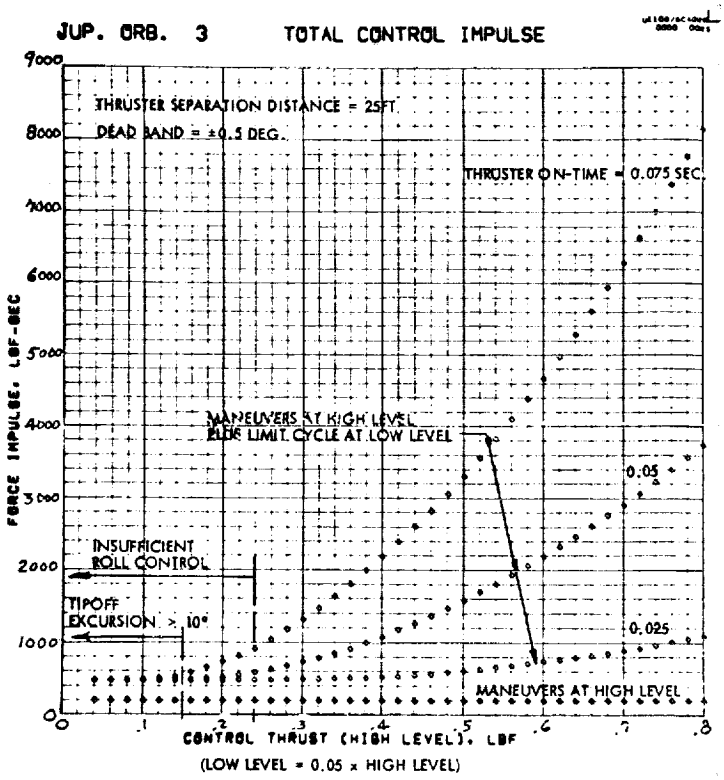
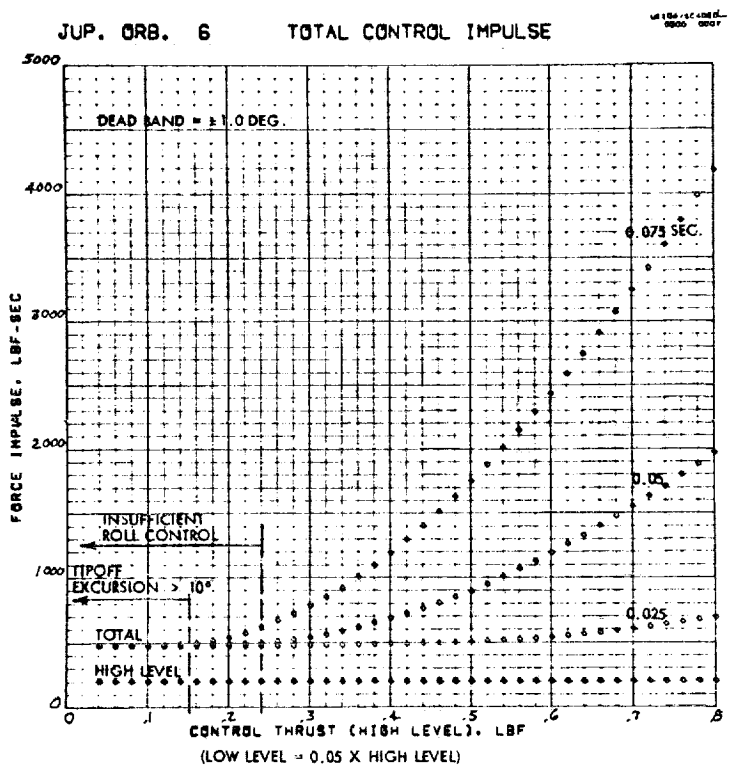
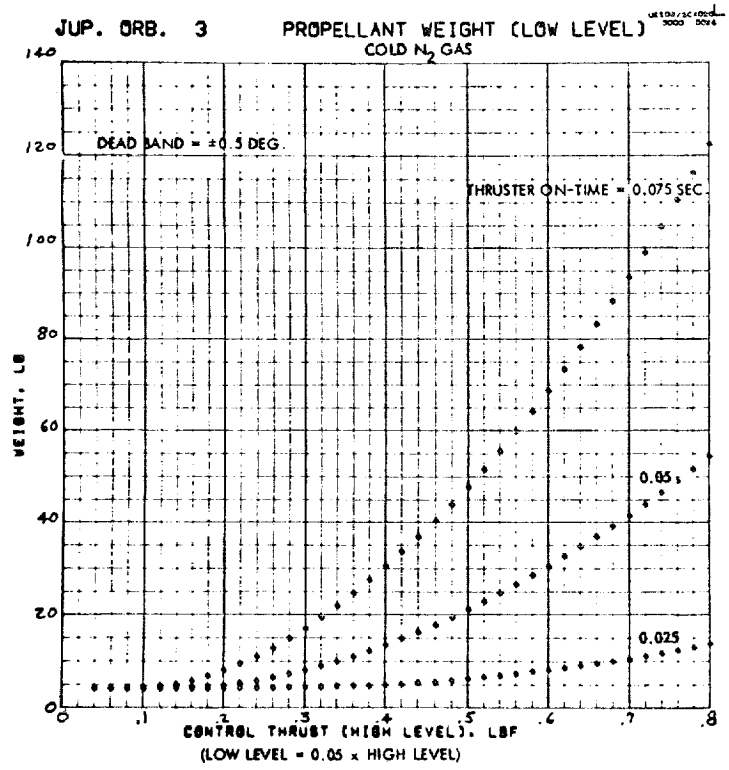


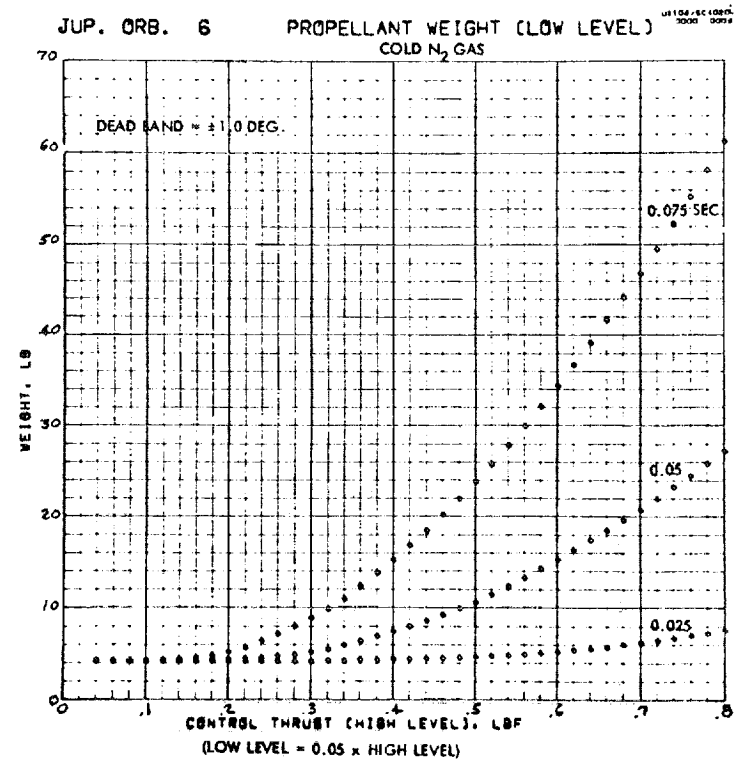
Fig. 58 Jupiter Orbiter ACS Requirements
Thruster Separation = 10 ft

FOLDOUT FRAME V









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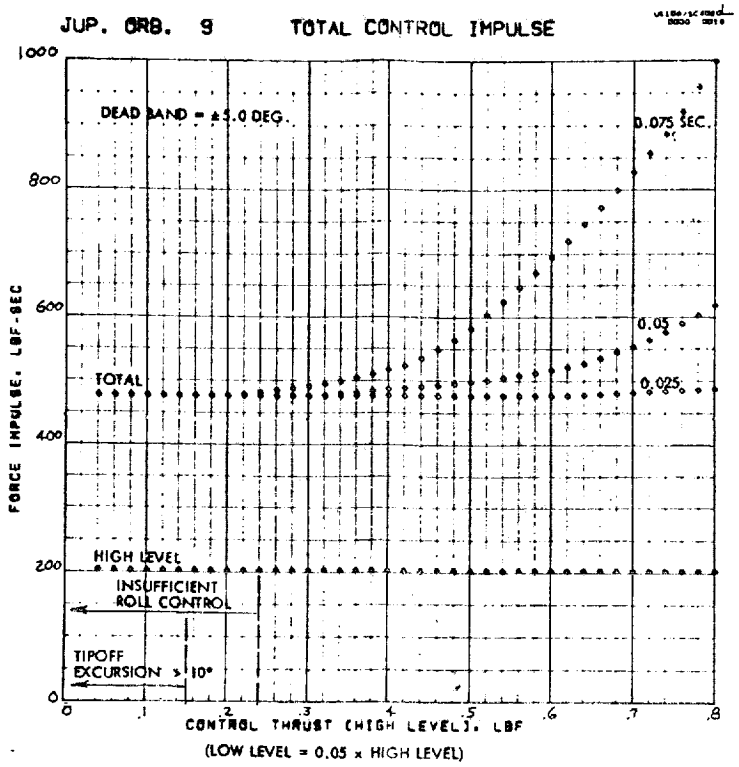
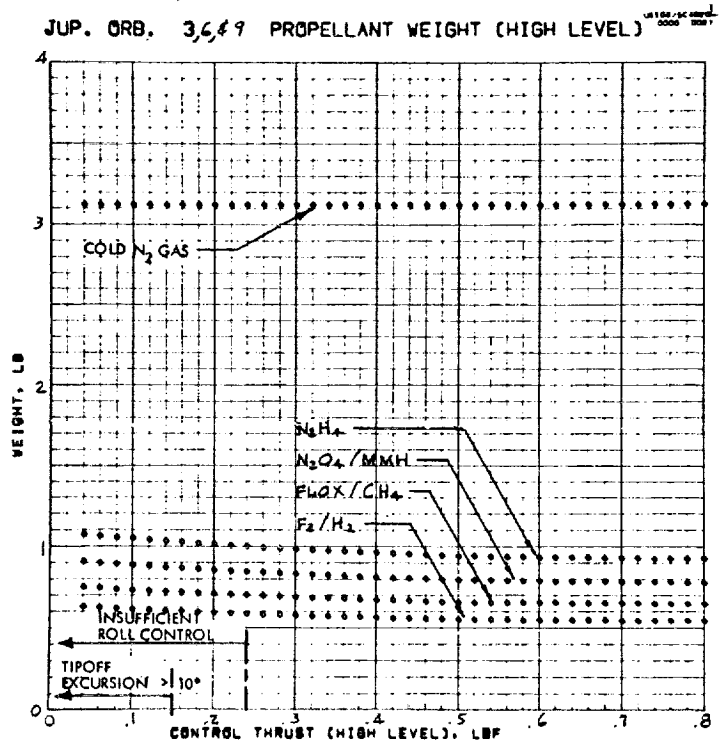




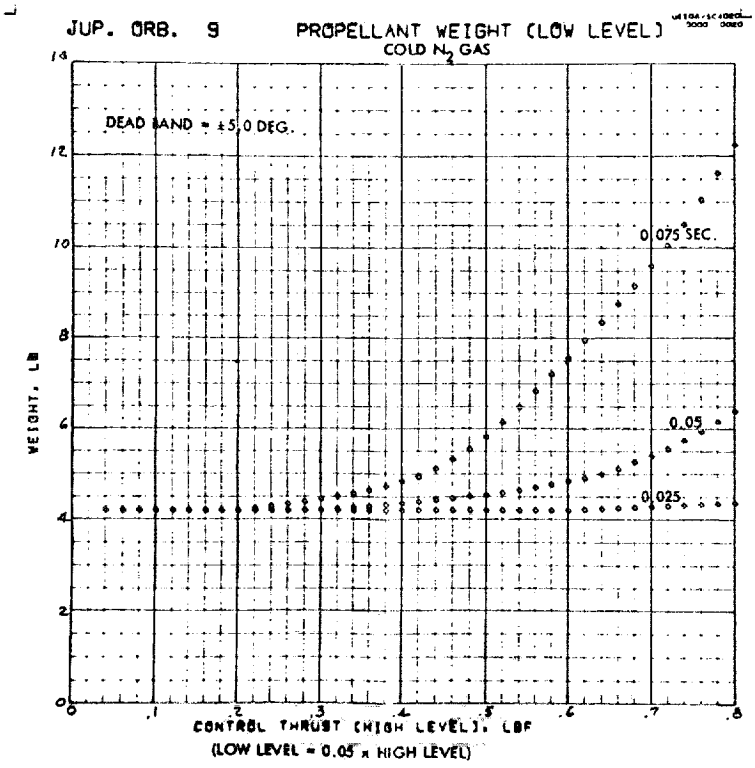


Fig. 59 Jupiter Orbiter ACS Requirements
Thruster Separation = 25 ft

FOLDOUT FRAME

Moments of Inertia	Configuration	
	Jupiter F.B. 1	Jupiter F.B. 2
I_x slug-ft^2	2685	1264
I_y slug-ft^2	2102	677
I_z slug-ft^2	2798	677

Seven mission elements are employed.

1. Type 1 with Configuration Jupiter Flyby 1.
   Tipoff rates of 0.5, 1.5, and 1.5 deg/sec, respectively, are removed using a dynamic factor of 2.0.
2. Type 2 with Configuration Jupiter Flyby 1.
   Two acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
3. Type 4 with dynamic factor = 2.
   Roll torque of 3.0 ft-lb for 322 sec, corresponding to the 5000 lbf main engine burn.
4. Type 2 with Configuration Jupiter Flyby 2.
   Six acquisition search maneuvers (0.2 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5,
5. Type 2 with Configuration Jupiter Flyby 2
   Four commanded roll-pitch-roll turns of 0.2 deg/sec with dynamic factor = 1.25, unwinds.
6. Type 4 with dynamic factor = 2.0.
   Roll torque of 0.06 ft-lb for 113 sec, corresponding to duration of 100 lbf main engine burns.
7. Type 3 with Configuration Jupiter Flyby 2.
   A 900 day transit cruise with unity deadband factors and solar torques of (0.2, 1.0, and 1.0) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.

Plotted results in Figs. 60 , 61 , and 62 and tabulated results in Table 37 show that the high level requirements for the Jupiter orbiter and flyby missions are not appreciably different but the minimum low level requirements of the flyby mission are half those of the orbiter. This is to be expected because of the shorter cruise duration. For higher thrust levels, however, the low level flyby requirements exceed those of the orbiter. This can be explained by the moment of inertia values. The flyby mission has lower values during cruise (nearly empty propellant tanks) and hence higher limit cycle frequency. Recall that minimum low level requirements are not dependent on moment of inertia.

Table 37

JUPITER FLYBY – CONTROL IMPULSE SUMMARY

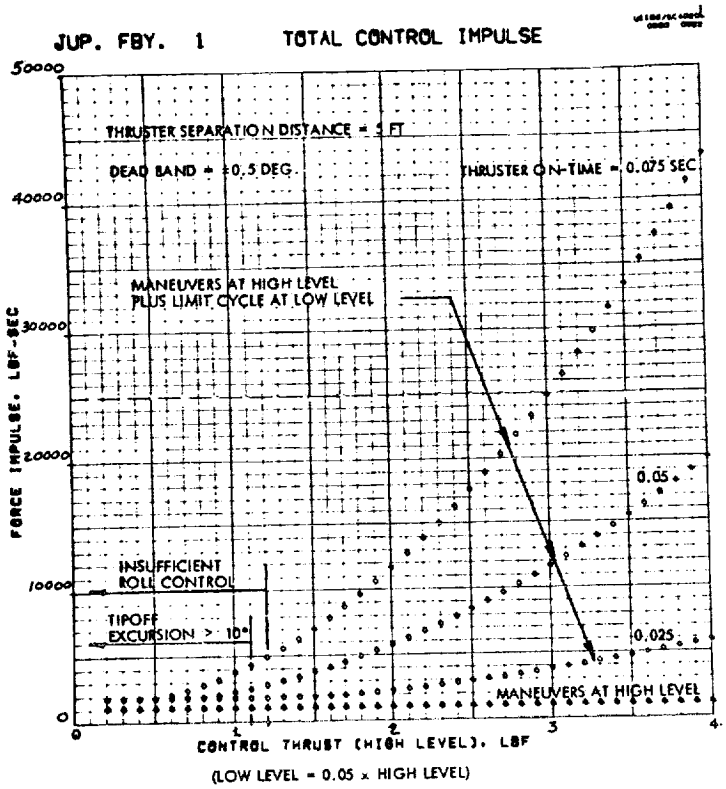
Thruster Separation L (ft)	5	10	25
High Level Impulse (lb-sec)	1129 ($F \geq 1.2$ lbf)	564 ($F \geq 0.6$ lbf)	226 ($F \geq 0.24$ lbf)
Min. Low Level Impulse (lb-sec)	711 ($F < 0.08$ lbf)	356 ($F < 0.042$ lbf)	142 ($F < 0.017$ lbf)
Min. Total Impulse (lb-sec)	1840	920	368

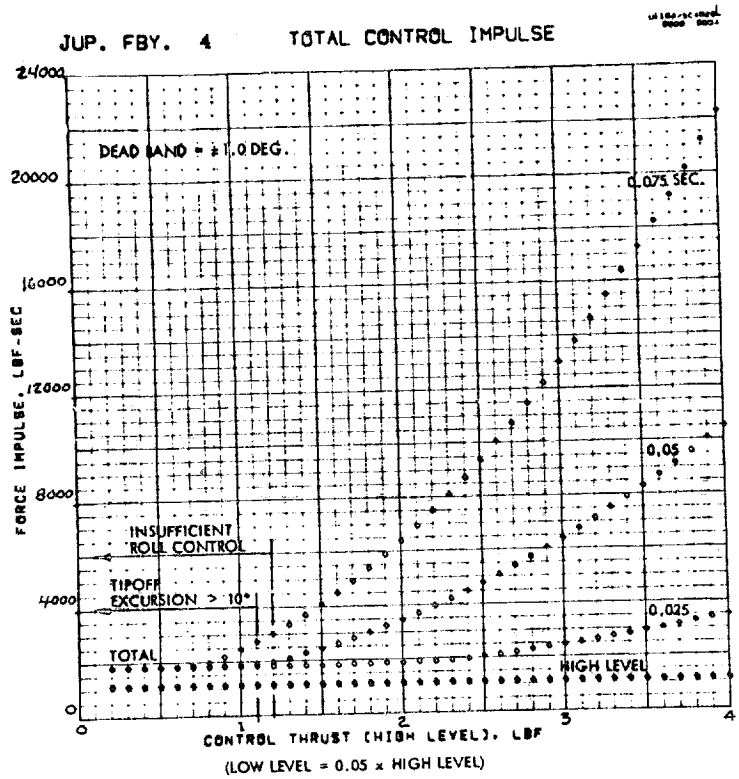
Note: Deadband = ± 0.5 deg.

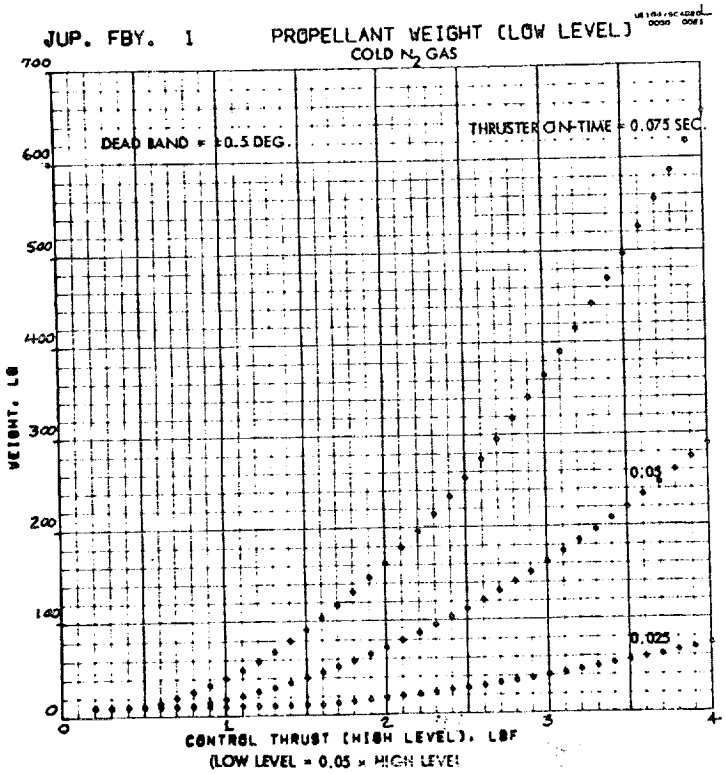
3.6.9 Intermediate Size Lunar Lander Mission

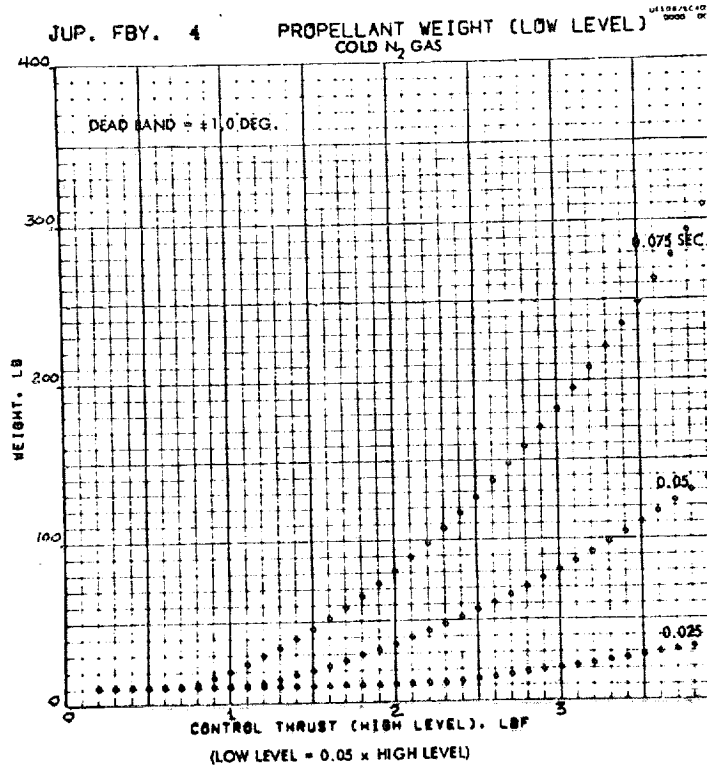
Two spacecraft configurations are of interest for the Lunar Lander Mission per Section 3.3 discussion. The inertias for these configurations are shown below:

Moments of Inertia	Configuration	
	Lunar 1	Lunar 2
I_x slug-ft^2	3890	2624
I_y slug-ft^2	3980	3109
I_z slug-ft^2	4119	3248









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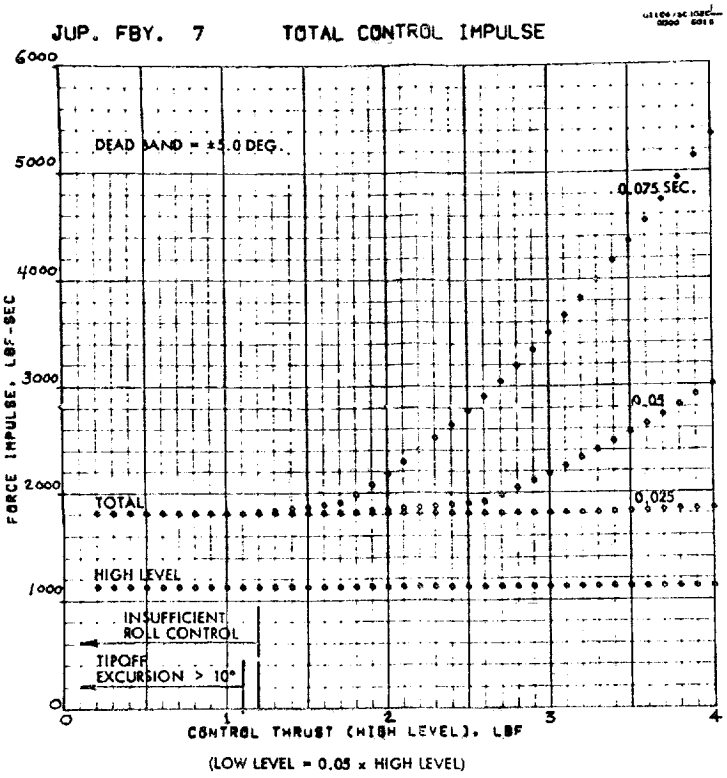
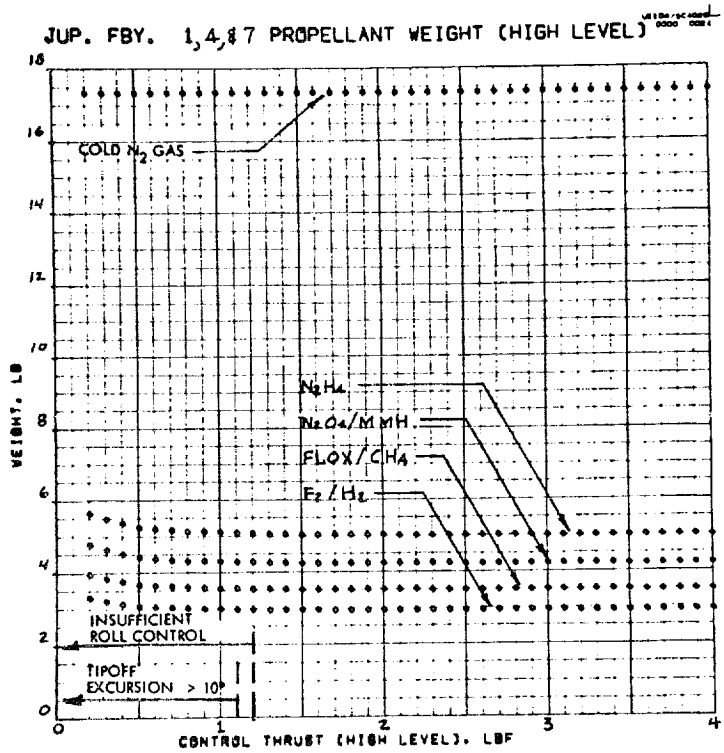




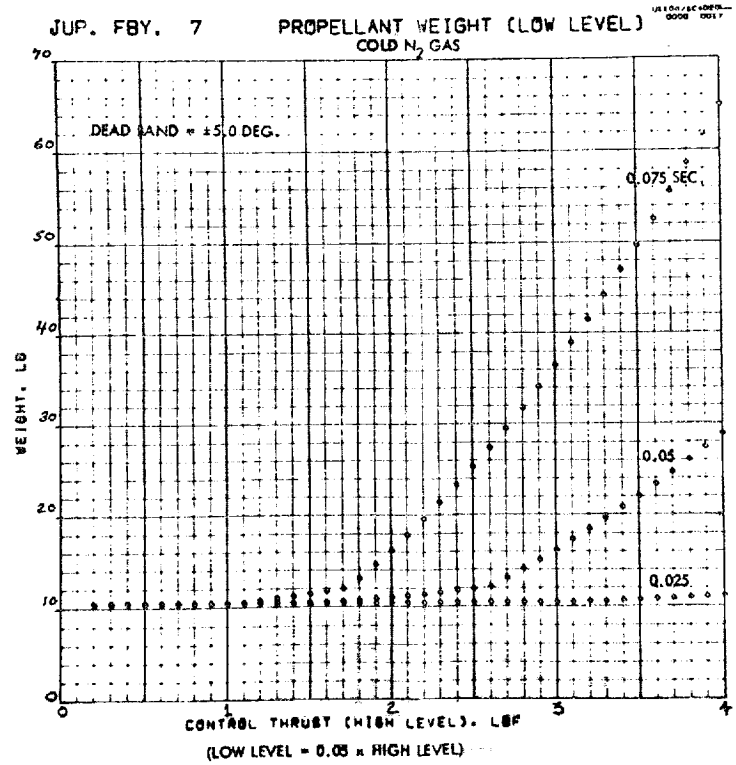
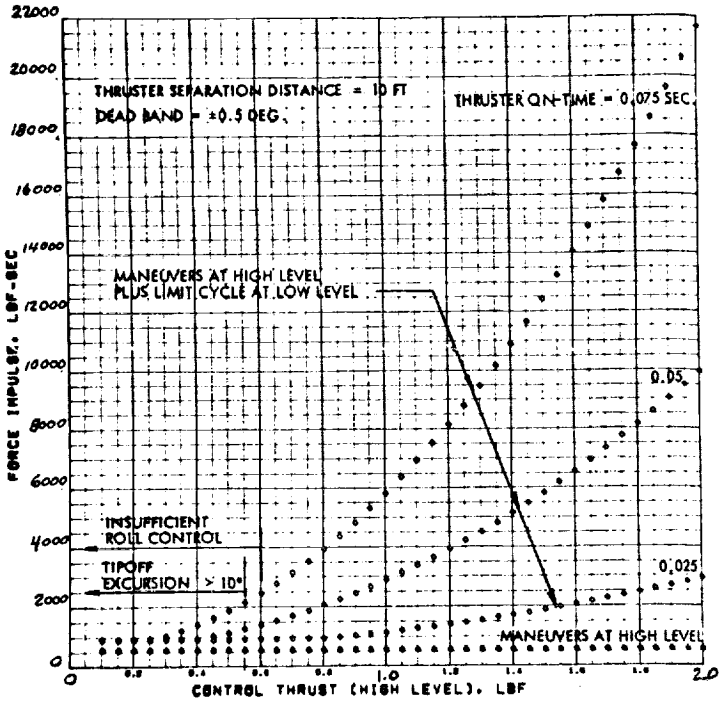


Fig. 60 Jupiter Flyby ACS Requirements
Thruster Separation = 5 ft

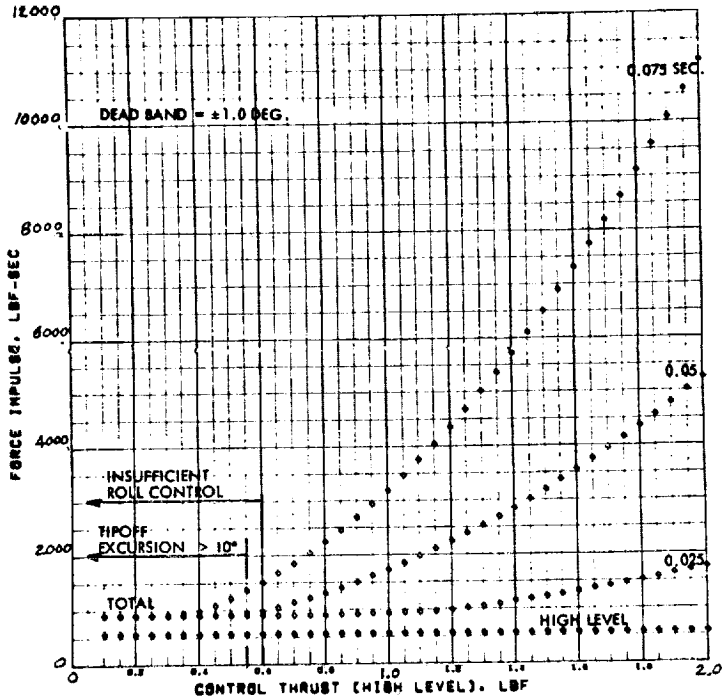
FOLDOUT FRAME

JUP. FBY. 2 TOTAL CONTROL IMPULSE



(LOW LEVEL = 0.05 x HIGH LEVEL)

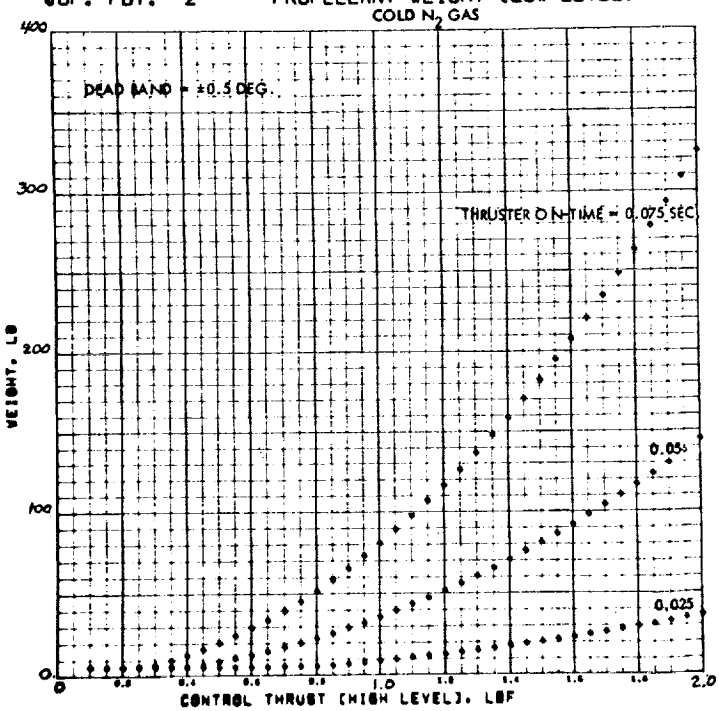
JUP. FBY. 5 TOTAL CONTROL IMPULSE



(LOW LEVEL = 0.05 x HIGH LEVEL)

JUP. FBY. 2 PROPELLANT WEIGHT (LOW LEVEL)

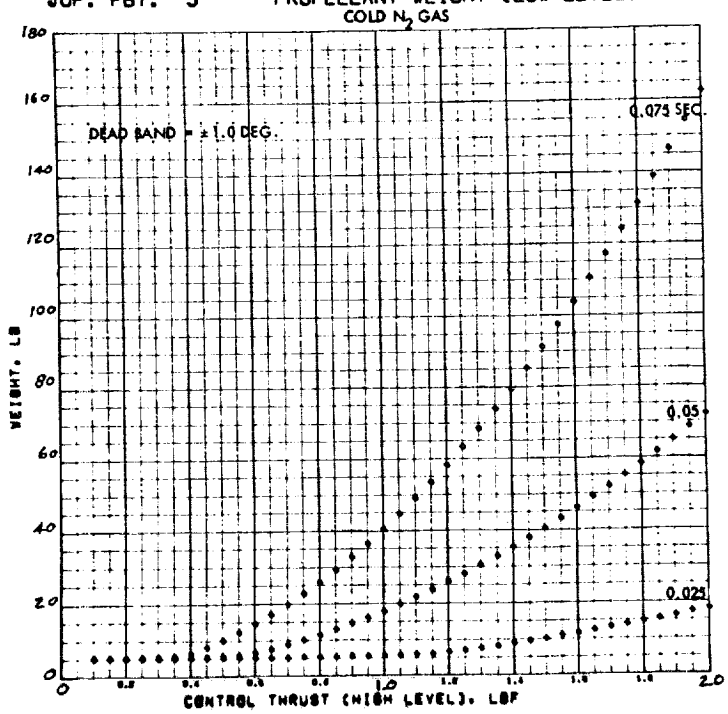
COLD N_2 GAS



(LOW LEVEL = 0.05 x HIGH LEVEL)

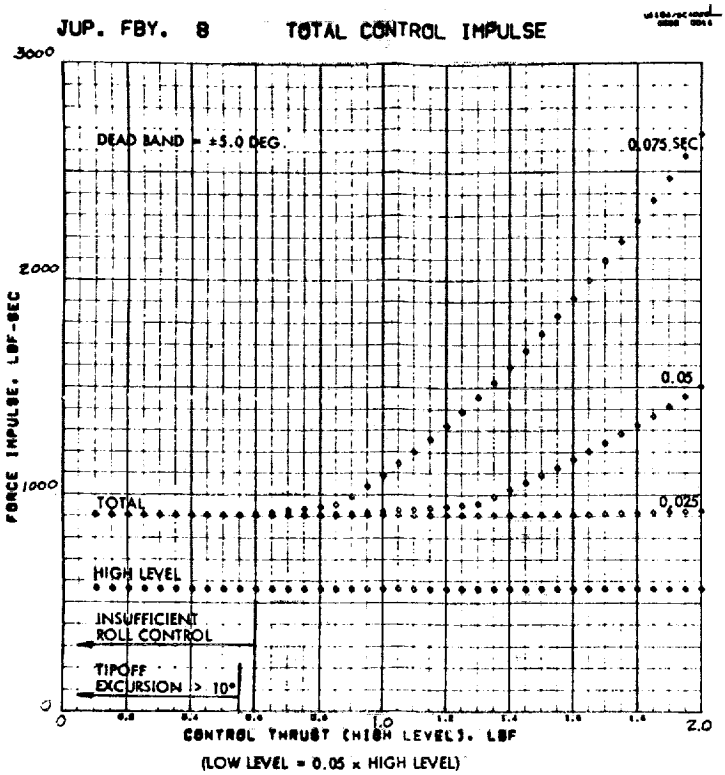
JUP. FBY. 5 PROPELLANT WEIGHT (LOW LEVEL)

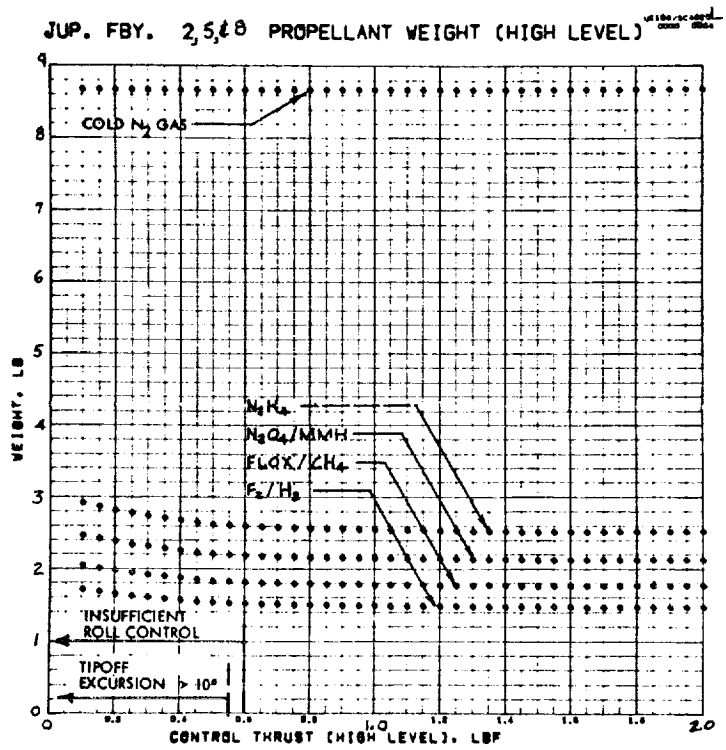
COLD N_2 GAS



(LOW LEVEL = 0.05 x HIGH LEVEL)

FOLDOUT FRAME





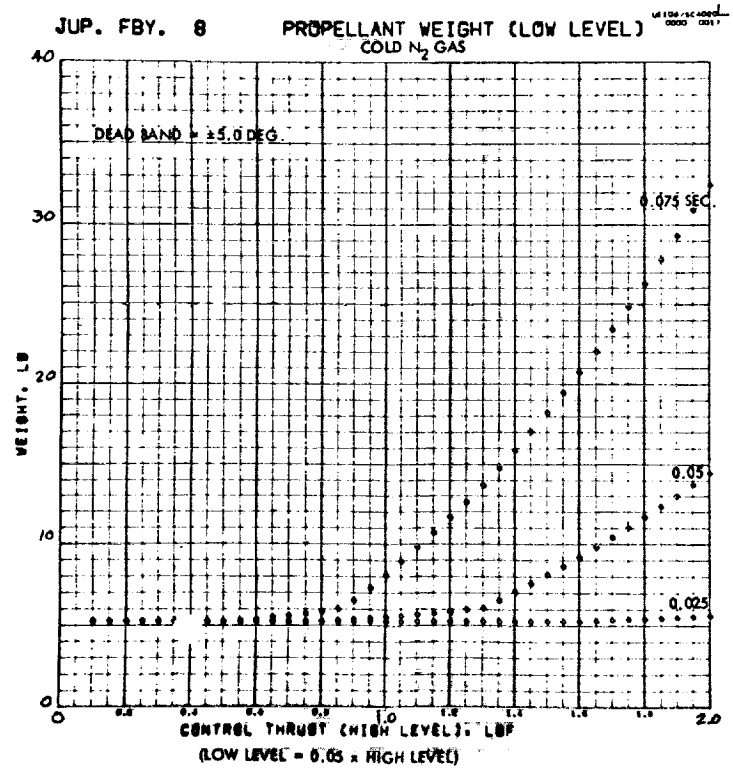


Fig. 61 Jupiter Flyby ACS Requirements
Thruster Separation = 10 ft

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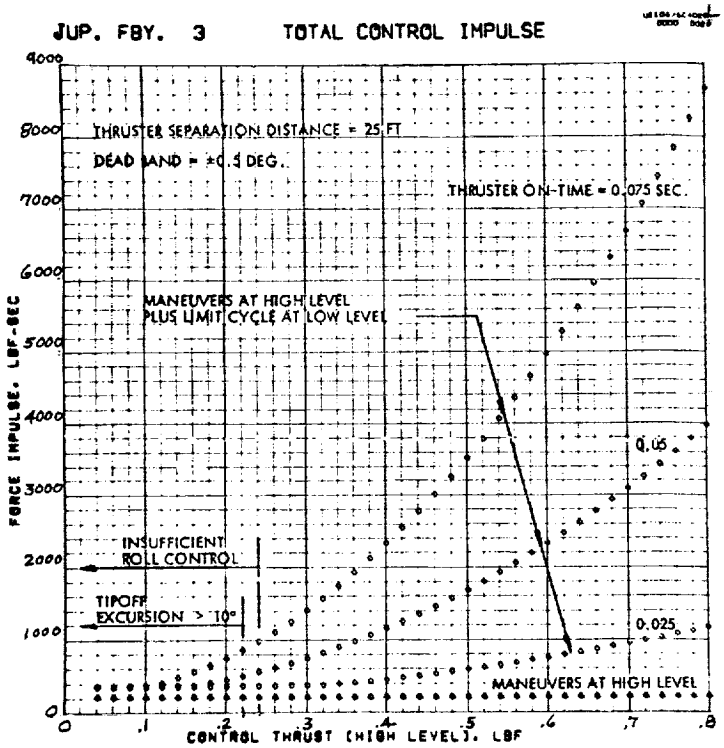

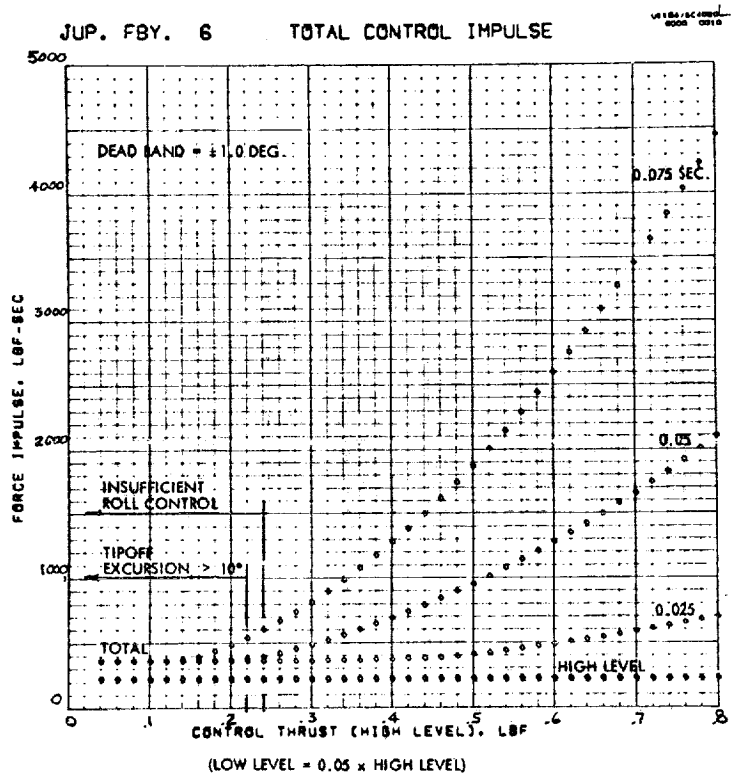

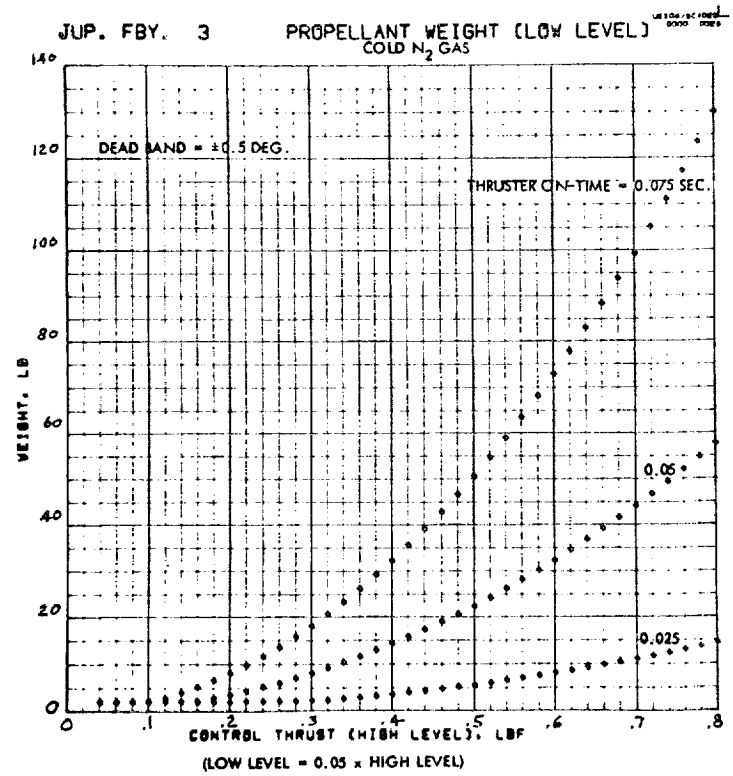

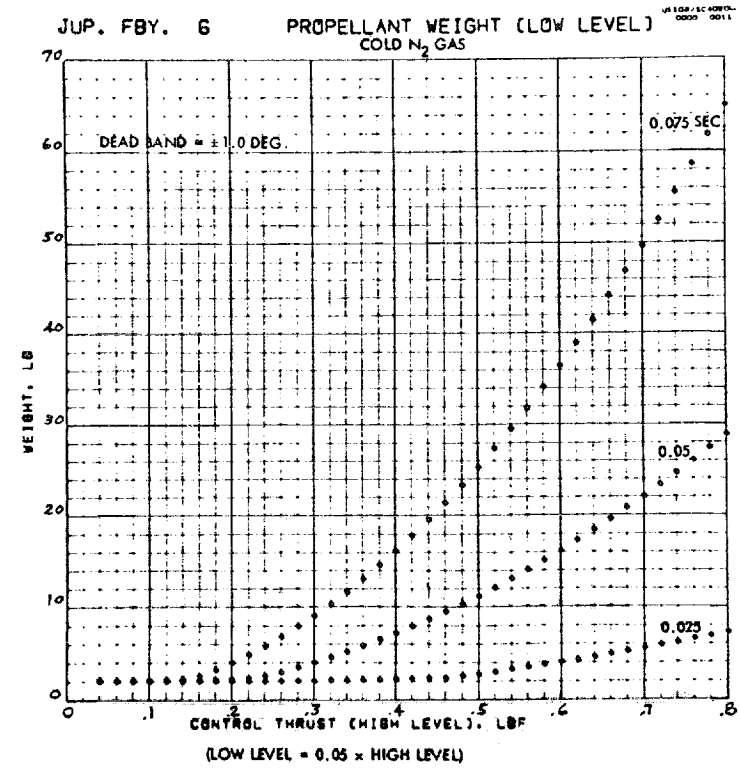

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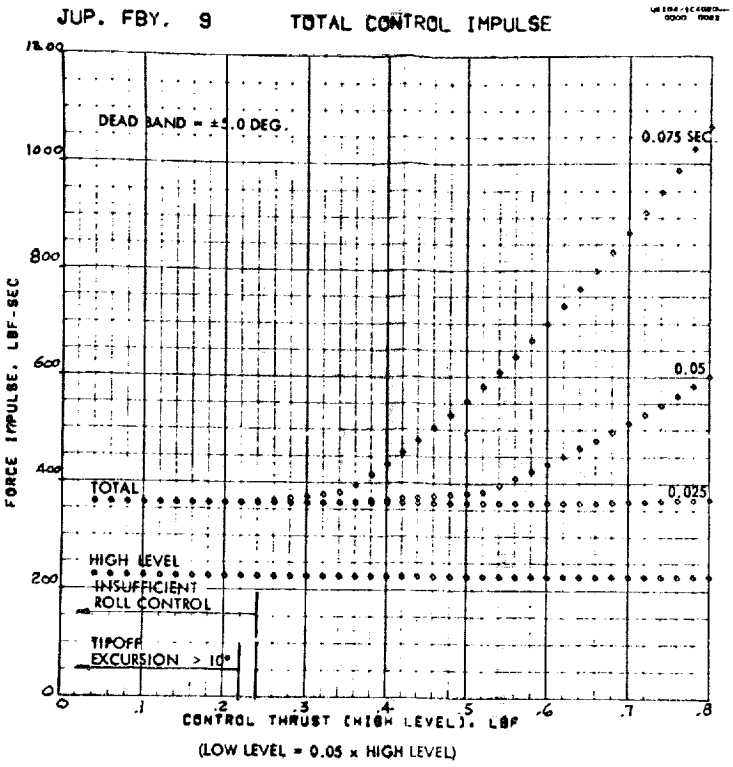

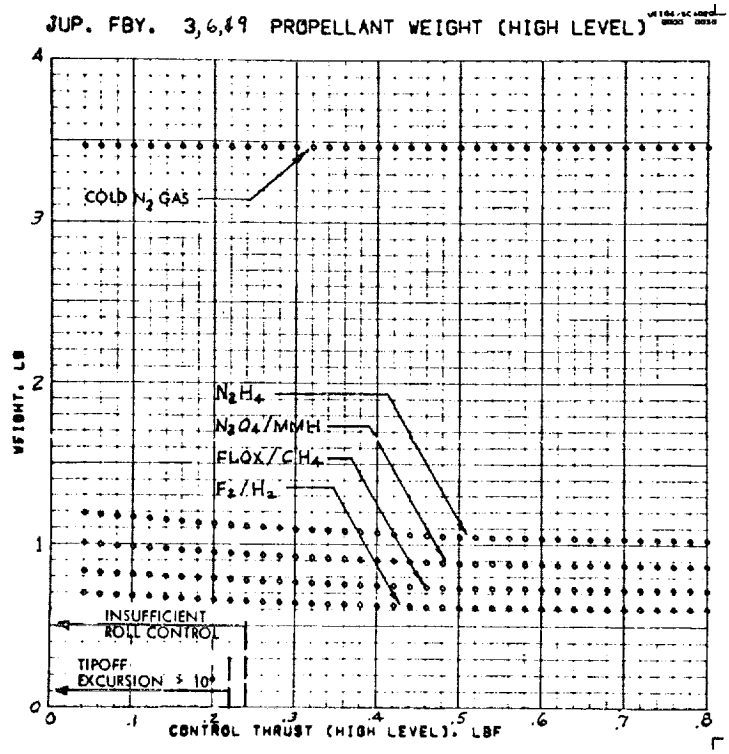

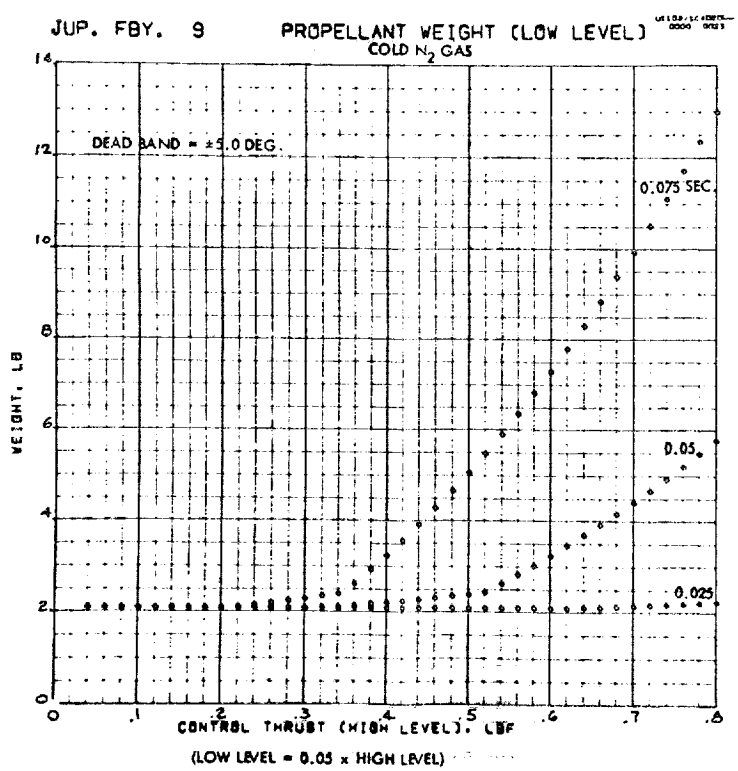

Fig. 62 Jupiter Flyby ACS Requirements
Thruster Separation = 25 ft

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Eight mission elements are employed.

1. Type 1 with Configuration Lunar 1.
   Tipoff rates of 0.5, 1.5, and 1.5 deg/sec, respectively, are removed using a dynamic factor of 2.0.
2. Type 2 with Configuration Lunar 1.
   Four acquisition search maneuvers (0.5 deg/sec about each axis) are initiated and terminated using a dynamic factor of 1.5.
3. Type 3 with Configuration Lunar 1.
   A 5.5 day transit cruise with unity deadband factors and solar torques of (0.2, 1.0, and 1.0) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.
4. Type 2 with Configuration Lunar 1.
   Four commanded roll-pitch-roll turns with unwinds of 0.5 deg/sec with dynamic factor = 1.25.
5. Type 4 with dynamic factor = 2.
   Roll torque of 3.0 ft-lb for 450 sec duration. Corresponds to a set of 5000 lbf main engine burns sufficient to empty propellant tanks.
6. Type 3 with Configuration Lunar 2.
   One half day orbital cruise with 0.5 deadband factors and solar torques of (0.2, 1.0, and 1.0) $\times 10^{-5}$ ft-lb about the X, Y, and Z axes respectively.
7. Type 2 with Configuration Lunar 2.
   Six initiated and terminated turns (0.5 deg/sec about each axis) pertaining to gyro reference updates in orbit (or acquisition searches). Dynamic factor = 1.25.
8. Type 2 with Configuration Lunar 2.
   Commanded high speed, roll-pitch-roll turns prior to descent burn. Dynamic factor = 1.25. To provide for one maneuver with unwind (aborted descent) and one without set $K_u = 1$, $K_R = 1.5$. Rate = 2.0 deg/sec.

Because of the relatively short transit cruise duration and the need for higher speed maneuvers, several differences in the mission structure will be noted. Most importantly, dual thruster force levels are not used since the associated propellant penalty

during cruise operation is offset by reliability and hardware weight considerations. Consequently, the exact values of solar/gravity gradient torques are of little importance since cruise mode expenditure is dominated by the threshold term discussed in paragraph 3.5.4. The low level/high level nomenclature is retained, however, since it serves to distinguish cruise mode expenditures from those associated with other maneuvers. We simply note that the ratio of high level thrust to low level thrust is nominally 1.0* for this mission.

Higher turning rate requirements are anticipated throughout the mission to meet guidance system constraints.

Subject to the preceding comments, computer results are presented in Figs. 63 , 64 , and 65 in a plot format that is identical to that of the other missions. An additional relationship between thrust level and the time required to impart a 1.0 deg/sec orbital configuration yaw rate is shown for reference purposes.** This consideration could have important guidance ramifications and cause the acceptable minimum thrust level to be raised in order to achieve high speed maneuvering capability.

The same summary table format previously used is also applied to this mission in Table 3.10. The term "low level" corresponds to the minimum thrust value defined by the tipoff yaw excursion constraint, not the much lower value defined by the (unrealizable) threshold condition. Nevertheless, the requirements for maneuvering are still 7.6 times the minimum cruise mode requirements.

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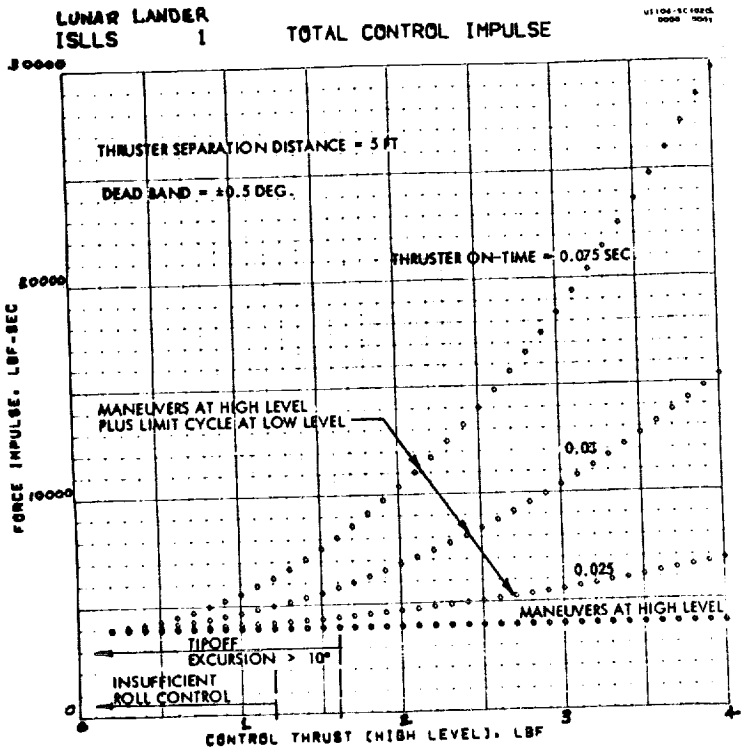
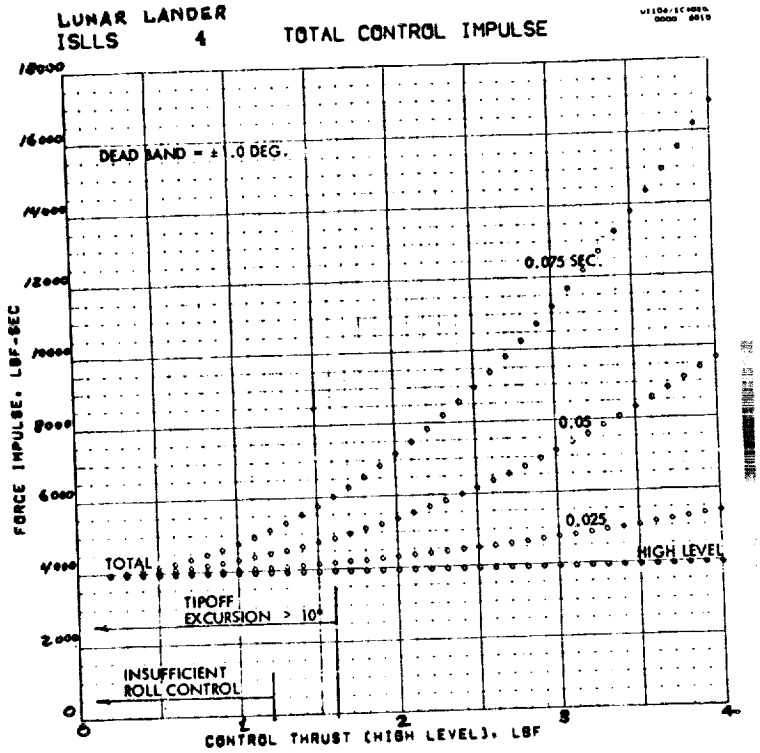
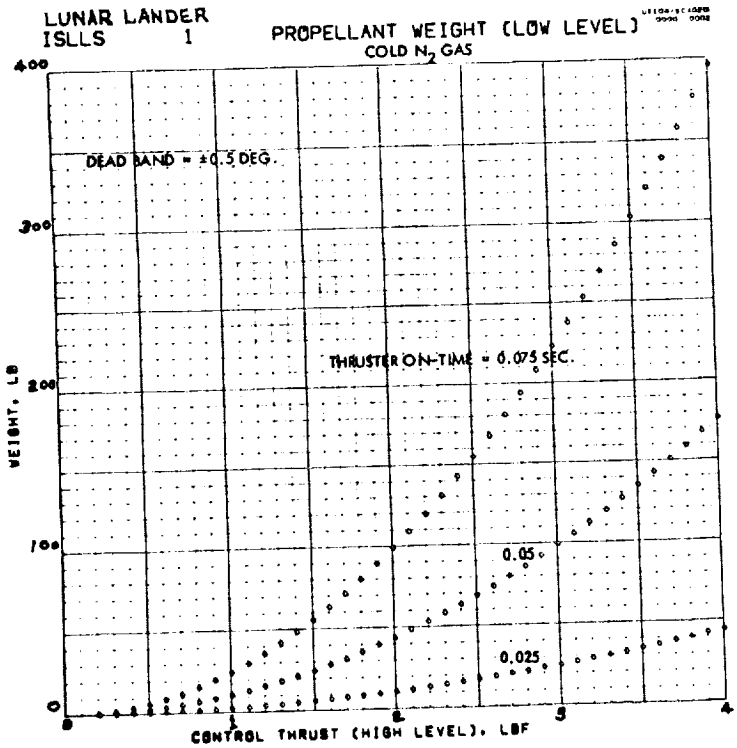
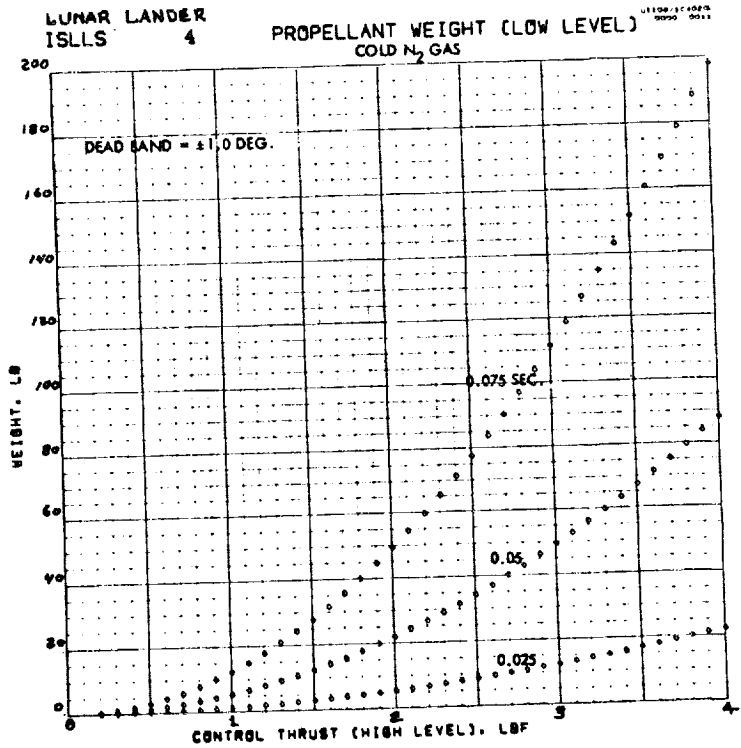
*The distinction between single pulse thrust level and steady state thrust level is neglected here. It can be accounted for in a variety of ways (e.g., an adjusted interpretation of effective on time, Δ).

**Yaw is selected because that moment of inertia is largest.









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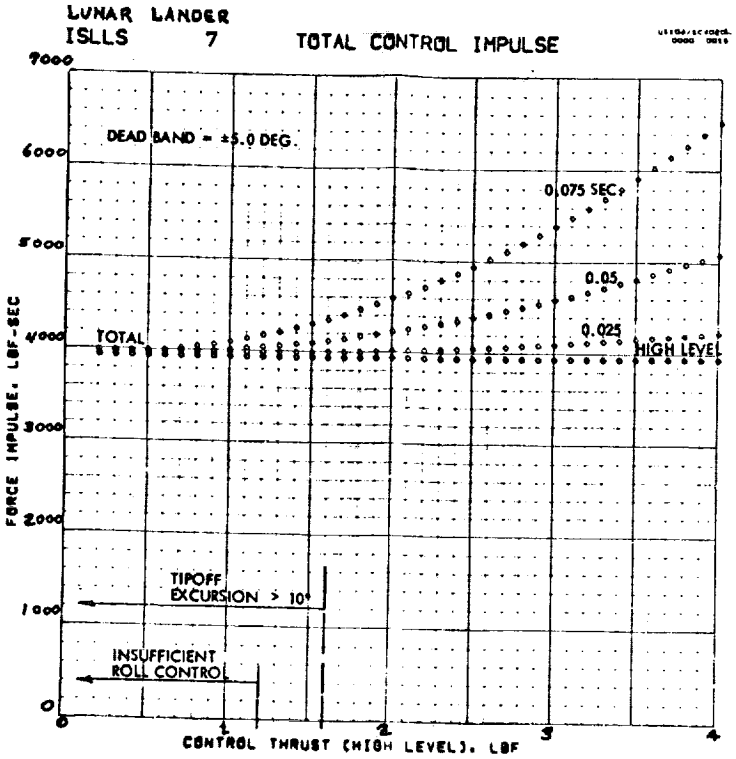

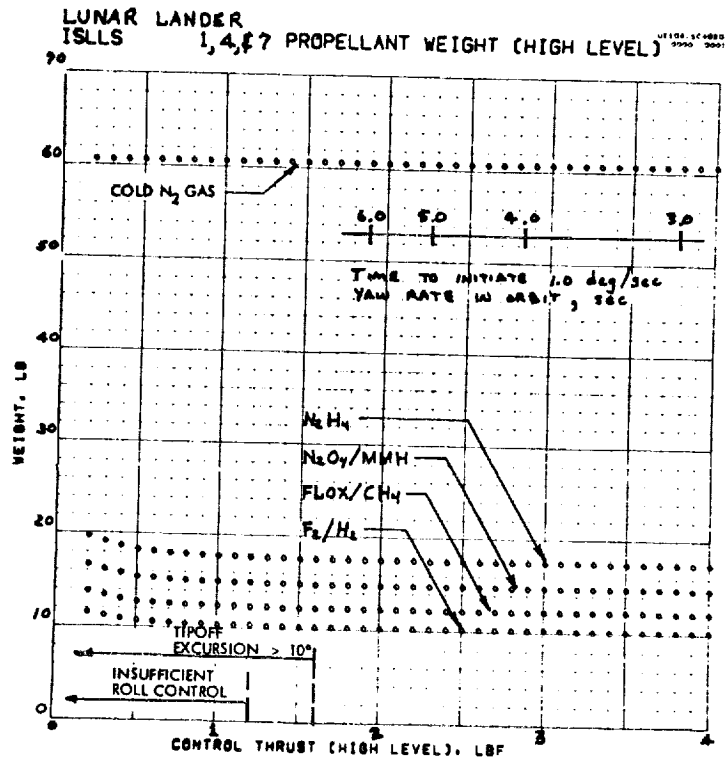

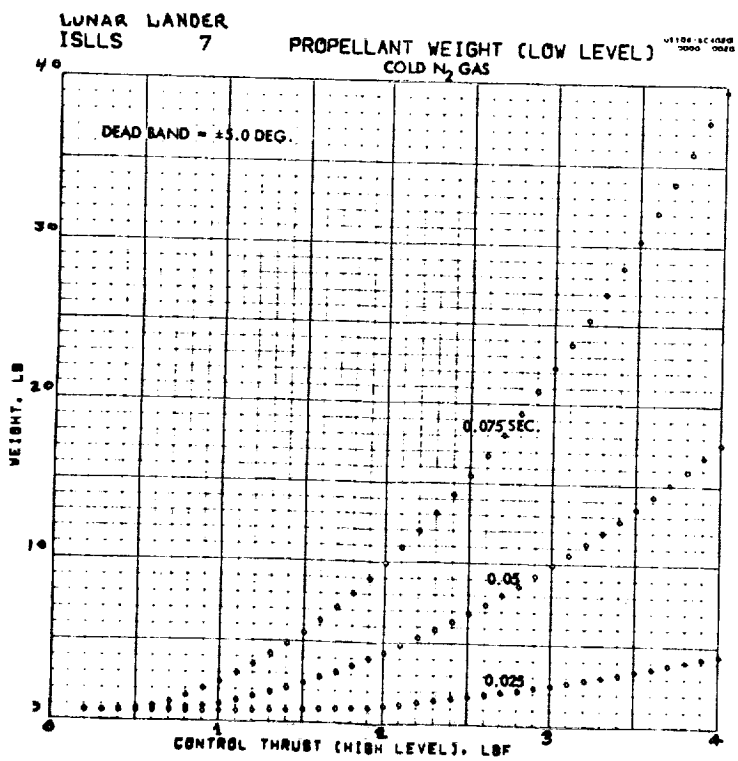

Fig. 63 Lunar Lander ACS Requirements
Thruster Separation = 5 ft

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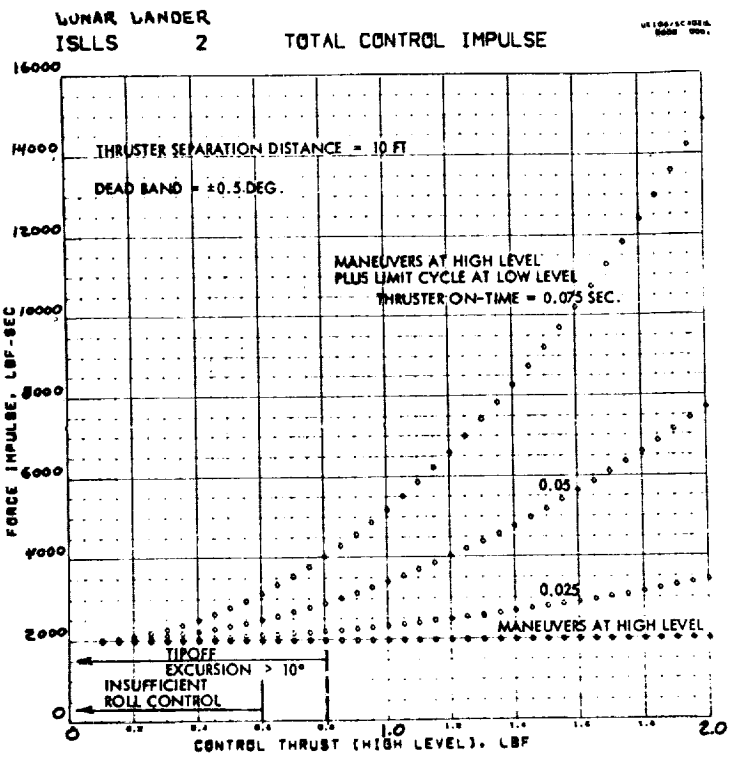

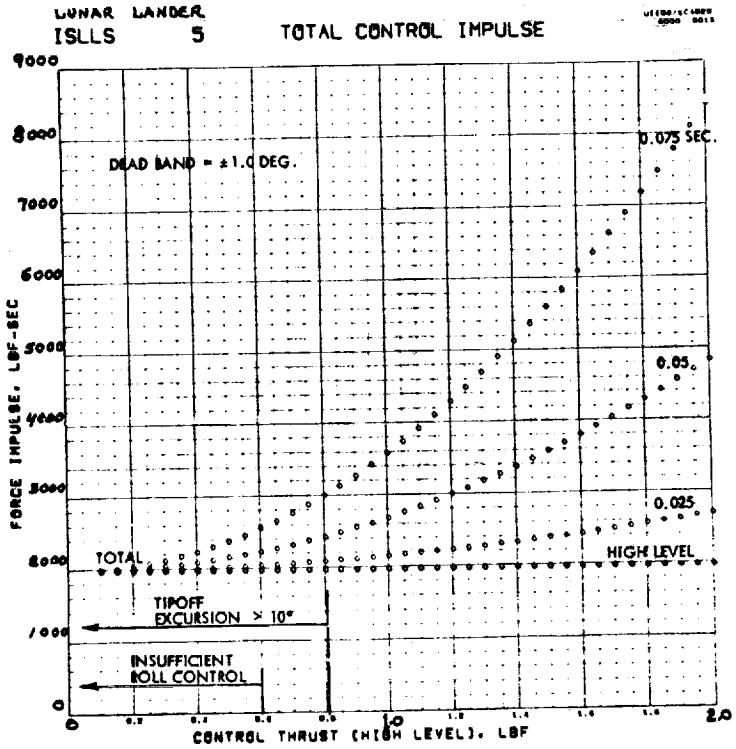

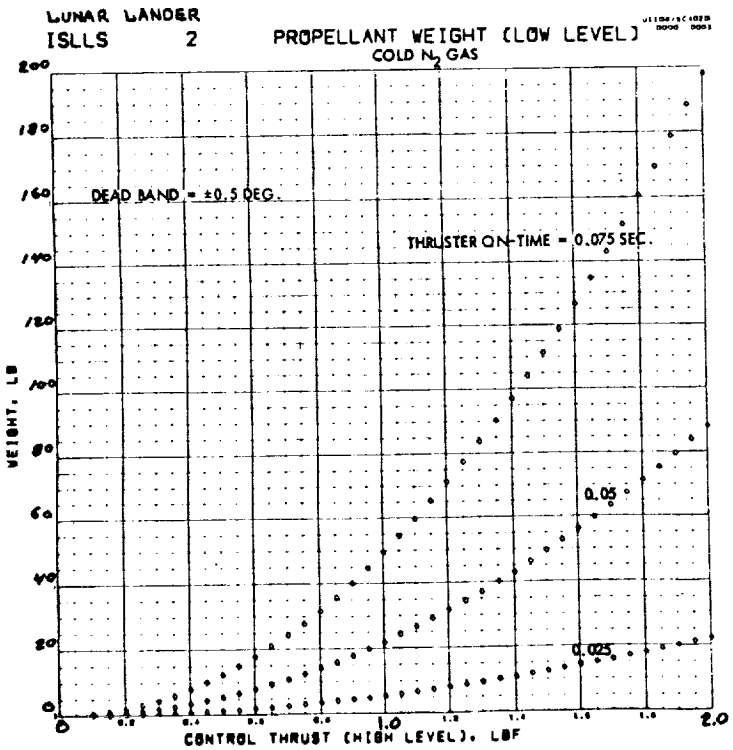

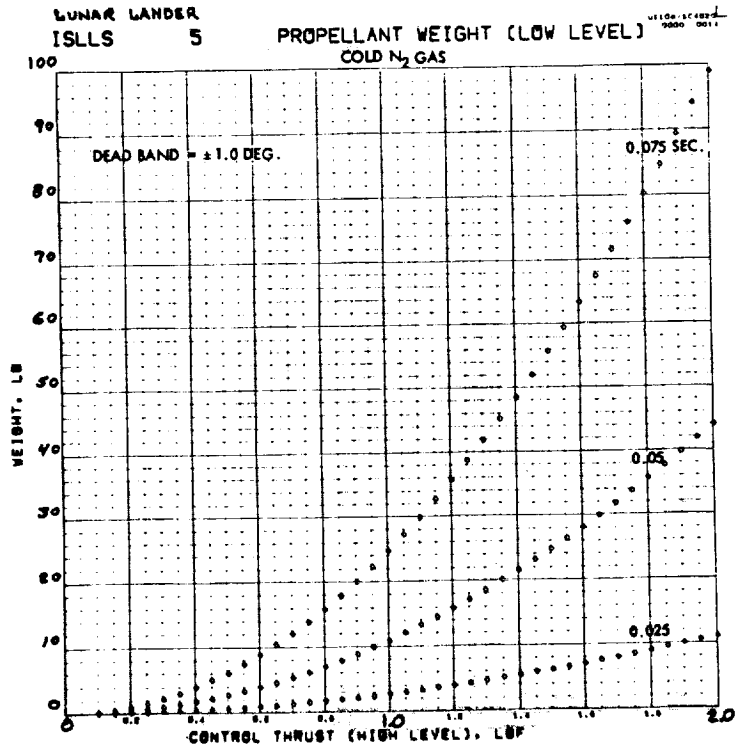

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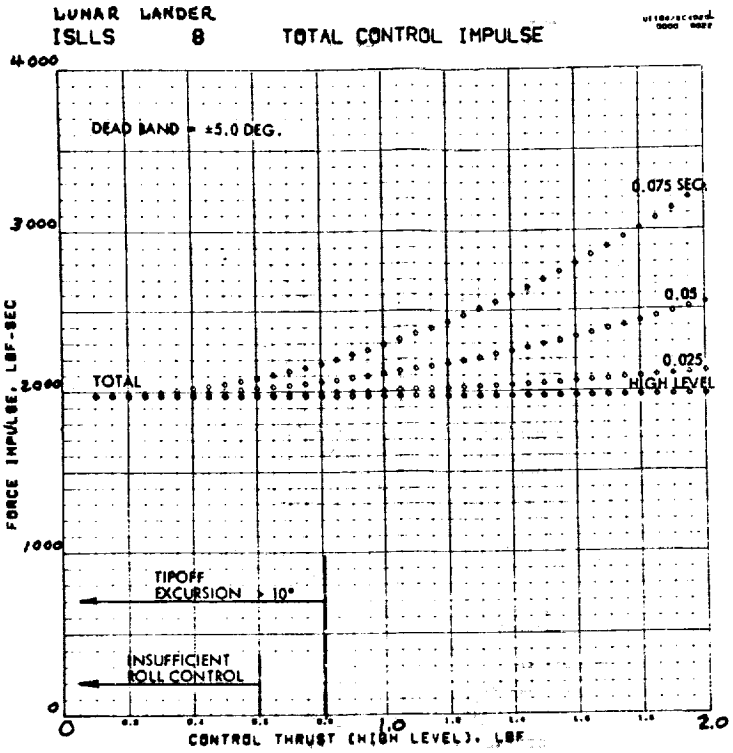

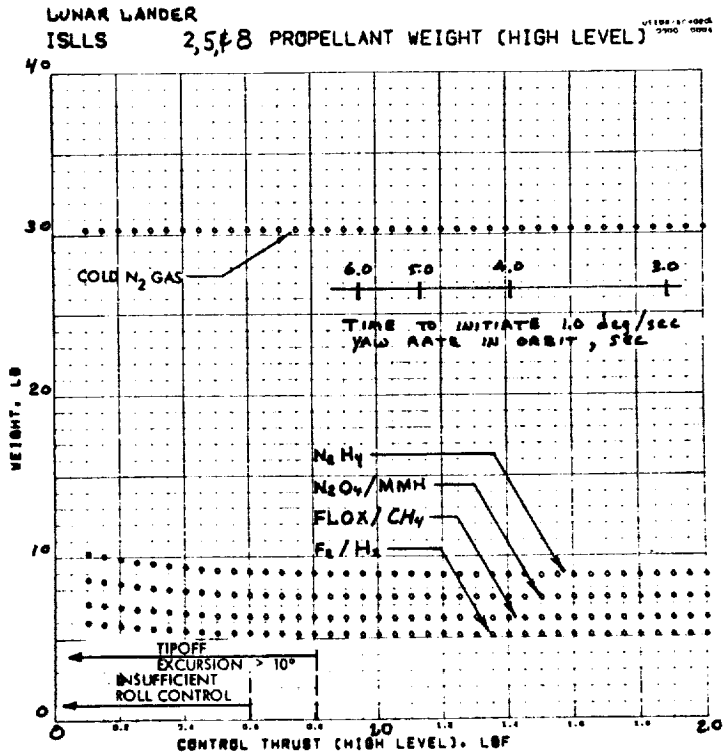

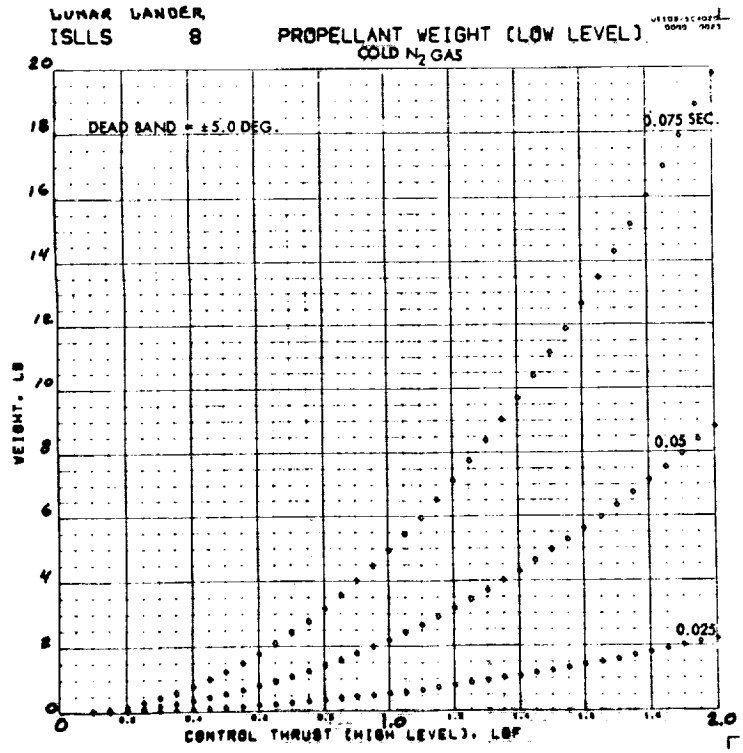

Fig. 64 Lunar Lander ACS Requirements
Thruster Separation = 10 ft

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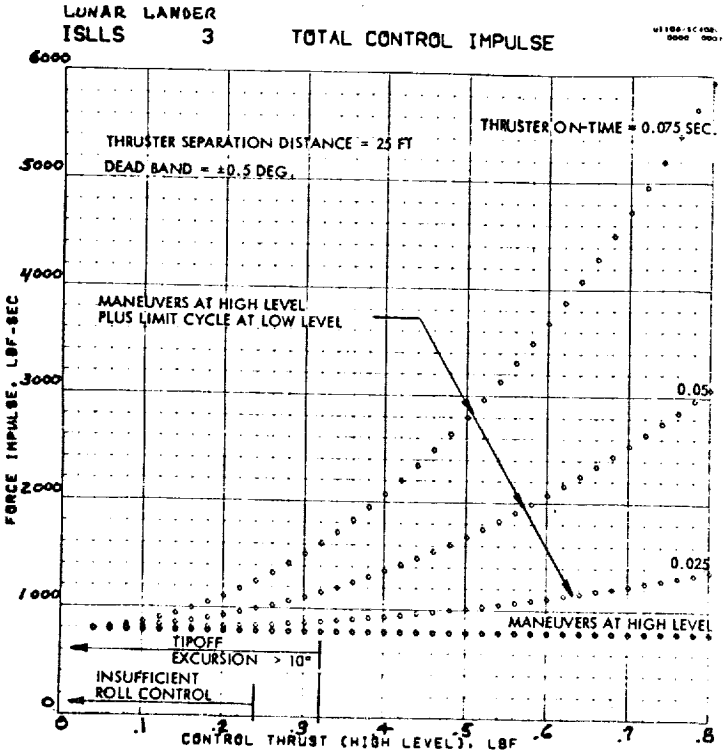

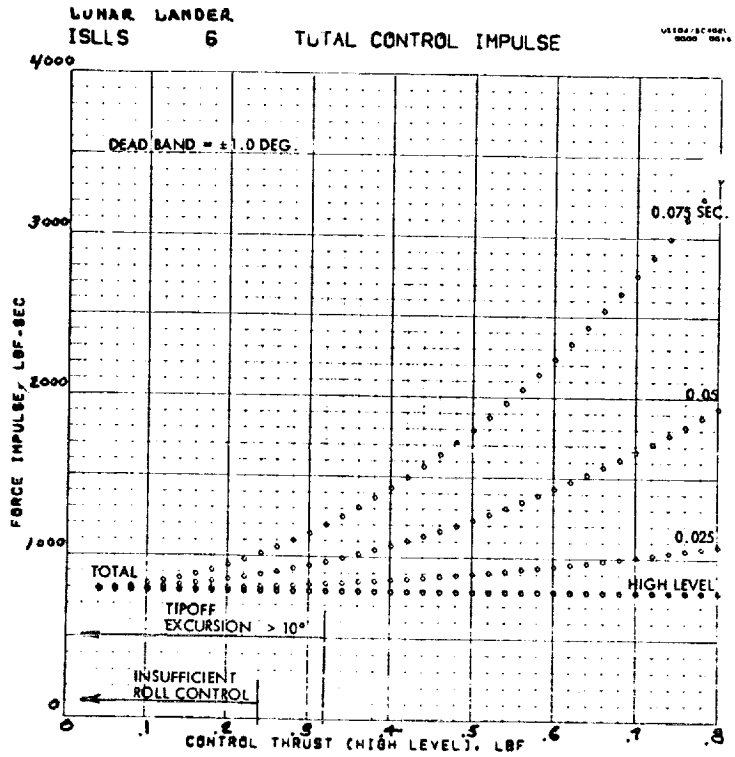

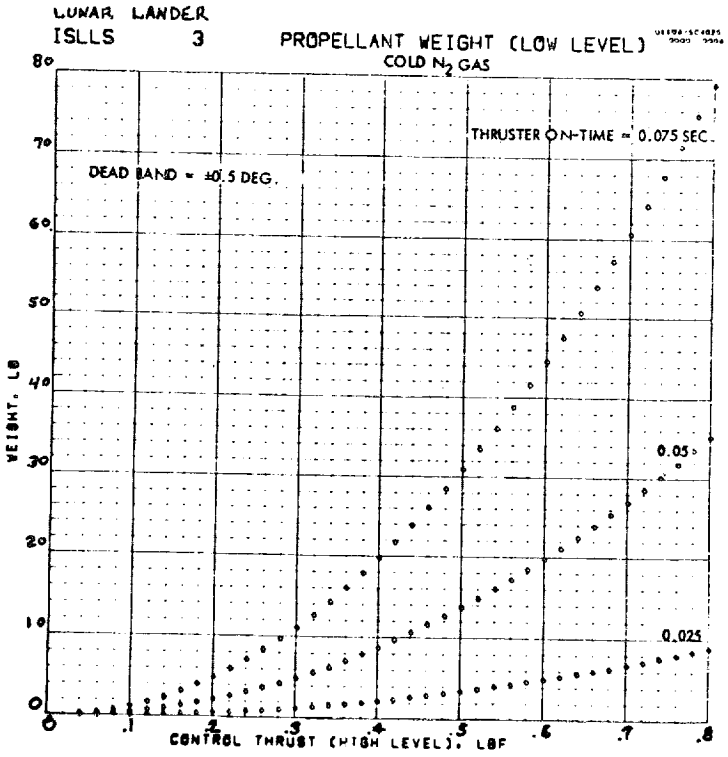

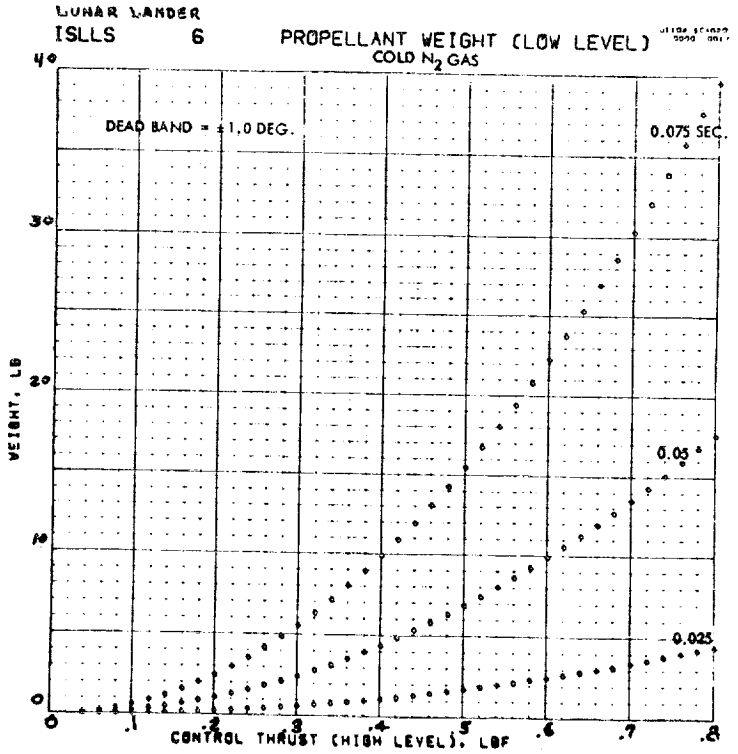

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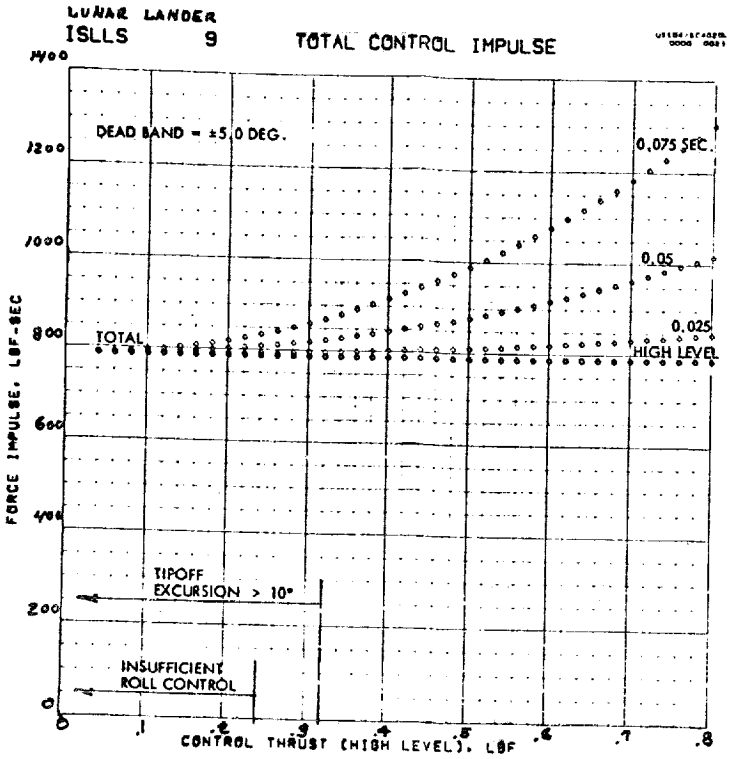

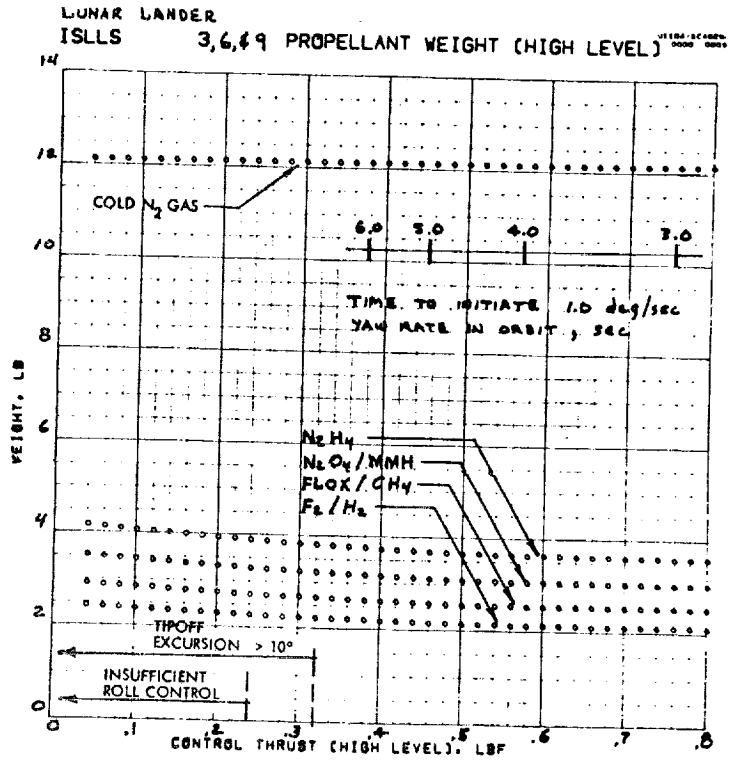

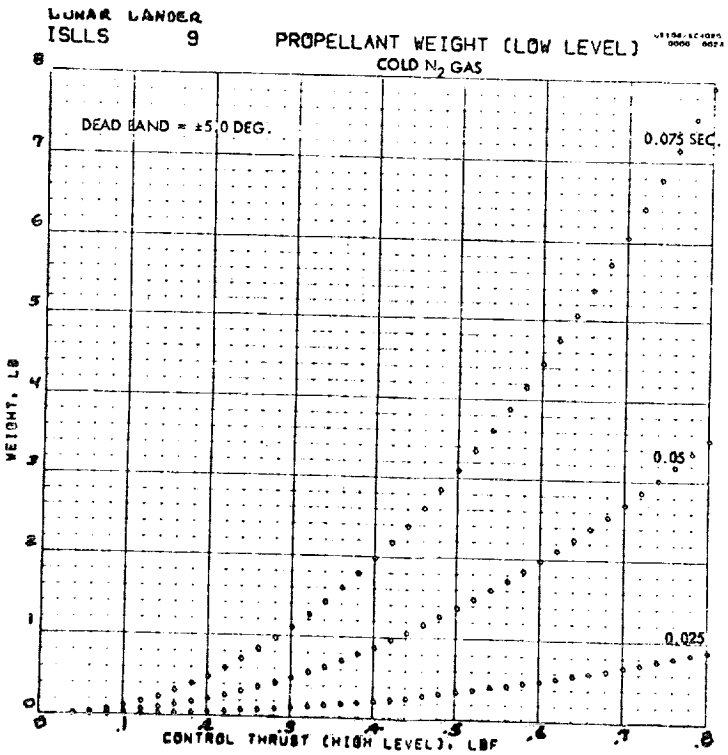

Fig. 65 Lunar Lander ACS Requirements
Thruster Separation = 25 ft

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Table 38

LUNAR LANDER – CONTROL IMPULSE SUMMARY

Thruster Separation, L (ft)	5	10	25
High Level Impulse (lb-sec)	3945 (F ≥ 1.6 lbf)	1973 (F ≥ 0.81 lbf)	789 (F ≥ 0.32 lbf)
Low Level Impulse (lb-sec)	517 (F = 1.6 lbf)	259 (F = 0.81 lbf)	103 (F = 0.32 lbf)
Total Impulse (lb-sec)	4462	2232	892

Note: Deadband = ±0.5 deg

3.7 ACS REQUIREMENTS SUMMARY

Following the analysis of the attitude control systems requirements calculations and computer outputs, the following general observations and conclusions are made:

1. Two levels of thrust are required, for most of the missions analyzed, in order to keep ACS propellant requirements to an acceptable low level. For the lunar lander mission the high-level thrusters could be used for the short transit and orbit times, as well as for maneuvers, at a relatively small penalty. High-level thrusters in the 1 to 5 lb thrust range are suitable for all the missions, while low-level thrust in the 0.05 lbf range is recommended (0.3 lbf for the lunar lander).
2. Limit cycle operation should be accomplished with the smallest combination of thrust and thruster on-time that will overcome external torques. Specifically, assuming a thruster separation of 10 ft, minimum inputs bits (MIB) of 0.001 lbf-sec or less were found desirable for the solar probe mission, 0.00125 lbs-sec for the Jupiter Orbiter and Jupiter Flyby, 0.0015 lbf-sec for the Mars Orbiter, and 0.0075 for the Lunar Lander.

3. Thruster location should be as far as practical from the center of mass and rotation. For example, if solar panels are installed then location of thrusters at panel extremities should be considered. Note, though, that there is an optimum Minimum Impulse Bit value associated with each thruster location, all other mission and spacecraft conditions being fixed. If that optimum MIB cannot be achieved, then some shorter moment arm can be found which will be more economical of ACS propellant.
4. Thruster on-time should be as low as practical for limit cycle operation. An on-time of 0.025 sec was chosen as the lower practical limit for the analysis, although it is known that somewhat lower values are achievable for specific propellants and systems.
5. Limit cycle deadband should be as large as mission requirements allow, especially during long-term cruise or orbit.
6. High-level propellant requirements are mildly sensitive to thrust level and directly proportional to specific impulse.
7. Mars Orbiter high-level propellant requirements vary from 21 percent (for F_2/H_2) to 60 percent (for cold N_2 gas) of total ACS propellant requirements when combined with an optimized low-level system using cold N_2 gas.

A summary of normal or recommended ACS parameters is presented in Table 39 together with estimated weights for propellants and total ACS systems. For all missions the use of a N_2H_4 monopropellant for the high-level system resulted in the lightest total weight. The estimation of system weights is described in the following subsection.

3.8 ACS SYSTEM WEIGHT ESTIMATES

A preliminary examination of methods for mechanising ACS systems was made in order to compare system weights and recommend a concept.

Cold N_2 Gas and Cold N_2H_4 Gas.

The only systems examined that were capable of delivering the very low thrust and minimum impulse bit required were cold N_2 gas and cold N_2H_4 gas.

Table 39

SUMMARY OF NOMINAL OR RECOMMENDED ACS PARAMETERS AND ESTIMATED WEIGHTS

Mission	Lunar Lander	Solar Probe	Jupiter Orbiter	Jupiter FlyBy	Mars Orbiter
Nominal Limit Cycle Deadband (± deg)	0.5	0.5	1.0	1.0	1.0
† Nominal Thruster Separation Distance Assumed (ft)	10	10	10	10	10
Recommended Limit Cycle Pulse Width (sec)	≤.025	≤.025	≤.025	≤.025	≤.025
Recommended Low-Level Thrust (lbf)	≤.30	≤.06	≤.04	≤.05	≤.05
Recommended High-Level Thrust (lbf)	1 to 5	1 to 5	1 to 5	1 to 5	1 to 5
Low Level Impulse Required (lb-sec)	21	454	2503	801	370
High Level Impulse Required at 1.4 lbf (lb-sec)	1972	717	460	508	564
Sum of Low and High-Level Impulse (lb-sec)	1993	1171	2963	1309	934
*Low Level Propellant (N_2 gas) (lb)	0.32	6.99	38.51	12.32	5.70
*High-Level Propellant (lb)					
N_2 gas	30.35	11.03	7.08	7.81	8.68
N_2H_4	8.83	3.21	2.06	2.30	2.55
N_2O_4/MMH	7.46	2.71	1.74	1.94	2.15
FLOX/CH_4	6.18	2.25	1.44	1.61	1.79
F_2/H_2	5.16	1.88	1.20	1.34	1.49
Estimated Total ACS System Wt (lb)					
Low-Level/High-Level (1.4 lbf)					
N_2 Gas/N_2 Gas	97.4	68.8	155.9	73.3	62.9
N_2 Gas/N_2H_4	48.6	55.1	125.9	66.6	54.5
N_2 Gas/N_2O_4/MMH	59.5	72.9	145.0	84.5	72.4
N_2 Gas/FLOX/CH_4	63.2	72.4	144.7	84.2	72.0
N_2 Gas/F_2/H_2	62.4	73.2	145.6	86.1	72.7
N_2H_4 Gas/N_2H_4	48.5	69.4	50.0	44.0	47.0

* Impulse Propellant With No Redundancy

† Thruster Separation of 20 to 25 ft, easily achievable with spacecraft using solar panels for power, would reduce thrust levels and impulse requirements in direct proportion to the increased separation distance.

The cold N_2 gas system is the most widely used to date, is inherently simple, and easily storable. Cycle life is limited by pressure regulators and solenoid valves and contamination is one of the most severe problems, leading to leakage. The dominant environmental problem is vibration with the pressure regulator most susceptible to vibration failure. Attractive applications are in thrust levels of one lb or less. A cold N_2 gas system, with two levels of thrust controlled by varying regulated pressure, can meet all requirements for the missions and spacecraft studied.

Use of cold N_2H_4 gas for low-level thrust in combinations with N_2H_4 monopropellant high-level thrusters was examined briefly. This system has the dual advantages of use of a common propellant coupled with the lowest estimated weight for each mission. For the most demanding mission (Solar Probe) the ACS system weight is estimated at 69.4 pounds as compared to 125.9 pounds for the next lightest system which uses cold N_2 gas for low-level and N_2H_4 for high level control.

Heating of the cold gases was not evaluated, but may prove attractive in increasing the delivered specific impulse.

N_2H_4 Monopropellant

N_2H_4 monopropellant is useful and attractive in the 1 to 5 lb thrust range and is capable of delivering a specific impulse of about 235 sec steady state. The main problem is keeping the propellant from freezing (at +34°F). This system is not suitable for the low-level ACS requirements studied but yields the lightest weights for the high-level thrust applications, as shown by Table 39.

Bipropellants

Bipropellants are most useful at thrust levels exceeding 5 or 10 lb. It is therefore marginal to consider bipropellants even for high-level ACS thrusters for the configurations studied. I_{sp} values varying from about 275 sec for MMH to about 400 sec for

F_2/H_2 are realizable in the 5 to 10 lb thrust range. Problems with low thrust bipropellant engines includes hard-start, valve leakage in hard vacuum, injector heat soakback in buried installations, and orifice plugging.

Total ACS system estimated weights are presented in Tables 40 through 44 for alternate missions and propellant combinations. Assumptions made in preparing these estimates are:

1. 10 percent excess propellants
2. Thruster redundancy on both low and high thrust levels. Use 4 squib valves to transfer
3. Titanium tanks used for all gases and liquids except aluminum tanks used for LH_2 and LF_2
4. 1.5 safety factor used in all tanks. 10 percent ullage in all liquid tanks
5. Minimum tank weight of 0.3 lb
6. N_2 thrusters have 15 psia chamber pressure. Monopropellant and bipropellant thrusters operate at 300 psia (except F_2/H_2 100 psia)
7. H_2 and F_2 are gaseous when introduced into chamber. Chamber pressure is 100 psia. Two pressure regulators required for separate gases
8. No instrumentation weight included

Note from the tables that the alternate bipropellants have essentially constant system weight for a given application in spite of differences in I_{sp}.

Table 40

MARS ORBITER ACS SYSTEM WEIGHT

Components	Propellant Combination, Low Level/High Level & Wts (Lb)					
	N_2/N_2	N_2/N_2H_4	N_2/N_2O_4-MMH	N_2/FLOX-CH_4	N_2/H_2-F_2	N_2H_4 Cold/N_2H_4
N_2 Gas	19.9	7.7	7.7	7.7	7.7	–
N_2 Tank	22.8	8.9	8.9	8.9	8.9	–
N_2 Fill Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Squib Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Regulator	1.2	1.2	1.2	1.2	1.2	1.2(1)
12 N_2 Thrusters	3.6	3.6	3.6	3.6	3.6	3.6
12 N_2 Thrusters (R)(2)	3.6	3.6	3.6	3.6	3.6	3.6
4 Squib Valves	0.8	0.8	0.8	0.8	0.8	0.8
Fuel Fill Valve	–	0.2	0.2	0.2	0.2	0.2
Oxid Fill Valve	–	–	0.2	0.2	0.2	–
N_2 Regulator	–	1.2	1.2	1.2	2.4	–
Fuel	–	3.5	1.2	0.4	0.2	8.5
Fuel Tank	–	0.3	0.3	0.3	0.3	0.8
Oxid	–	–	1.8	2.1	1.9	2.4(3)
Oxid Tank	–	–	0.3	0.3	0.3	2.2(4)
12 Thrusters	6.0	9.6	18.0	18.0	18.0	9.6
12 Thrusters (R)(2)	6.0	9.6	18.0	18.0	18.0	9.6
Lines, Brackets Clamps, Fittings	4.5	4.5	5.5	5.5	5.5	4.5
Total Wt, (lb)	68.8	55.1	72.9	72.4	73.2	47.0

(1) He Regulator
(2) Redundant Thrusters
(3) Gas Generator
(4) Gas Generator Plenum

Table 41

SOLAR PROBE ACS SYSTEM WEIGHT

Components	Propellant Combination, Low Level/High Level & Wts (Lb)					
	N_2/N_2	N_2/N_2H_4	N_2/N_2O_4-MMH	N_2/FLOX-CH_4	N_2/H_2-F_2	N_2H_4 Cold/N_2H_4
N_2 Gas	50.2	42.4	42.4	42.4	42.4	–
N_2 Tank	79.6	47.4	47.4	47.4	47.4	–
N_2 Fill Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Squib Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Regulator	1.2	1.2	1.2	1.2	1.2	1.2 (1)
12 N_2 Thrusters	3.6	3.6	3.6	3.6	3.6	3.6
12 N_2 Thrusters (R) (2)	3.6	3.6	3.6	3.6	3.6	3.6
4 Squib Valves	0.8	0.8	0.8	0.8	0.8	0.8
Fuel Fill Valve	–	0.2	0.2	0.2	0.2	0.2
Oxid Fill Valve	–	–	0.2	0.2	0.2	–
N_2 Regulator	–	–	1.2	1.2	2.4	–
Fuel	–	2.3	0.7	0.2	0.1	29.8
Fuel Tank	–	0.3	0.3	0.3	0.3	1.9
Oxid	–	–	1.2	1.4	1.2	2.4 (3)
Oxid Tank	–	–	0.3	0.3	0.3	2.2 (4)
12 Thrusters	6.0	9.6	18.0	18.0	18.0	9.6
12 Thrusters (R) (2)	6.0	9.6	18.0	18.0	18.0	9.6
Lines, Brackets Clamps, Fittings	4.5	4.5	5.5	5.5	5.5	4.5
Total Wt, (lb)	155.9	125.9	145.0	144.7	145.6	69.4

(1) He Regulator
(2) Redundant Thrusters
(3) Gas Generator
(4) Gas Generator Plenum

Table 42

JUPITER ORBITER ACS SYSTEM WEIGHT

Components	Propellant Combination, Low Level/High Level & Wts (Lb)					
	N_2/N_2	N_2/N_2H_4	N_2/N_2O_4-MMH	N_2/FLOX-CH_4	N_2/H_2-F_2	N_2H_4 Cold/N_2H_4
N_2 Gas	22.1	13.6	13.6	13.6	13.6	–
N_2 Tank	25.1	15.5	15.5	15.5	15.5	–
N_2 Fill Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Squib Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Regulator	1.2	1.2	1.2	1.2	1.2	1.2 (1)
12 N_2 Thrusters	3.6	3.6	3.6	3.6	3.6	3.6
12 N_2 Thrusters (R) (2)	3.6	3.6	3.6	3.6	3.6	3.6
4 Squib Valve	0.8	0.8	0.8	0.8	0.8	0.8
Fuel Fill Valve	–	0.2	0.2	0.2	0.2	0.2
Oxid Fill Valve	–	–	0.2	0.2	0.2	–
N_2 Regulator	–	1.2	1.2	1.2	2.4	–
Fuel	–	2.5	0.8	0.3	0.2	11.3
Fuel Tank	–	0.3	0.3	0.3	0.3	1.0
Oxid	–	–	1.3	1.5	2.3	2.4 (3)
Oxid Tank	–	–	0.3	0.3	0.3	2.2 (4)
12 Thrusters	6.0	9.6	18.0	18.0	18.0	9.6
12 Thrusters (R) (2)	6.0	9.6	18.0	18.0	18.0	9.6
Lines, Brackets Clamps, Fittings	4.5	4.5	5.5	5.5	5.5	4.5
Total Wt, (lb)	73.3	66.6	84.5	84.2	86.1	50.0

(1) He Regulator
(2) Redundant Thrusters
(3) Gas Generator
(4) Gas Generator Plenum

Table 43

JUPITER FLY-BY ACS SYSTEM WEIGHT

Components	Propellant Combination, Low Level/High Level & Wts (Lb)					
	N_2/N_2	N_2/N_2H_4	N_2/N_2O_4-MMH	N_2/FLOX-CH_4	N_2/H_2-F_2	N_2H_4 Cold/N_2H_4
N_2 Gas	15.8	6.3	6.3	6.3	6.3	–
N_2 Tank	17.5	7.0	7.0	7.0	7.0	–
N_2 Fill Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Squib Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Regulator	1.2	1.2	1.2	1.2	1.2	1.2 (1)
12 N_2 Thrusters	3.6	3.6	3.6	3.6	3.6	3.6
12 N_2 Thrusters (R) (2)	3.6	3.6	3.6	3.6	3.6	3.6
4 Squib Valves	0.8	0.8	0.8	0.8	0.8	0.8
Fuel Fill Valve	–	0.2	0.2	0.2	0.2	0.2
Oxid Fill Valve	–	–	0.2	0.2	0.2	–
N_2 Regulator	–	1.2	1.2	1.2	2.4	–
Fuel	–	2.8	0.9	0.3	0.1	9.1
Fuel Tank	–	0.3	0.3	0.3	0.3	0.9
Oxid	–	–	1.5	1.7	1.4	2.4 (3)
Oxid Tank	–	–	0.3	0.3	0.3	2.2 (4)
12 Thrusters	6.0	9.6	18.0	18.0	18.0	9.6
12 Thrusters (R) (2)	6.0	9.6	18.0	18.0	18.0	9.6
Lines, Brackets Clamps, Fittings	4.5	4.5	5.5	5.5	5.5	4.5
Total Wt, (lb)	62.9	54.5	72.4	72.0	72.7	44.0

(1) He Regulator
(2) Redundant Thrusters
(3) Gas Generator
(4) Gas Generator Plenum

Table 44

LUNAR LANDER ACS SYSTEM WEIGHT

Components	Propellant Combination, Low Level/High Level & Wts (Lb)					
	N_2/N_2	N_2/N_2H_4	N_2/N_2O_4-MMH	N_2/FLOX-CH_4	N_2/H_2-F_2	N_2H_4 Cold/N_2H_4
N_2 Gas	33.8	3.0	2.5	2.3	1.8	–
N_2 Tank	37.5	0.6	0.6	0.6	0.4	–
N_2 Fill Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Squib Valve	0.2	0.2	0.2	0.2	0.2	–
N_2 Regulator	1.2	1.2	1.2	1.2	1.2	1.2(1)
12 N_2 Thrusters	3.6	3.6	3.6	3.6	3.6	3.6
12 N_2 Thrusters (R)(2)	3.6	3.6	3.6	3.6	3.6	3.6
4 Squib Valve	0.8	0.8	0.8	0.8	0.8	0.8
Fuel Fill Valve	–	0.2	0.2	0.2	0.2	0.2
Oxid Fill Valve	–	–	0.2	0.2	0.2	–
N_2 Regulator	–	1.2	1.2	1.2	2.4	–
Fuel	–	9.7	3.2	1.0	0.4	9.9
Fuel Tank	–	0.6	0.5	0.3	0.3	0.9
Oxid	–	–	5.1	5.8	5.3	2.4(3)
Oxid Tank	–	–	0.5	0.5	0.3	2.2(4)
12 Thrusters	6.0	9.6	18.0	18.0	18.0	9.6
12 Thrusters (R)(2)	6.0	9.6	18.0	18.0	18.0	9.6
Lines, Brackets, Clamps, Fittings, Elec Harness	4.5	4.5	5.5	5.5	5.5	4.5
Total Wt, (lb)	97.4	48.6	59.5	63.2	62.4	48.5

(1) He Regulator
(2) Redundant Thrusters
(3) Gas Generator
(4) Gas Generator Plenum

3.9 RECOMMENDATIONS

It is recommended that further examination of ACS systems for the spacecraft and missions studied concentrate on the following alternatives:

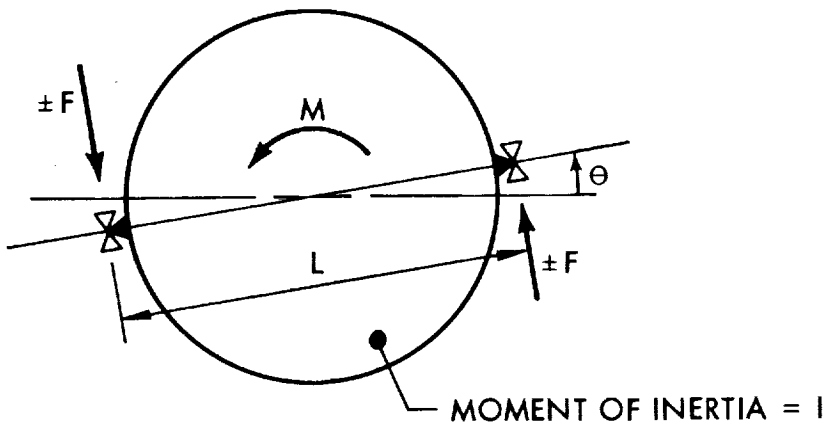
1. Cold N_2 gas for both low and high-level thrust as the most simple system
2. Cold N_2 gas for low-level and N_2H_4 for high-level thrust as a system within current technology and of lighter weight than all cold N_2 gas
3. Cold N_2H_4 gas for low-level and N_2H_4 for high-level thrust as the lightest weight system
4. Consideration of heating the cold gases to improve I_{sp} and reduce system weight

Appendix A

ATTITUDE CONTROL LIMIT CYCLE EXPENDITURE IN PRESENCE OF CONSTANT APPLIED EXTERNAL TORQUE

A.1 SYSTEM DESCRIPTION

An idealized bang-bang type controller for a single spacecraft control axis is assumed as sketched below.



Controller action generates a sequence of thruster pulses (duration = Δ seconds) such that the angle Θ is maintained within a deadband defined by $-D \leq \Theta \leq D$. In the absence of an external torque, M, the spacecraft centers a two-sided, single pulse limit cycle.

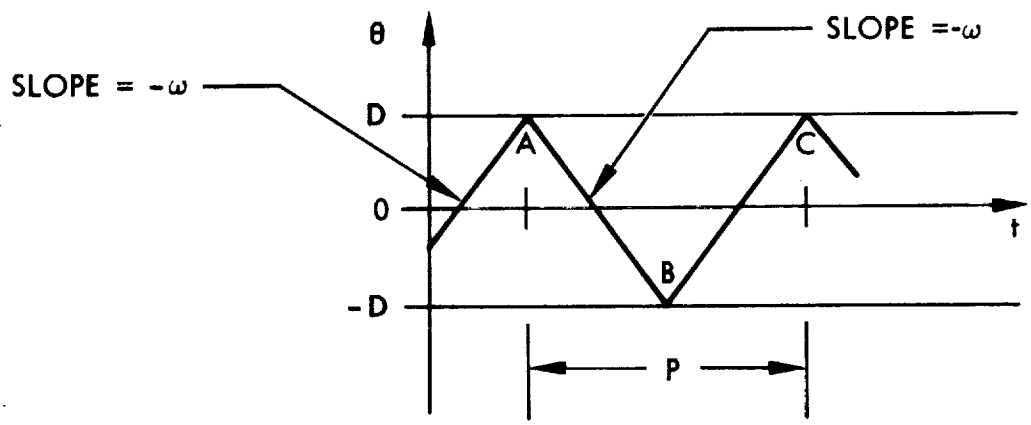
We are interested in the time integral of thruster pulses (denoted control impulse, Q, lbf-sec) since this defines attitude control propellant requirements during cruise mode operation. For convenience in treating variable length missions we emphasize the time rate of Q (i.e., $\dot{Q}$, lbf-sec per sec).

Two extreme cases are analyzed with relative ease: (a) $M = 0$, and (b) M large enough that single sided pulsing is assured. For the range of constant M levels between these limits, however, system performance is more complex owing to the combination of single and two sided pulsing and dependence on initial conditions.

A.2 ZERO EXTERNAL TORQUE CASE

In the diagram below is shown the Θ history for a symmetric limit cycle where $\Theta \triangleq \pm\omega$. Negative sense F pulses occur at points A and C, and a positive sense pulse occurs at point B. The change in angular velocity at each pulse, $\pm 2\omega$, is given by $\pm 180\ FL\Delta/\pi I$ so that

$$\omega\ ,\ \text{deg/sec} \triangleq \frac{90\ FL\Delta}{\pi I} \tag{1}$$

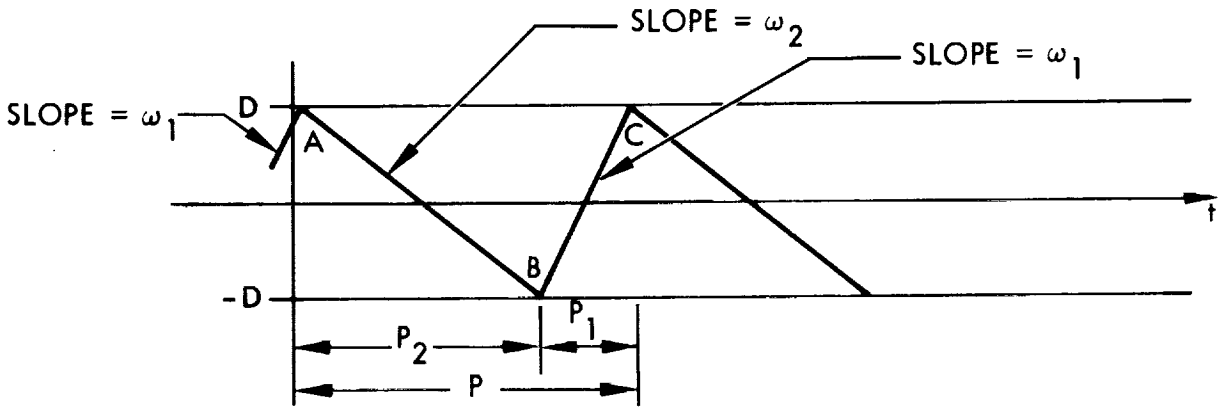


Observing that $P = 4D/\omega$ and $\dot{Q}_o = 4F\Delta/P$ we find

$$\dot{Q}_o = \frac{F\Delta\omega}{D} = \frac{90\ (F\Delta)^2\ L}{\pi I D} \tag{2}$$

We wish to consider a more general initial condition for $\dot{\Theta}$, however, given by $\omega_1 = C\omega$ where $0 \leq C \leq 2$. This contains the entire range of interest since the

$C > 2$ condition would simply generate more than one pulse (ideally) at $t = 0$ before settling into a non-symmetrical limit cycle such as shown below.



We know that $\omega_2 = \omega_1 - 2\omega = (C - 2)\omega$. Therefore,

$$P = 2D\left(\frac{1}{\omega_1} - \frac{1}{\omega_2}\right) = \frac{4D}{\omega(2C - C^2)}$$

and

$$\dot{Q} = \frac{90\,(F\Delta)^2\,L\,(2C - C^2)}{\pi I D} \tag{3}$$

Inspection of Eq. (1) shows that $\dot{Q}$ is maximized with respect to C when $C = 1$, in which case the equation reduces to Eq. (2). So initial conditions corresponding to the symmetric case give maximum propellant expenditure; a conservative approach to requirement estimation must therefore consider the $\dot{Q}_o$ defined by Eq. (2).

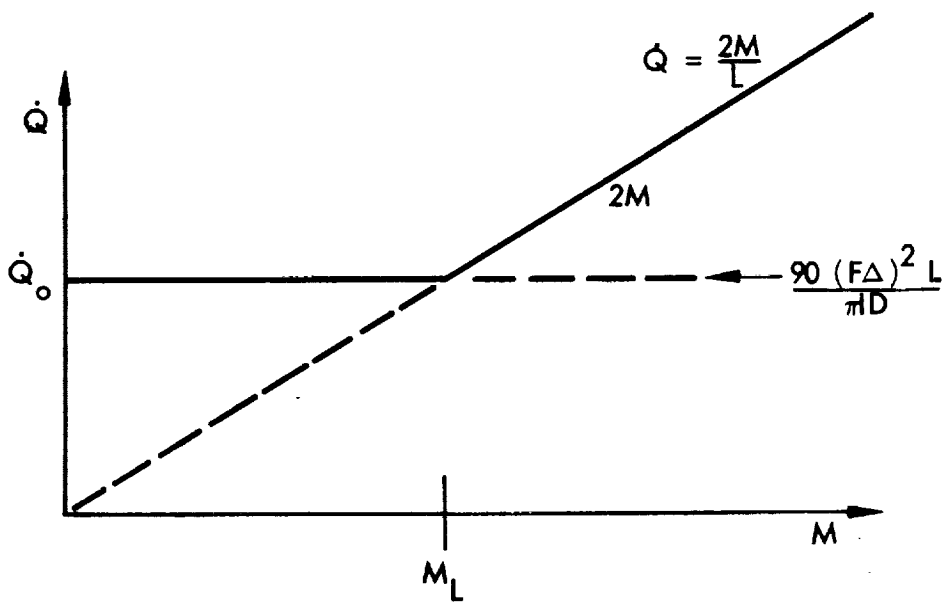
A.3 LARGE EXTERNAL TORQUE CASE

Relatively large applied torques are counteracted over long time intervals by periodic single sided pulsing. The condition that average $\dot{\theta} = 0$ over one period leads to the relation $FL\Delta = MP$. Since $\dot{Q} = 2F\Delta/P$ we see that

$$\dot{Q} = \frac{2M}{L} \tag{4}$$

Equation (4) indicates that propellant expenditure to counteract large torque levels is proportional to the torque, inversely proportional to the lever arm, and independent of thrust level or pulse width (assuming fixed I_{SP}).

At this point it is instructive to construct the following diagram.



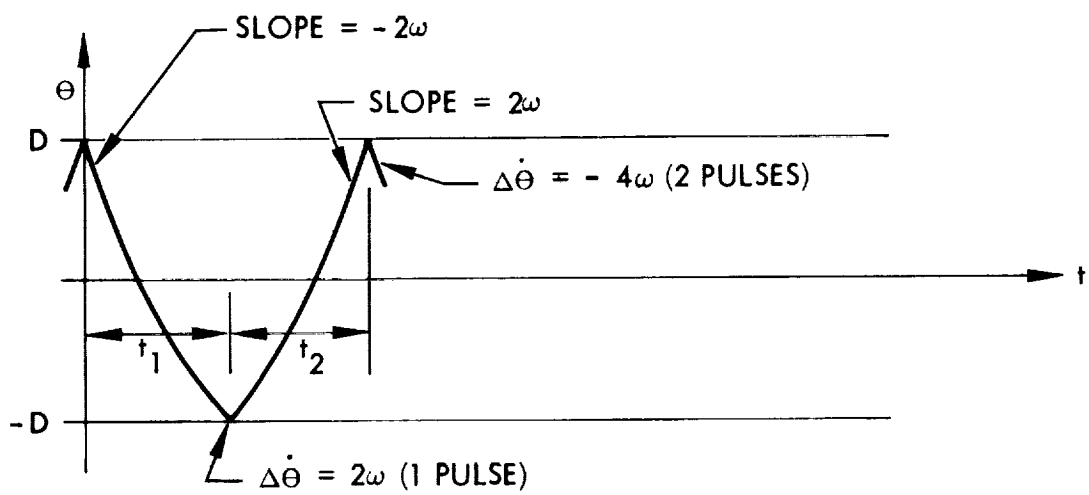
Solving for the indicated intersection yields

$$M_L = \frac{45\,(F\Delta L)^2}{\pi I D} = \frac{\pi\omega^2 I}{180\,D} \tag{5}$$

One is tempted to use the above figure in estimating propellant requirements. That is, Eq. (4) is used when $M \geq M_L$ while the $\dot{Q}_o$ threshold for $M = 0$ is used for smaller torque levels. As will be indicated in the next section, however, this approach is erroneous and leads to distinctly incorrect results in the region $0 < M < M_L$.

A.4 INTERMEDIATE EXTERNAL TORQUE CASE

We first seek that M value above which we must have a sustained single sided limit cycle (denoted the "1-1" mode) regardless of initial condition $\dot{\Theta}_0 = C\omega$, $0 \leq C \leq 2$. This was accomplished with the aid of a small computer program (conversational desk-top type). For a particular value of C, M was gradually reduced until the controller switched into a 2-1-2-1-2-1-... pulse pattern (henceforth denoted the "2-1 mode"). It was found that the highest M value at which switching occurred is associated with the extreme C values, $C = 0$ or $C = 2$. A limiting condition for the 2-1 mode limit cycle was also found that enables calculation of this M value. As sketched in the diagram below this limiting condition is characterized by incident and departure rates at the positive deadband of 2ω and -2ω, respectively. Furthermore, the two time intervals, t_1 and t_2, are equal.



We can now write two equations involving angular acceleration, a, owing to the external torque, where a, deg/sec^2 = 180 M/πI.

$$-2D = -2\omega t_1 + \frac{at_1^2}{2}$$

$$2at_1 = 2\omega$$

Eliminating t_1 we find

$$a = \frac{3}{4}\frac{\omega^2}{D}$$

We can now compute the upper switch point between the 2-1 and 1-1 modes.

$$M = \frac{3}{4}\frac{\pi\omega^2 I}{180\ D} = \frac{3}{4} M_L$$

In similar fashion it was also shown that the lower switch point between these two modes is given by $M = 1/4\ M_L$.

It is easy to see that, for a given M, the 2-1 mode $\dot{Q}$ level is three times that of the 1-1 mode. This is true since the un-paired negative pulses in the 2-1 mode must occur with the same frequency as in the 1-1 mode to counteract M. The additional two pulses in this interval simply expend additional propellant. So the appropriate expression in the 2-1 mode is

$$\dot{Q} = \frac{6M}{L}$$

The information found thus far is presented in the following diagram.

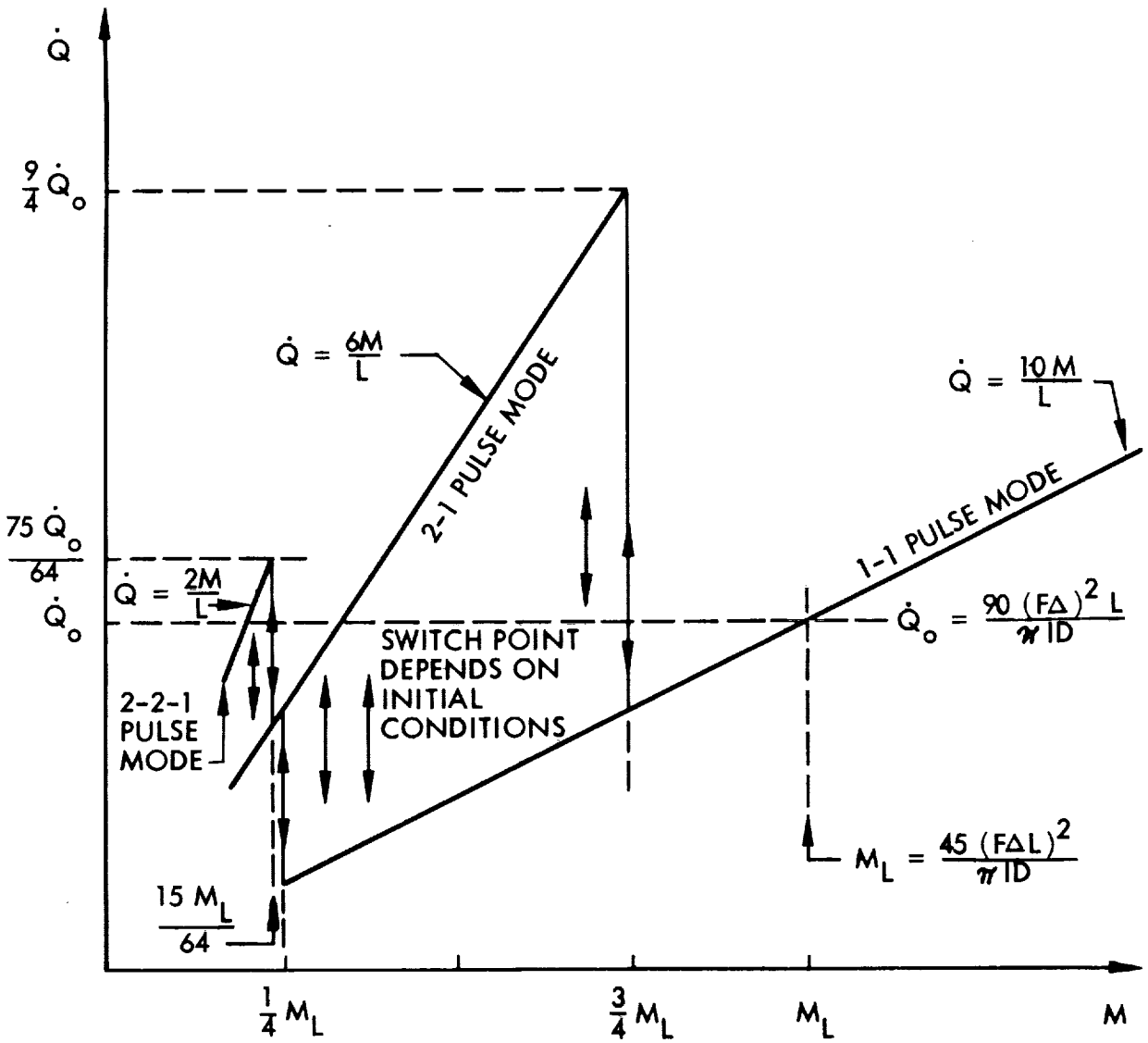
Also displayed is the performance line for the 2-2-1 mode, the upper switch point for which is at $M = 15/64\ M_L$.

The most important observation to be made is that unfavorable initial conditions can cause propellant expenditure rates 125% higher than those predicted by the over-simplified rule $\dot{Q} = \text{Max}\ (\dot{Q}_o,\ 2M/L)$. A conservative approach suggests the following expression for preliminary design purposes.

$$\dot{Q} = \text{Max}\ (2.25\ \dot{Q}_o\ ,\ 2M/L) \qquad (6)$$

Equation (6) places the threshold at the peak of the operating range on the 2-1 mode line. Since no peaks at lower M values exceed this threshold, further analysis is not required.

A secondary observation involves the strong (quadratic) dependence of $\dot{Q}_o$ on thrust level and pulse width. The desirability of low thrust levels and small pulse widths for long cruise missions is evident.



Appendix B

TRANSIENT PERFORMANCE OF AN ATTITUDE CONTROL CHANNEL USING A DIGITAL PSEUDO RATE CONTROLLER

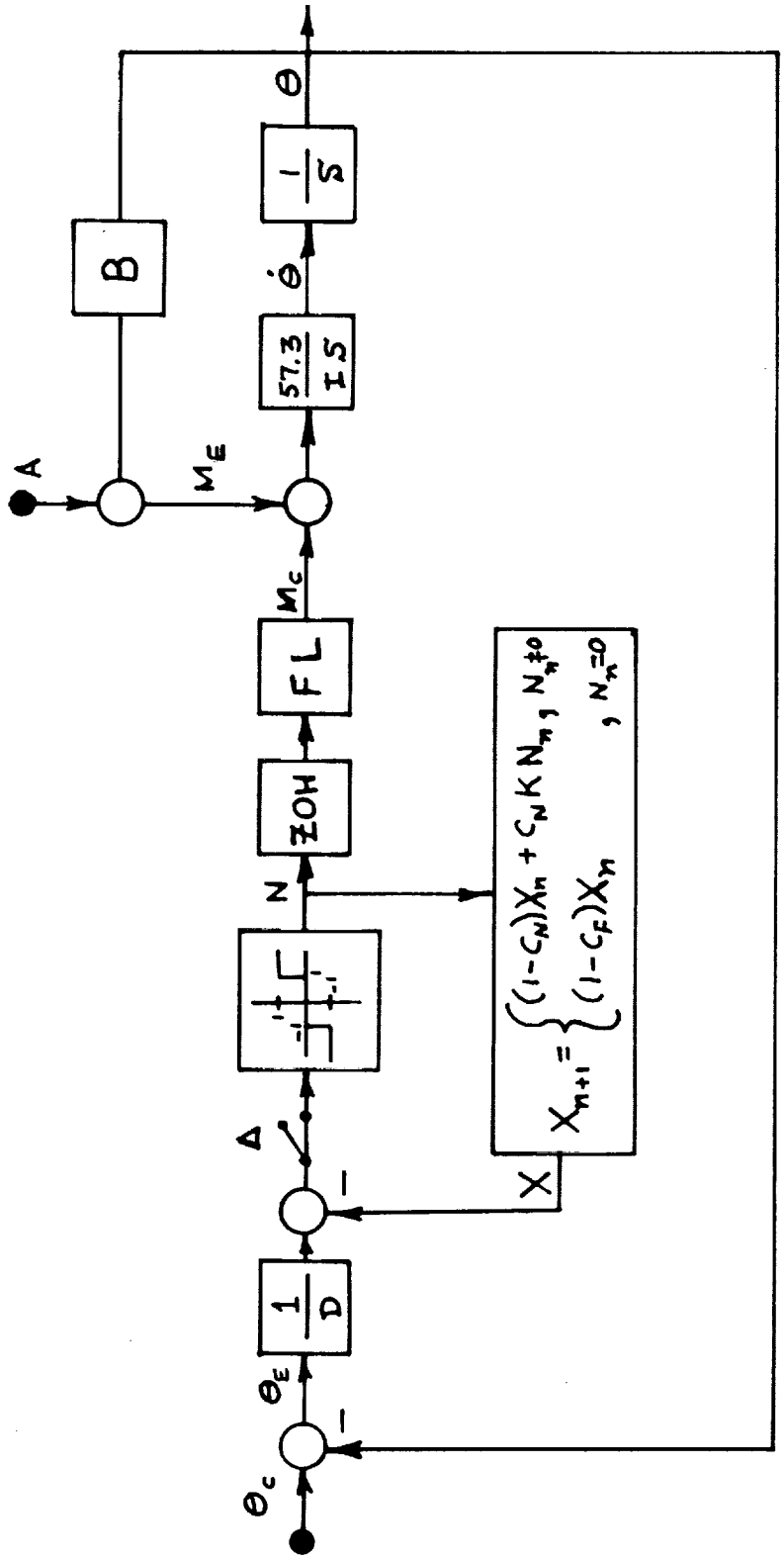
B.1 INTRODUCTION

The ratio between actual and ideal control impulse expenditure for a specified spacecraft maneuver is termed the "dynamic factor" in the attitude control system requirements task. It differs from 1.0 because realizable closed loop controller mechanizations involve rate overshoot and undershoot behavior that expends propellant in excess of that required for ideal tracking of the command. To provide a sound basis for estimating dynamic factors a digital computer model of a promising controller candidate and single spacecraft control axis was employed.

As established in the primary analysis task, dual attitude control thruster levels are indicated for the interplanetary missions. A secondary objective of this simulation effort was to make a preliminary determination of whether the same type of controller (with possible parameter adjustments) is suitable for both thrust levels.

B.2 SYSTEM DESCRIPTION

A block diagram of the system simulated is shown in Fig. B.1. Spacecraft dynamics are modeled relatively simply by equating angular acceleration to the sum of control and external torques, the latter consisting (generally) of constant and angle proportional components. Note that all angles are expressed in degree units. The diagram is appropriate to angle sensing by means of sun sensors, star sensors, or integrating rate gyros since low frequencies are of interest. A simple diagram (hence computer model) modification permits simulation of a caged (rate) gyro sensing mode.



NOTE: Symbols are defined at the end of Appendix B

Fig. B.1 Attitude Control Channel With Discrete Pseudo Rate

The controller is a sampled data version of the standard pseudo rate controller featuring unequal feedback time constants in the on and off modes. Such a controller is particularly well adapted to implementation by means of a time-shared onboard digital computer, a consideration appropriate to the spacecraft generation of interest. Primary sampling occurs at Δ second intervals which, as indicated by the zero order hold function, coincides with the thruster minimum on time. Accordingly, the control torque generated by two thrusters (separated by distance L) consists of a modulated string of pulses, each having area $\pm FL\Delta$ (or zero) where F = thruster force.

Stabilization is accomplished by the feedback difference equation that generates a static variable (denoted X) for comparison with the sampled angle error signal sealed by the reciprocal deadband. The use of a unit deadband inside the controller loop is, of course, an arbitrary arrangement. Three numerical parameters are employed in the controller difference equation. Coefficients C_N and C_F are related to the analog "on" and "off" time constants, respectively, while K is a gain term for the "on" state. Observe that, owing to the zero order hold function, there is no need for a hysteresis element such as usually found in an analog mechanization.

B.3 CRUISE MODE CONTROLLER PARAMETER SELECTION

A series of computer runs was made with typical Mars cruise spacecraft data ($I = 3500$ slug-ft^2, $D = \pm 0.5$ deg, $L = 11$ ft, $F = 0.1$ lbf, $\Delta = 0.05$ sec) to establish a satisfactory set of values for C_N, C_F, and K.

Figures B.2a, B.2b, and B.2c show a capture transient from the initial condition $\Theta_o = 1.0$ deg, $\dot{\Theta}_o = -0.0005$ deg/sec. Controller parameters are $C_N = 0.012$, $C_F = 0.0002$, $K = 7.5$. External torque was set to zero for this run, denoted Run 21. Note that Θ enters the deadband to remain thereafter in the single pulse mode at $t = 700$ sec. The two angle rates in this mode are $+0.0004$ and -0.0005 deg/sec as governed by system data and initial condition choices. Figure B.2b is the associated capture phase plane† where the switching surface and 25 thruster pulses leading

† Approximate shape constructed by 2000 linear segments thus artificially smoothing display of true discontinuous phase plane.

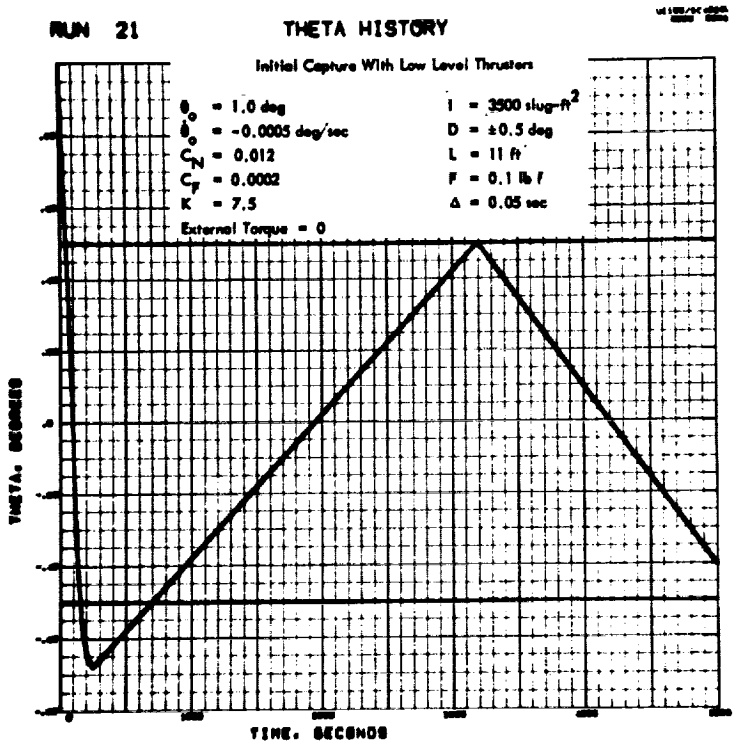


Fig. B.2a Theta Capture Transient – Run 21

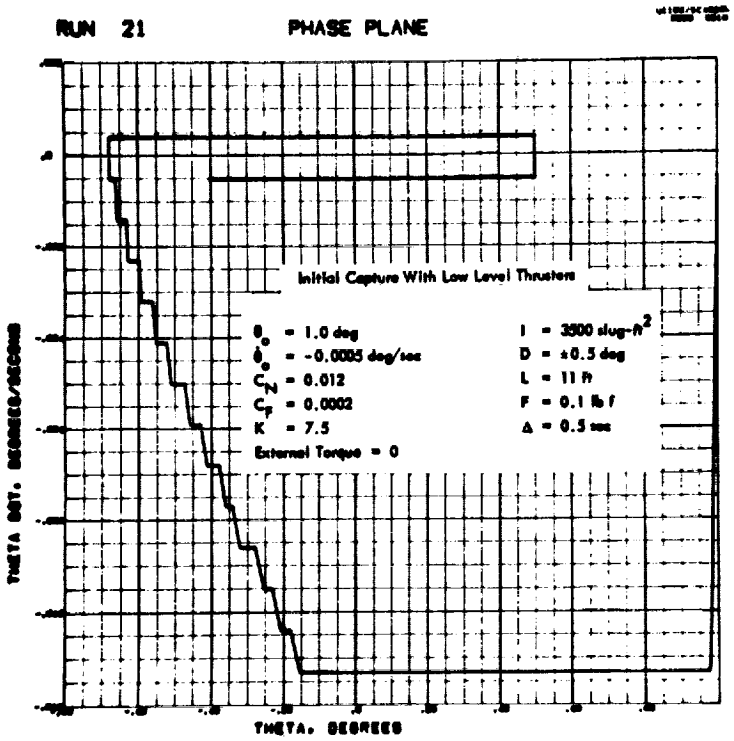


Fig. B.2b Phase Plane – Run 21

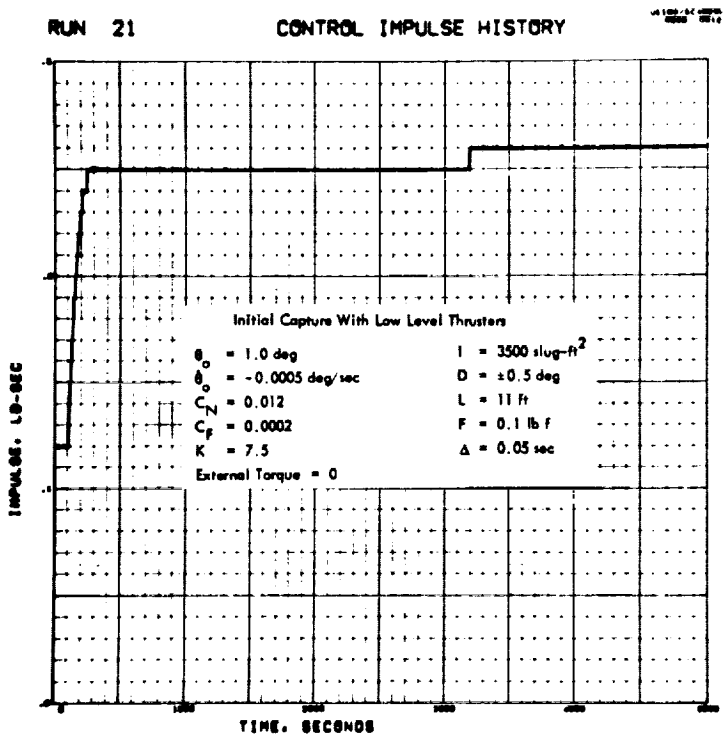


Fig. B.2c Control Impulse History – Run 21

to the single pulse mode may be seen. Cumulative control impulse expenditure is shown in Fig. B.2c where we note that the single pulse mode is entered after 25 pulses of 0.01 lb-sec impulse each.

B.4 SPACECRAFT MANEUVERS WITH LOW LEVEL THRUSTERS

The controller of Run 21 and low level thrusters were used to generate a relatively slow 180° turn maneuver from zero initial conditions and in the presence of a constant 1×10^{-5} ft-lb external torque. The command signal for Run 22 was ramped linearly from 0 to 180 degrees at a 0.2 deg/sec rate, thus simulating a typical torqued gyro reference system that ideally executes the maneuver in 15 minutes. Figures B.3a, B.3b, B.3c, and B.3d show the performance of this "nominal cruise" controller in various ways. Of particular interest are the 6–7% rate overshoot/undershoot levels

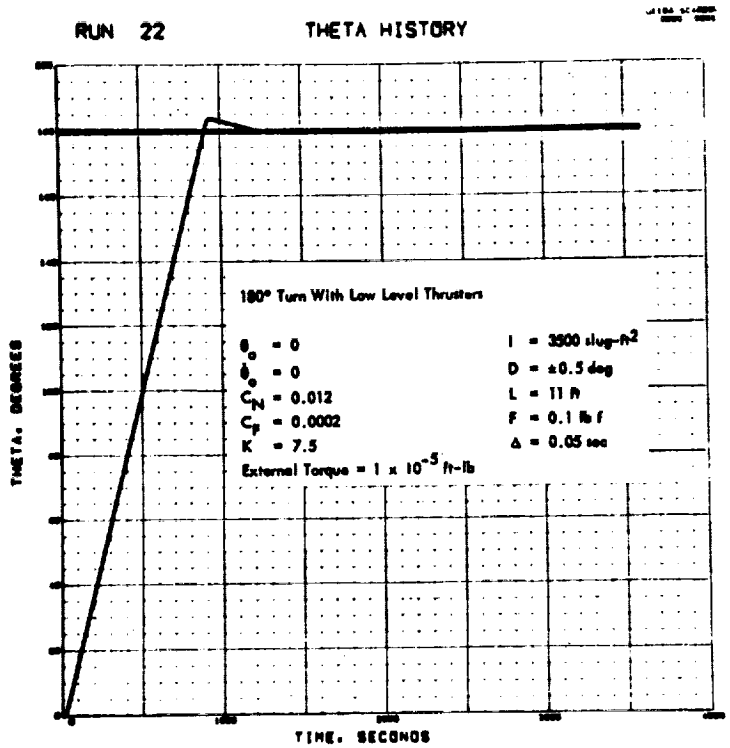


Fig. B.3a Theta Turn Maneuver History – Run 22

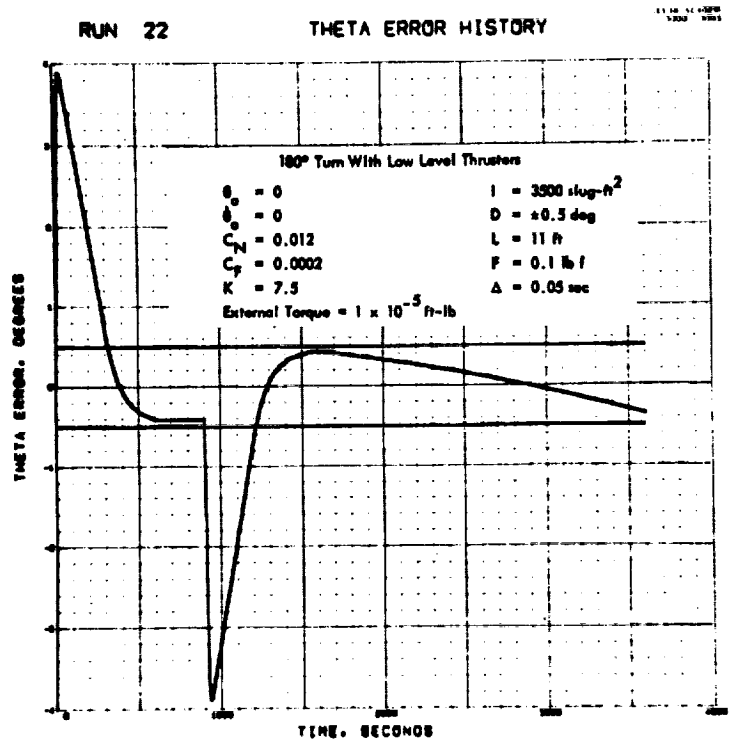


Fig. B.3b Theta Turn Error History – Run 22

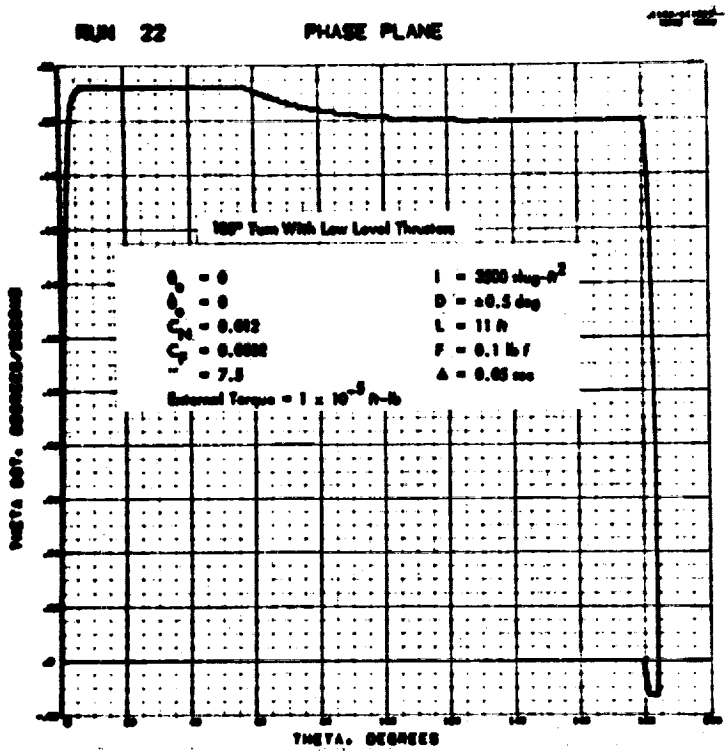


Fig. B.3c Phase Plane – Run 22

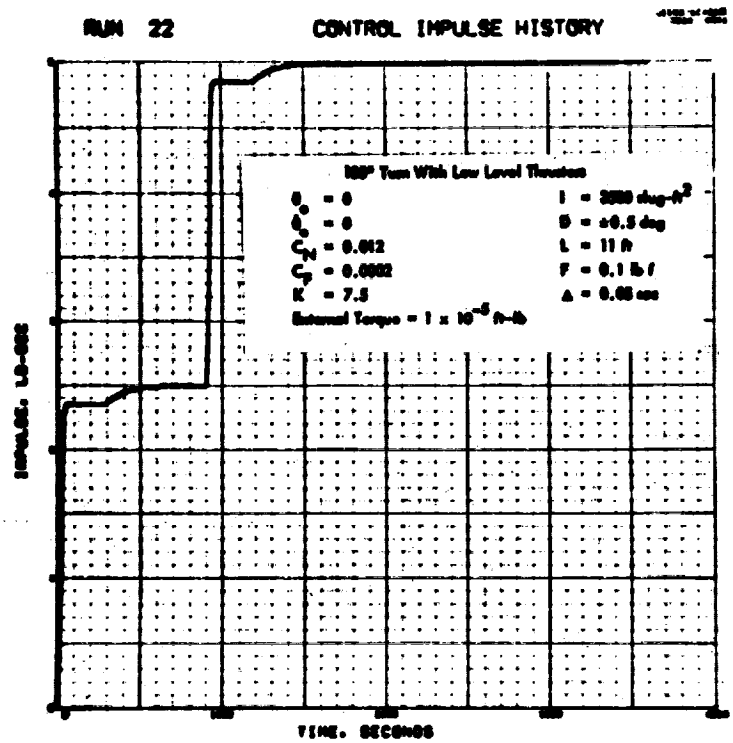


Fig. B.3d Control Impulse History – Run 22

(see Fig. B.3c) and the acceptable 4 degree peaks in spacecraft/gyro position difference (see Fig. B.3b).

Figure B.3d shows the characteristic control impulse expenditure history for such a maneuver; 2.5 lbf-sec to initiate the turn and 2.5 more to terminate it. The ideal expenditure is readily computed for this type of maneuver as:

$$Q_{Ideal} = \frac{2\pi I \dot{\Theta}_C}{90L} = 4.45 \text{ lbf-sec}$$

The indicated dynamic factor is therefore 5.00/4.45 = 1.123. This value can also be obtained directly by considering the rate overshoot levels previously mentioned.

A number of runs containing individual parameter variations to the Run 22 data bases were made to establish dynamic factor sensitivity to such changes. Results are summarized in Table B.1. Note the improvement associated with a smaller deadband, an affect that makes such a change prior to maneuvers advisable. The F and L changes correspond to raising and lowering loop gain, respectively. Since this type of change would normally be compensated by appropriate controller parameter adjustments (as would the change in Δ) these particular results are not indicative of the ultimate effect on dynamic factor.

The insensitivity of dynamic factor to flight variations in spacecraft moment of inertia is relatively important, however. Apparent sensitivity to the commanded turning rate is a matter of some concern since there was not sufficient time to establish the anticipated effectiveness of various controller compensation measures. Relatively high speed maneuvers will likely be performed with high level thrusters in any case.

B.5 SPACECRAFT MANEUVERS WITH HIGH LEVEL THRUSTERS

A number of runs were made with 2.0 lbf thrusters separated by 10 ft to confirm satisfactory performance of the selected controller. A brief search effort resulted

Table B-1

SPACECRAFT MANEUVER
PARAMETER VARIATION SUMMARY

Nominal Data

$I = 3500$ slug-ft^2	$L = 11$ ft	$F = 0.1$ lbf
$\Delta = 0.050$ sec	$D = \pm 0.5$ deg	$\dot{\theta}_C = 0.2$ deg/sec
$A = 1.0 \times 10^{-5}$ ft-lb	$B = 0$	
$C_N = 0.012$	$C_F = 0.0002$	$K = 7.5$

Run	Description	Control Impulse (lb-sec) Q @ 1 Hr	Percent Deviation From Nominal	Dynamic Factor
22	Nominal	5.00	–	1.12
23	$D = 0.25$ deg	4.77	-4.6%	1.07
24	$F = 0.2$ lbf	4.94	-1.2%	1.11
25	$L = 5.5$ ft	16.60	232.0%	1.87
26	$\Delta = 0.025$ sec	5.90	18.0%	1.33
27	$I = 3000$ slug-ft^2	4.29	-14.2%	1.12
28	$\dot{\theta}_C = 0.3$ deg/sec	10.29	105.8%	1.54
29	$K = 10.0$	5.16	3.2%	1.16
30	$C_N = 0.015$, $K = 6$	4.96	-0.8%	1.12
31	$C_F = 0.0004$	5.53	10.6%	1.24

in the following representative set of controller parameters used for the first two runs to be discussed: $C_N = 0.012$, $C_F = 0.004$, $K = 1.0$. All runs employed a pulse width of 0.025 sec.

Run 44 illustrates system behavior when subjected to a relatively large torque; the 3.0 ft-lb roll torque assumed to exist during the 5000 lbf main engine burn. A low value of spacecraft inertia appropriate to the Jupiter orbital configuration, 600 slug-ft^2, was used with a 0.25 deg deadband. Figures B.4a, B.4b, and B.4c show system response. In Fig. B.4a the steady state error of 0.335 deg required to generate the counteracting control torque pulse train is immediately evident. The amount by which this error exceeds the deadband (0.085 deg in Run 44) is essentially proportional to controller parameter K. This consideration dictates both a small deadband and a small value of K during main engine burns in order to maintain small steady state errors. The relatively high frequency dither in the pseudo steady state of $\Theta \cong 0.335$ may be seen in the phase plane of Fig. B.4b. A terminal rate excursion of ±0.025 deg/sec is observed; this correlates precisely with the angular velocity increment generated by a single pulse ($180\ FL\Delta/\pi I = 0.048$ deg/sec). Finally, the essentially linear build-up of control impulse shown in Fig. B.4c agrees precisely with that predicted from the time integral of external torque,

$$Q = \frac{2}{L}\int_0^t M_E\, dt = 2At/L = 180 \text{ lb-sec}$$

The high level thrusters and controller employed in Run 44 were used to execute the turning maneuver studied in Run 22. As in the nominal low level thruster case, spacecraft moment of inertia was set at 3500 slug-ft^2. A 0.25 deg deadband was used, however, to raise the limit cycle frequency. As can be observed in Figs. B.5a and B.5b, the controller tracks the command very accurately, with all error amplitudes less than 0.32 deg. The phase plane, Fig. B.5c, shows small rate overshoots and a terminal single pulse limit cycle featuring rates of ±0.004 deg/sec (an initial rate condition of 0.004 deg/sec was selected). The ideal control impulse expenditure for

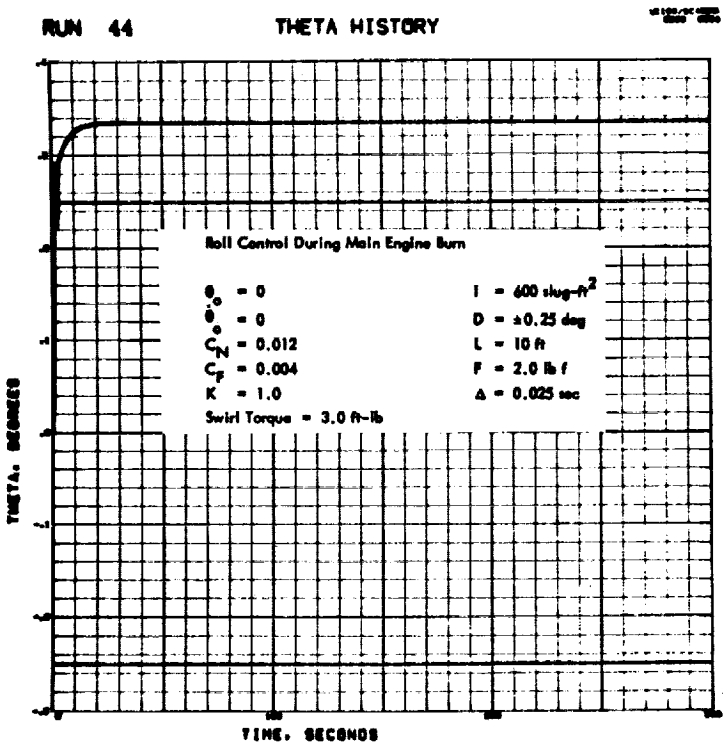


Fig. B.4a Theta Turn Maneuver History – Run 44

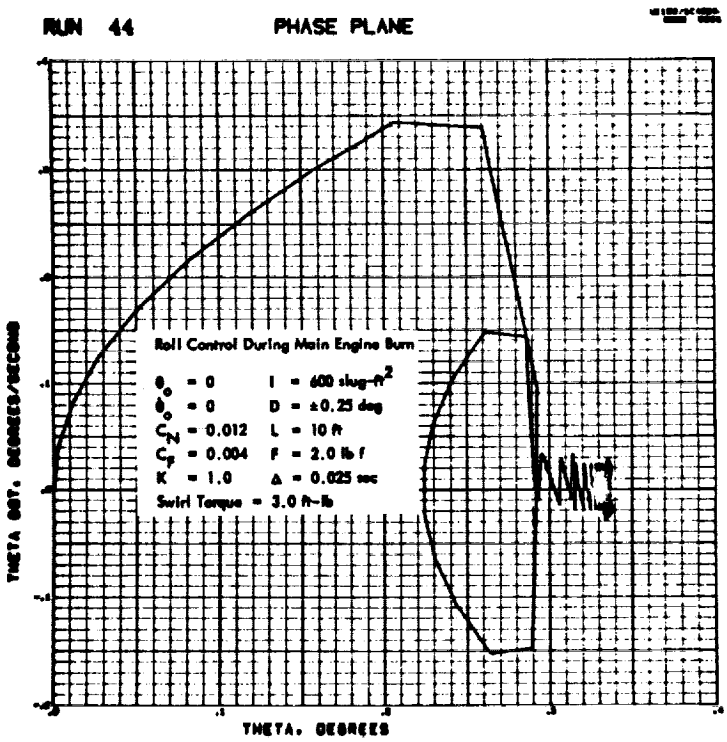


Fig. B.4b Phase Plane – Run 44

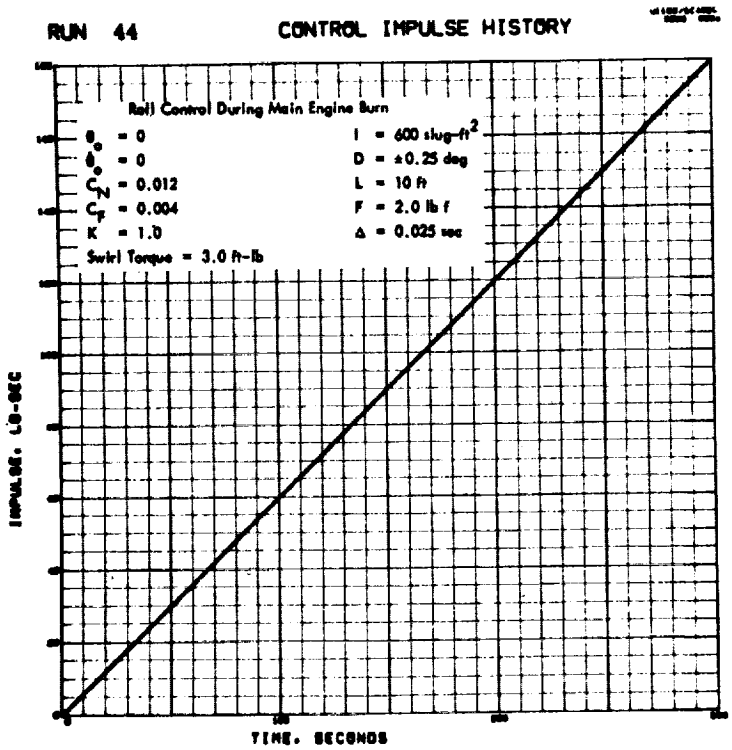


Fig. B.4c Control Impulse History – Run 44

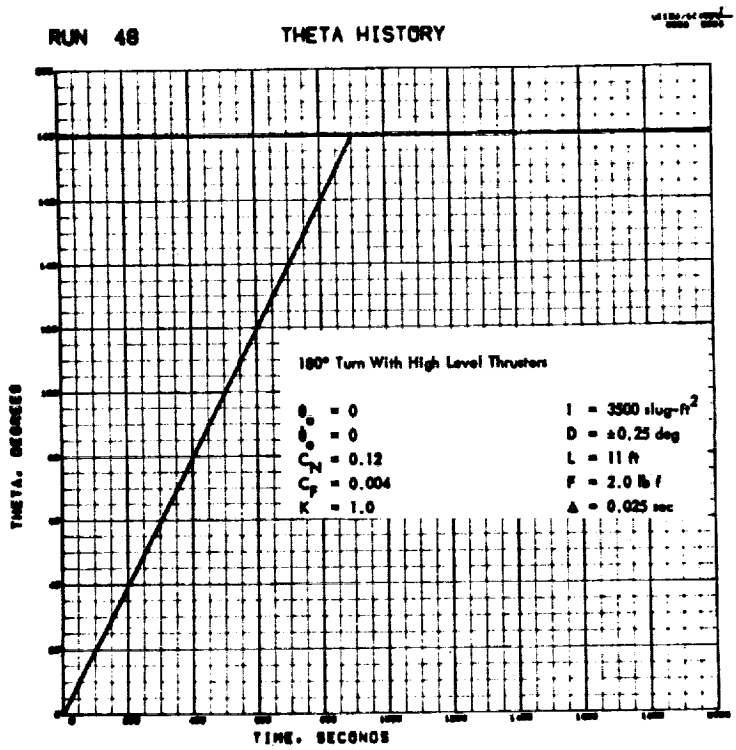


Fig. B.5a Theta Turn Maneuver History – Run 48

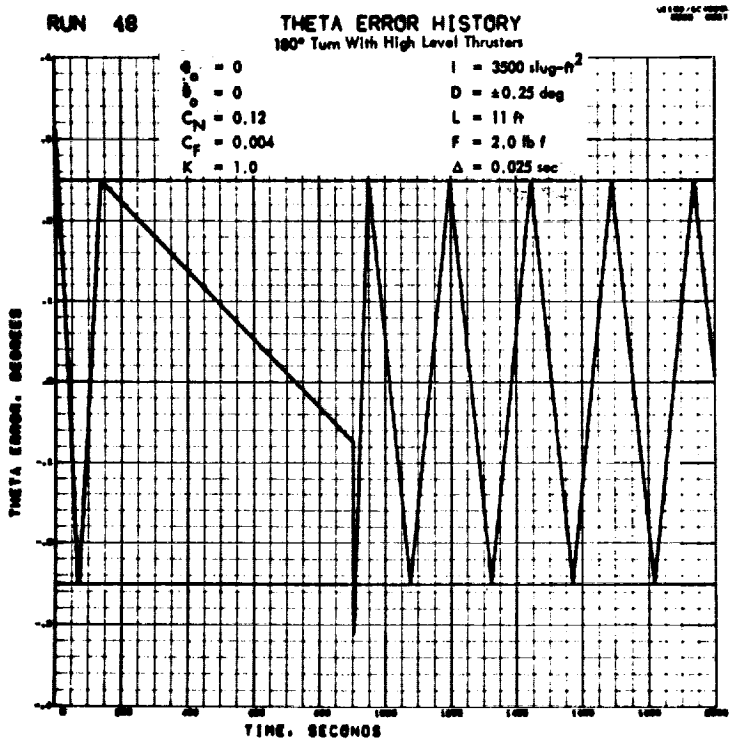


Fig. B.5b Theta Turn Error History – Run 48

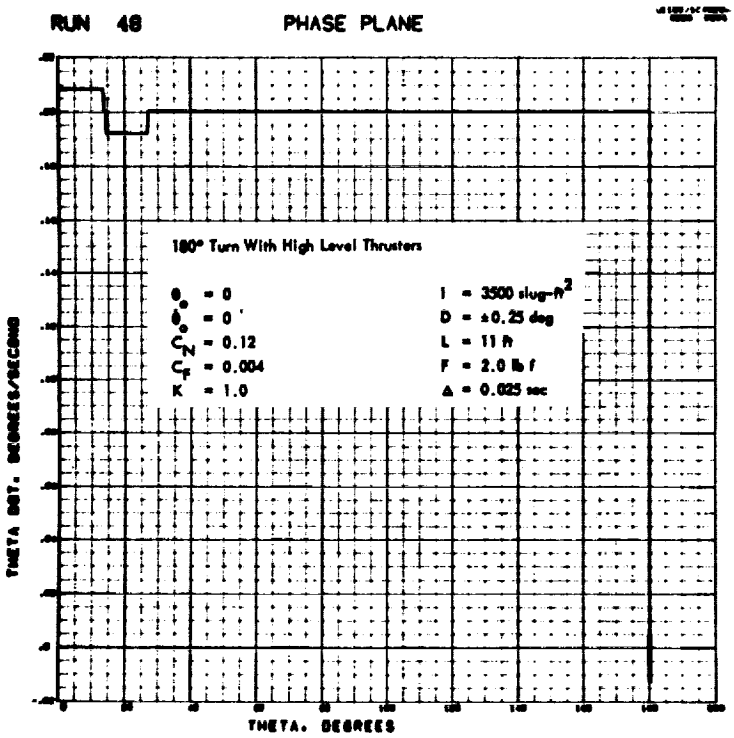


Fig. B.5c Phase Plane – Run 48

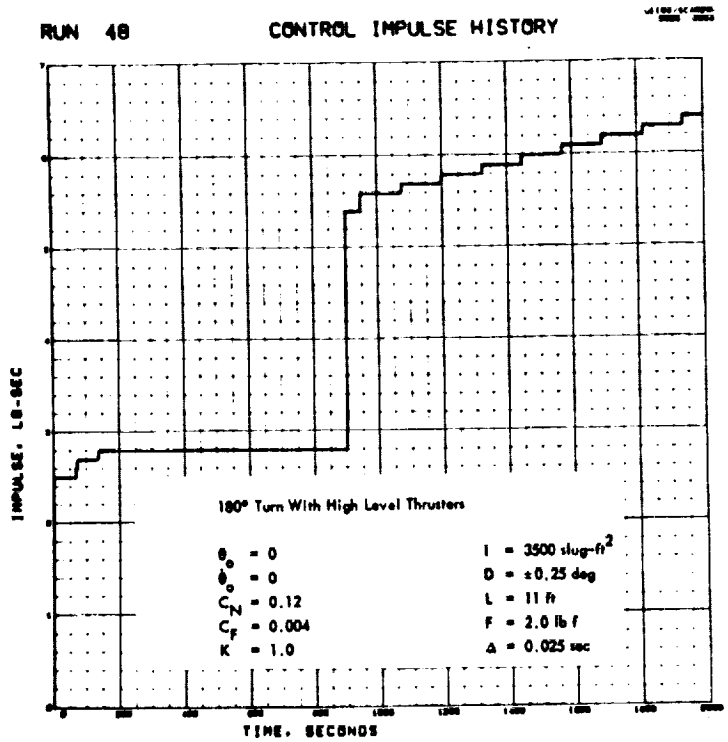


Fig. B.5d Control Impulse History – 48

the maneuver is 4.9 lbf-sec. Application of a 1.25 dynamic factor gives 6.1 lbf-sec. Figure B.5d indicates that the maneuver can be completed by the high level thrusters with this expenditure if prolonged single pulse limit cycling is not permitted. The plot emphasizes the expensive nature of such limit cycling and the necessity of minimizing the duration of same.

Results from a third type of run are included for illustration purposes only – the relatively large propellant expenditure for the maneuver precludes its use as a practical operational capture mode. The spacecraft/controller combination is identical to that used in Run 48 above except that a 0.5 deg deadband was used and K = 10 instead of 1. An initial error condition of Θ = 10 deg, $\dot{\Theta}$ = 0.004 deg/sec is the only system disturbance.

Figures B.6a, B.6b, and B.6c display the capture transient in different ways. Figure B.6a shows that the terminal single pulse limit cycle begins at 220 seconds. The phase plane of Fig. B.6b shows the degree of system damping in regions far from the deadband. The observed peak rate of - 1.8 deg/sec generated by the large thrusters explains the large control impulse expenditure seen in Fig. B.6c (corresponds to approximately 0.89 lbs of nitrogen gas). The sum of velocity increment magnitudes imparted to the spacecraft during the first four peaks may be measured from Fig. B.6b as 4.65 deg/sec. Using the relation $Q = \pi I(4.65)/90L$ we find $Q = 56.9$ lbf-sec to check the Fig. B.6c result.

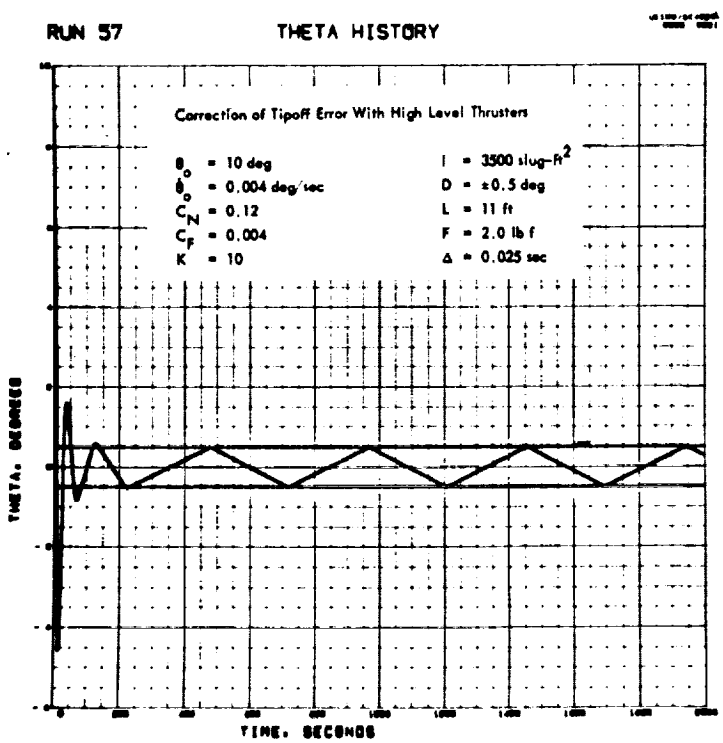


Fig. B.6a Theta Tipoff Error History — Run 57

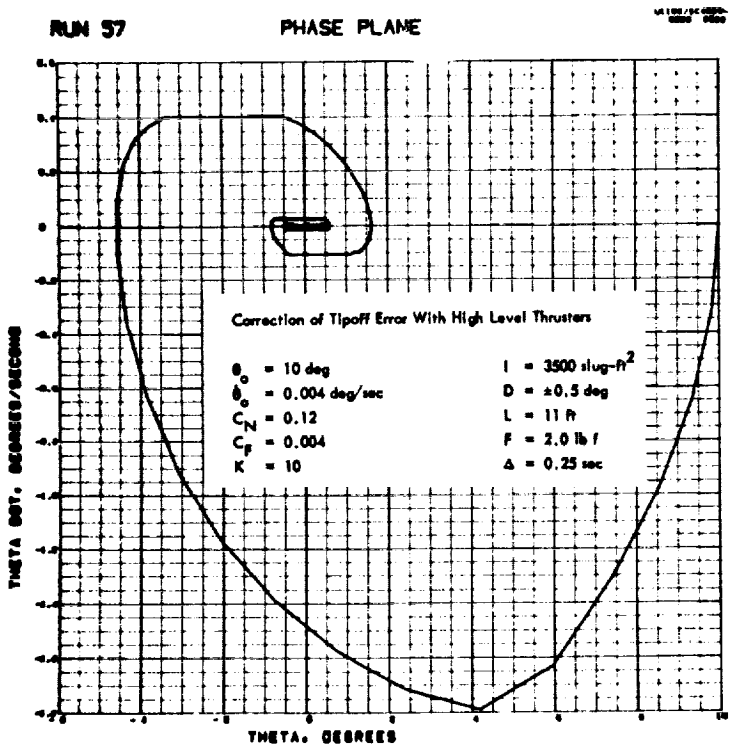


Fig. B.6b Phase Plane – Run 57

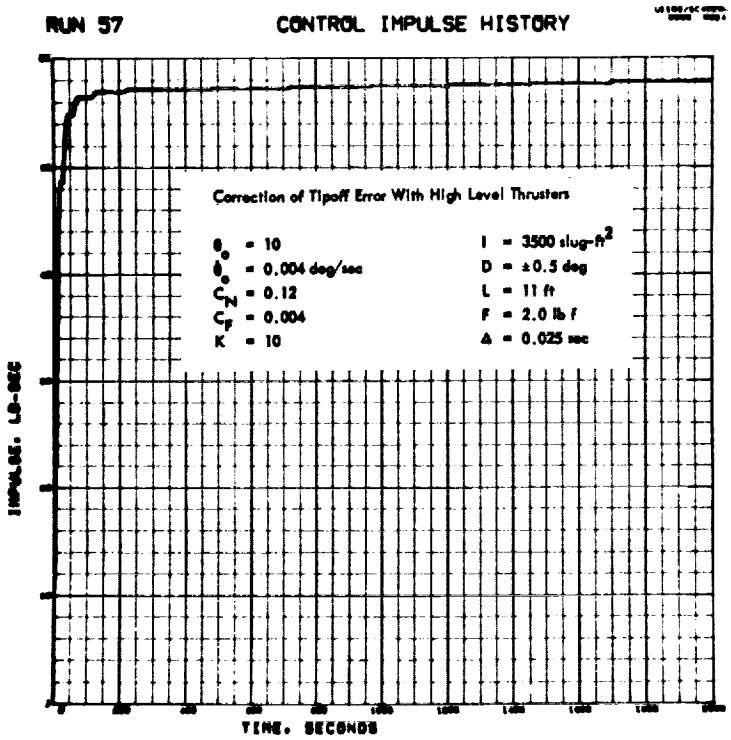


Fig. B.6c Control Impulse History – Run 57

A potential problem area in controller parameter selection was brought to light by runs similar to Run 57. Parameter value sets which give small "steady state" errors for large external torque disturbances tend to give poor damping in regions far from the deadband and conversely. This is the reason for the use of $K = 10$ in Run 57 rather than $K = 1$. The latter value produced quite light damping for the capture run, leading to a control impulse expenditure of 438 lbf-sec. The implications of these results, not critical since the controller can be made mode adaptive and in view of the impracticality of capture modes without rate limiting, were not explored further in this study.

B.6 CONCLUSIONS FOR THE PRIMARY STUDY TASK

As a result of this effort the following general ground rules were established for the overall study task.

A dynamic factor of 1.25 is appropriate for relatively well defined maneuvers such as have been simulated. This applies equally to maneuvers performed with the low or high thruster levels although the duration of limit cycling with high thruster levels must be minimized.

A dynamic factor of 1.5 is estimated for maneuvers that involve sensor switching transients (e.g., acquisitions).

A dynamic factor of 2.0 is estimated for relatively ill defined maneuvers such as the removal of tipoff rates.

No serious problems were uncovered with the digital pseudo rate controller. It can be adapted to high and low thruster force levels and to all spacecraft configurations by simple parameter value alterations.

All maneuvers must be based on carefully specified and limited spacecraft angular rates. Controller parameters must be selected to give good damping, thus

avoiding expensive overshoot/undershoot transients, particularly with high level thruster forces.

Definition of Symbols

Δ	=	Spacecraft attitude control deadband ($\pm$deg)
L	=	thruster separation (ft)
F	=	thruster force
X	=	controller feedback state variable
Θ_E	=	Θ error (deg)
Θ_C	=	controller Θ command (deg)
C_N	=	analog "on" time
C_F	=	analog "off" time
K	=	gain term for the "on" state
ZOH	=	zero order hold function
N	=	torque command signal, either 0, +1, or -1
I	=	moment of inertia (slug-ft^2)
S	=	Laplace operator
B	=	function of spacecraft orientation (only zero used for this analysis)
M_E	=	external torque = $A + B\Theta$
M_C	=	control torque
A	=	external torque (ft-lb)

Appendix C
COMPUTER OUTPUT OF THERMODYNAMIC OPTIMIZATION

This appendix constitutes the tabulated output data of the Thermodynamic Optimization Program (TOP). All systems are grouped first by mission and then by propellant. For each mission and propellant combination there is a design summary for each optimized propellant followed by the weight summary of the propulsion module. All weights are given in pounds.

Table C-1a

F_2/H_2 MARS ORBITER – BASELINE OPTIMIZATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =       .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20000+01 (  %  )
        VENT PRESSURE                         =  .82280+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .16909+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                  =   06.881   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .15430+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2 LF2(H)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21482+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .43700+02 ( PSI )
        PROPELLANT TANK                       =  .66115+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .33995+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16444+01 (FEET )      RESIDUAL                          =  .30692+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .63278+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .97121-00 ( LBS )      INSULATION                        =  .32579+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .35554-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .17498+01 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .18832-00 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .28790-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .22220+01
   NONOPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        PROPELLANT TANK VOLUME                =  .85920+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .96295+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .30727+01 (FEET )      INSULATION                        =  .41527+02 ( LBS )
        INITIAL PROPELLANT LOAD               =  .31936+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .10817+01 ( LBS )
        PROPELLANT TANK                       =  .10924+03 ( LBS )      RESIDUAL                          =  .51154+01 ( LBS )
   TOTALS
        PAYLOAD                               =  .50527+04 ( LBS )      LAUNCH WEIGHT                     =  .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .28527+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70199+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .16375+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .60253-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .59679-01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .35479+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .55537+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .26989-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .27387-01 ( LBS )
        OPERATING PRESSURE                    =  .23698+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .12033+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .66205+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .15948+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .76280+01 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .53431+01 ( LBS )
        OPERATING PRESSURE                    =  .24084+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .34553+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .33170+03 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .68019+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .14737+03 ( LBS )
```

Table C-1b

F_2/H_2 MARS ORBITER – BASELINE OPTIMIZATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      2.250 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME        = .14965+02 ( % )
        VENT PRESSURE                           = .18770+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .96000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 = .31936+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                          CALCULATION TIME                = 06.497   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .50526+02 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .96331+02 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .90702+02 ( PSI )
        PROPELLANT TANK                         = .10924+03 ( LBS )     PROPELLANT TANK SURFACE AREA    = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .85930+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .30727+01 (FEET )     RESIDUAL                        = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .39539+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .00000    ( LBS )     INSULATION                      = .41527+02 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # =     4 - LH2(F)PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .35558-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .17498+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .18832-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .22220+01
   NONOPTIMUM TANK(S)   =  2  LF2(H)TANK(S)
        PROPELLANT TANK VOLUME                  = .18638+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .33995+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16448+01 (FEET )     INSULATION                      = .32579+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .16909+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12546-00 ( LBS )
        PROPELLANT TANK                         = .66115+02 ( LBS )     RESIDUAL                        = .30692+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .50550+04 ( LBS )     LAUNCH WEIGHT                   = .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .29904+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70231+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .27291+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.30976-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.69741-01 ( LBS )
        OPERATING PRESSURE                      = .20453+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .35479+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .92562-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.32754-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .36378-01 ( LBS )
        OPERATING PRESSURE                      = .81899+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .12033+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .26579+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .33471-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .28825+02 ( LBS )
        OPERATING PRESSURE                      = .88274+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .34553+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .33484+03 (FT/SC)     WEIGHT OF PROPELLANT USED       = .11442+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .90702+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .14875+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                    08 MAY 69
      PROPELLANTS :        LH2(F)+LF2(H)
      USABLE WEIGHT          .36217+04
    STRUCTURE                                                          .25070+03
      BASE STRUCTURE                                     .16678+03
      TANK SUPPORTS                                      .50919+02
      ATTACHMENTS                                        .15000+02
      BULKHEAD INSULATION (1**)                          .18000+02
    PROPELLANT FEED ASSEMBLY                                           .41873+03
      TANKS                                              .24147+03
      VALVES+FILTERS+PLUMBING+ULLAGING                   .37000+02
      INSULATION (FIXED AND VARIABLE)                    .63443+02
      METEOROID BUMPER                                   .77214+02
    PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .26950+02
    ENGINE SYSTEM                                                      .98000+02
    INERT SUB-TOTAL                                                    .79438+03
    CONTINGENCY 10%                                                    .75438+02
    RESIDUALS                                                          .11669+03
      PROPELLANT                                         .66500+02
      VAPOR                                              .50864+02
      HE GAS                                             .22220+01
    PERFORMANCE RESERVE (1% ΔV)                                        .29904+02
  IMPULSE PROPELLANTS                                                              .36217+04
  PROPULSION MODULE WEIGHT                                                         .46450+04
  PAYLOAD WEIGHT                                                                   .50550+04
```

Table C-2a

FLOX/CH_4 MARS ORBITER-BASELINE OPTIMIZATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =       .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20000+01 (  %  )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .50000+02 (  %  )
        VENT PRESSURE                         =  .79500+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .11250+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .17186+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                  =   01.844   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .15860+03 (DEG-R)    COUNTER FOR PENALTIES             =     0

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21137+03 ( PSI )    MAX PRESSURE FOR THIS MISSION     =  .64583+02 ( PSI )
        PROPELLANT TANK                       =  .66708+02 ( LBS )    PROPELLANT TANK SURFACE AREA      =  .35129+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .19570+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16720+01 (FEET )    RESIDUAL                          =  .27216+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .74751+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .11059+01 ( LBS )    INSULATION                        =  .33665+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .42906-00 (FEET )    PRESSURANT SPHERE WEIGHT          =  .35983+01 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .33085-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .19093-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .16046+01
   NONOPTIMUM TANK(S)    =  2  CH4(F)TANK(S)
        PROPELLANT TANK VOLUME                =  .13068+02 (FT-3 )    PROPELLANT TANK SURFACE AREA      =  .26830+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .14612+01 (FEET )    INSULATION                        =  .25712+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .32711+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .65729-01 ( LBS )
        PROPELLANT TANK                       =  .62365+02 ( LBS )
        TOTAL USED                            =  .00000    ( LBS )    RESIDUAL                          =  .51840+01 ( LBS )
   TOTALS
        PAYLOAD                               =  .47881+04 ( LBS )    LAUNCH WEIGHT                     =  .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY       =  .30485+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70196+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .17186+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .62046-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .61462-01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .40918+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .58266+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .30727-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .31562-01 ( LBS )
        OPERATING PRESSURE                    =  .64583+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .13873+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .16219+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .73365+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .62286+01 ( LBS )
        OPERATING PRESSURE                    =  .64583+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .38618+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .33140+03 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .66458+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .43559+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .15823+03 ( LBS )
```

Table C-2b

FLOX/CH_4 MARS ORBITER-BASELINE OPTIMIZATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =       .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME      = .20000+01 (  %  )
        ULLAGE PERCENTAGE OF TANK VOLUME      = .11250+03 (  %  )
        VENT PRESSURE                         = .67310+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .50000-00 ( LBS )
        MAXIMUM TANK PRESSURE                 = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               = .32710+03 ( LBS )

MISSION PARTICULARS
        # BURN MISSION                                          CALCULATION TIME                  =  01.532   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .19737+03 (DEG-R)  COUNTER FOR PENALTIES            =    0

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .24181+03 ( PSI )  MAX PRESSURE FOR THIS MISSION    = .31269+02 ( PSI )
        PROPELLANT TANK                       = .62365+02 ( LBS )  PROPELLANT TANK SURFACE AREA     = .26830+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .13067+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .14613+01 (FEET )  RESIDUAL                         = .51940+01 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )  CALCULATED RESIDUAL PROPELLANT VAPOR= .11616+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .43078-00 ( LBS )  INSULATION                      = .25712+01 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # =    17 -  HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .42904-00 (FEET )  PRESSURANT SPHERE WEIGHT         = .35983+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .33025-00 (FT-3 )  PRESSURANT REMAINING IN PRESS SPHR = .69261-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .36107+01
   NONOPTIMUM TANK(S)   =  2  FLOX TANK(S)
        PROPELLANT TANK VOLUME                = .13579+02 (FT-3 )  PROPELLANT TANK SURFACE AREA     = .35128+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16720+01 (FEET )  INSULATION                       = .33665+01 ( LBS )
        INITIAL PROPELLANT LOAD               = .17107+04 ( LBS )  CALCULATED INITIAL PROPELLANT VAPOR = .12444+00 ( LBS )
        PROPELLANT TANK                       = .66708+02 ( LBS )
        TOTAL USED                            = .00000    ( LBS )  RESIDUAL                         = .27216+02 ( LBS )
   TOTALS
        PAYLOAD                               = .47882+04 ( LBS )  LAUNCH WEIGHT                    = .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )  1 % PERFORMANCE CONTINGENCY      = .30429+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73195+04
   BURN #   1
        DELTA VELOCITY OBTAINED               = .50500+02 (FT/SC)  WEIGHT OF PROPELLANT USED        = .32734+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .90825-02 (DEG-R)  WEIGHT OF PROPELLANT VAPORIZED AFTER= .13946-01 ( LBS )
        OPERATING PRESSURE                    = .20007+02 ( PSI )  TOTAL WEIGHT OF MIXTURE USED     = .40918+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               = .17170+02 (FT/SC)  WEIGHT OF PROPELLANT USED        = .11098+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .32055-02 (DEG-R)  WEIGHT OF PROPELLANT VAPORIZED AFTER= .50791-02 ( LBS )
        OPERATING PRESSURE                    = .27437+02 ( PSI )  TOTAL WEIGHT OF MIXTURE USED     = .13873+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               = .66205+04 (FT/SC)  WEIGHT OF PROPELLANT USED        = .30894+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .56244+01 (DEG-R)  WEIGHT OF PROPELLANT VAPORIZED AFTER= .11046+01 ( LBS )
        OPERATING PRESSURE                    = .27437+02 ( PSI )  TOTAL WEIGHT OF MIXTURE USED     = .38618+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               = .33128+03 (FT/SC)  WEIGHT OF PROPELLANT USED        = .12654+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00000    (DEG-R)  WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .31269+02 ( PSI )  TOTAL WEIGHT OF MIXTURE USED     = .15818+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                    76MAR 69
        PROPELLANTS :        FLOX:CH4(F)
        USABLE WEIGHT        .40443+04
      STRUCTURE                                                        .17711+03
        BASE STRUCTURE                                   .94262+02
        TANK SUPPORTS                                    .49848+02
        ATTACHMENTS                                      .15000+02
        BULKHEAD INSULATION (1**)                        .18000+02
      PROPELLANT FEED ASSEMBLY                                         .38057+03
        TANKS                                            .25815+03
        VALVES*FILTERS*PLUMBING,ULLAGING                 .38500+02
        INSULATION (FIXED AND VARIABLE)                  .31875+02
        METEOROID BUMPER                                 .52045+02
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                         .27397+02
      ENGINE SYSTEM                                                    .98000+02
      INERT SUB-TOTAL                                                  .68307+03
      CONTINGENCY 10%                                                  .68307+02
      RESIDUALS                                                        .85747+02
        PROPELLANT                                       .64800+02
        VAPOR                                            .17336+02
        HE GAS                                           .36107+01
      PERFORMANCE RESERVE (1% AV)                                      .30485+02
    IMPULSE PROPELLANTS                                                            .40443+04
    PROPULSION MODULE WEIGHT                                                       .49119+04
    PAYLOAD WEIGHT                                                                 .47881+04
```

Table C-3a

OF_2/B_2H_6 MARS ORBITER – BASELINE OPTIMIZATION FOR OF_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .80000+01 (  %  )
        VENT PRESSURE                           =  .71860+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .39000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .24100+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .15514+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                  CALCULATION TIME                =   20.073   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .24028+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  OF2(C)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21098+03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .24062+03 ( PSI )
        PROPELLANT TANK                         =  .69270+02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .35245+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .19676+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16797+01 (FEET )    RESIDUAL                        =  .23325+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .85927+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .38315+01 ( LBS )    INSULATION                      =  .33779+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 =   HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .79378-00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .19354+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .17235+01 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .68540-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .21935+01
   NONOPTIMUM TANK(S)    =  2   B2H6 TANK(S)
        PROPELLANT TANK VOLUME                  =  .19322+02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .34821+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16696+01 (FEET )    INSULATION                      =  .33370+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .51712+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .42935-01 ( LBS )
        PROPELLANT TANK                         =  .66598+02 ( LBS )    RESIDUAL                        =  .77750+01 ( LBS )
   TOTALS
        PAYLOAD                                 =  .46062+04 ( LBS )    LAUNCH WEIGHT                   =  .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY     =  .30688+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70197+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .50503+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .15725+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .98591-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .60357-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .41933+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .53310+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .58689-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .37592-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .19216+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .14619+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .11996+02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .63158+01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .36983+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .33194+03 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .60900+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .16107+03 ( LBS )
```

Table C-3b

OF_2/B_2H_6 MARS ORBITER – BASELINE OPTIMIZATION FOR B_2H_6 SYSTEM

```
                    INTERMEDIATE OPTIMIZATION OUTPUT
                    ********************************

PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20000+01 (  %  )
        VENT PRESSURE                           = .58100+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .54300+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .10000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .51970+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                  CALCULATION TIME                = 03.050   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .28103+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  B2H6 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21256+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .15500+03 ( PSI )
        PROPELLANT TANK                         = .66974+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .34723+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .19290+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM      = .16623+01 (FEET )     RESIDUAL                        = .77750+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .60190-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .37879+01 ( LBS )     INSULATION                      = .33276+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .79378-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .19354+02 ( LBS )
        PRESSURANT SPHERE VOLUME                = .17235+01 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .12079-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .22512+01
   NONOPTIMUM TANK(S)    =  2  OF2(C)TANK(S)
        PROPELLANT TANK VOLUME                  = .19883+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .35489+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM      = .16805+01 (FEET )     INSULATION                      = .39015+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15520+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .11737+01 ( LBS )
        PROPELLANT TANK                         = .68812+02 ( LBS )     RESIDUAL                        = .23326+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .96091+04 ( LBS )     LAUNCH WEIGHT                   = .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 0 PERFORMANCE CONTINGENCY     = .30660+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70196+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .50503+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .52917+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .11427-01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .20958-01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .91933+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .17770+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .18501-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .30582-02 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .19216+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .98729+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .97191+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .66799-00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .38983+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .53136+03 (FT/SC)     WEIGHT OF PROPELLANT USED       = .20130+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16109+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                        13 MAY 69

        PROPELLANTS :        B2H6 ,OF2(C)
        USABLE WEIGHT        .90699+04
      STRUCTURE                                                          .17811+03
        BASE STRUCTURE                                     .95059+02
        TANK SUPPORTS                                      .60050+02
        ATTACHMENTS                                        .15000+02
        BULKHEAD INSULATION (1")                           .10000+02
      PROPELLANT FEED ASSEMBLY                                           .90576+03
        TANKS                                              .27062+03
        VALVES,FILTERS,PLUMBING,ULLAGING                   .30500+02
        INSULATION (FIXED AND VARIABLE)                    .33457+02
        METEOROID BUMPER                                   .63191+02
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .56909+02
      ENGINE SYSTEM                                                      .15380+03
      INERT SUB-TOTAL                                                    .79578+03
      CONTINGENCY 10%                                                    .79578+02
      RESIDUALS                                                          .10881+03
        PROPELLANT                                         .62208+02
        VAPOR                                              .20702+02
        HE GAS                                             .17189+02
      PERFORMANCE RESERVE (10.AV)                                        .30660+02
    IMPULSE PROPELLANTS                                                                 .98699+04
    PROPULSION MODULE WEIGHT                                                            .60709+04
    PAYLOAD WEIGHT                                                                      .96091+04
```

Table C-4a

N_2O_4/A-50 MARS ORBITER – BASELINE OPTIMIZATION FOR N_2O_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20000+01 (  %  )
        VENT PRESSURE                           =  .19700+04 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .21000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .17500+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .15070+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                  CALCULATION TIME                  =   87.440   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .97622+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  N2O4 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21985+03 ( PSI )    MAX PRESSURE FOR THIS MISSION     =  .15976+03 ( PSI )
        PROPELLANT TANK                         =  .65311+02 ( LBS )    PROPELLANT TANK SURFACE AREA      =  .32459+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .17389+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16072+01 (FEET )    RESIDUAL                          =  .29954+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .90842-03 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .20280+01 ( LBS )    INSULATION                        =  .31106+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT = *     17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .60062-00 (FEET )    PRESSURANT SPHERE WEIGHT          =  .14978+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .90757-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .72049-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .22799+01
   NONOPTIMUM TANK(S)   =  2   A-50 TANK(S)
        PROPELLANT TANK VOLUME                  =  .18279+02 (FT-3 )    PROPELLANT TANK SURFACE AREA      =  .33576+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16396+01 (FEET )    INSULATION                        =  .32177+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .99319+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .33006-01 ( LBS )
        PROPELLANT TANK                         =  .65896+02 ( LBS )    RESIDUAL                          =  .15596+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .37972+04 ( LBS )    LAUNCH WEIGHT                     =  .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1'S PERFORMANCE CONTINGENCY       =  .33622+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70196+04
   BURN =   1
        DELTA VELOCITY OBTAINED                 =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .16097+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .35095-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .93792-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .52316+02 ( LBS )

   BURN =   2
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .69533+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.65308-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .84375-02 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .17723+02 ( LBS )

   BURN =   3
        DELTA VELOCITY OBTAINED                 =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .19333+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .15590+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .19199+01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .96503+04 ( LBS )

   BURN =   4
        DELTA VELOCITY OBTAINED                 =  .33191+03 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .63390+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .17335+03 ( LBS )
```

Table C-4b

N_2O_4/A-50 MARS ORBITER – BASELINE OPTIMIZATION FOR A-50 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .625 (INCHES)
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20000+01 (  %  )
        VENT PRESSURE                           =  .16940+04 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .21000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .75000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .99277+03 ( LBS )


MISSION PARTICULARS
        4 BURN MISSION                                                  CALCULATION TIME                    =   12.560   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .50307+03 (DEG-R)


WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  [A-50] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21619+03 ( PSI )    MAX PRESSURE FOR THIS MISSION       =  .15500+03 ( PSI )
        PROPELLANT TANK                         =  .65471+02 ( LBS )    PROPELLANT TANK SURFACE AREA        =  .33567+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .18287+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .14399+01 (FEET )    RESIDUAL                            =  .15596+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .15132-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .23605+01 ( LBS )    INSULATION                          =  .90210+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    1F -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .59693-00 (FEET )    PRESSURANT SPHERE WEIGHT            =  .19629+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .88697-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR  =  .39506-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .22220+01
   N2O4OPTIMUM TANK(S)    =  2  N2O4 TANK(S)
        PROPELLANT TANK VOLUME                  =  .17395+02 (FT-3 )    PROPELLANT TANK SURFACE AREA        =  .32960+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .14072+01 (FEET )    INSULATION                          =  .31107+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .15071+04 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .82843-01 ( LBS )
        PROPELLANT TANK                         =  .65312+02 ( LBS )    RESIDUAL                            =  .24954+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .37978+04 ( LBS )    LAUNCH WEIGHT                       =  .97000+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY         =  .33701+02 ( LBS )


SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .70199+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .53500+02 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .10061+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .75378-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .16294-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .52316+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .39083+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.83923-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .15987-02 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .17723+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .09582+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.77937-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .38332-00 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .96583+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .33172+03 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .33360+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .17381+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                                  13 MAY 69
        PROPELLANTS :          [A-50] , N2O4
        USABLE WEIGHT          .96680+04
      STRUCTURE                                                                  .21030+03
        BASE STRUCTURE                                          .11110+03
        TANK SUPPORTS                                           .66115+02
        ATTACHMENTS                                             .15000+02
        BULKHEAD INSULATION (1%)                                .18000+02
      PROPELLANT FEED ASSEMBLY                                                   .38195+03
        TANKS                                                   .26241+03
        VALVES,FILTERS,PLUMBING,BLEEDING                        .28000+02
        INSULATION (FIXED AND VARIABLE)                         .39269+02
        METEOROID BUMPER                                        .56763+02
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .70717+02
      ENGINE SYSTEM                                                              .15100+03
      INERT SUB-TOTAL                                                            .82197+03
      CONTINGENCY 10%                                                            .82197+02
      RESIDUALS                                                                  .92609+02
        PROPELLANT                                              .61188+02
        VAPOR                                                   .21716+01
        HE GAS                                                  .93329+01
      PERFORMANCE RESERVE (15 AV)                                                .33701+02
    IMPULSE PROPELLANTS                                                                   .96680+04
    PROPULSION MODULE WEIGHT                                                              .56960+04
    PAYLOAD WEIGHT                                                                        .37978+04
```

Table C-5a

F_2/H_2 VENUS ORBITER – COMPUTATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        LAUNCH WEIGHT                           =  .93970+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .23782+01 (  %  )
        VENT PRESSURE                           =  .82280+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .12000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .16893+04 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                                  CALCULATION TIME                =   87.995   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .16989+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2 LF2 THITANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21482+03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .21305+03 ( PSI )
        PROPELLANT TANK                         =  .44116+02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .33996+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16494+01 (FEET )    RESIDUAL                        =  .30692+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .20317+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .11125+01 ( LBS )    INSULATION                      =  .32580+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .37784+00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .20996+01 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .22594+00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .87498+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .72270+01
        [illegible]
        [illegible]
   NONOPTIMUM TANK(S)    =  1 LH2(F)TANK(S)
        PROPELLANT TANK VOLUME                  =  .85933+02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .96295+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .30727+01 (FEET )    INSULATION                      =  .55370+02 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .29096+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .16120+01 ( LBS )
        PROPELLANT TANK                         =  .10924+03 ( LBS )    RESIDUAL                        =  .51154+01 ( LBS )
   TOTALS
        PAYLOAD                                 =  .47797+04 ( LBS )    LAUNCH WEIGHT                   =  .93970+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY     =  .28098+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73398+04
   BURN =   1
        DELTA VELOCITY OBTAINED                 =  .53500+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .15863+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .58975-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .58240-01 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .34371+02 ( LBS )

   BURN =   2
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .53803+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .32692-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .33873-01 ( LBS )
        OPERATING PRESSURE                      =  .29315+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .11657+02 ( LBS )

   BURN =   3
        DELTA VELOCITY OBTAINED                 =  .72721+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .16632+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .30967+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .36036+04 ( LBS )
```

Table C-5b

F_2/H_2 VENUS ORBITER – COMPUTATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =      3.000 (INCHS)
        LAUNCH WEIGHT                         = .93932+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      = .22375+02 (  %  )
        VENT PRESSURE                         = .18770+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 = .70000+02 ( PSI )
        INITIAL PROPELLANT LOAD               = .29096+03 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                            CALCULATION TIME                  =   03.484   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .50906+02 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .96331+02 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .93994+02 ( PSI )
        PROPELLANT TANK                       = .10924+03 ( LBS )     PROPELLANT TANK SURFACE AREA      = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .85930+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .30727+01 (FEET )     RESIDUAL                          = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .11903+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .00000    ( LBS )     INSULATION                        = .55370+02 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # -     4 - LH2(F)PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .37786-00 (FEET )     PRESSURANT SPHERE WEIGHT          = .20998+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .22598-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .26664+01
   NONOPTIMUM TANK(S)   =  2  LF2(M)TANK(S)
        PROPELLANT TANK VOLUME                = .18630+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .33496+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16448+01 (FEET )     INSULATION                        = .32580+01 ( LBS )
        INITIAL PROPELLANT LOAD               = .16840+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .15026-00 ( LBS )
        PROPELLANT TANK                       = .66116+02 ( LBS )     RESIDUAL                          = .30692+02 ( LBS )
   TOTALS
        PAYLOAD                               = .47792+04 ( LBS )     LAUNCH WEIGHT                     = .93932+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY       = .28400+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73406+04
   BURN #  1
        DELTA VELOCITY OBTAINED               = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .26428+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.48198-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.10472+00 ( LBS )
        OPERATING PRESSURE                    = .20422+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .34357+02 ( LBS )
   BURN #  2
        DELTA VELOCITY OBTAINED               = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .89635-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.16556-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.90665-01 ( LBS )
        OPERATING PRESSURE                    = .83727+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .11653+02 ( LBS )
   BURN #  3
        DELTA VELOCITY OBTAINED               = .72730+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .27711+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .93994+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .36024+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                                  14 MAY 69
   PROPELLANTS :        LH2(F)+LF2(M)
   USABLE WEIGHT        .36200+04
 STRUCTURE                                                                  .25070+03
   BASE STRUCTURE                                                .16678+03
   TANK SUPPORTS                                                 .50919+02
   ATTACHMENTS                                                   .15000+02
   BULKHEAD INSULATION (1"")                                     .18000+02
 PROPELLANT FEED ASSEMBLY                                                   .42765+03
   TANKS                                                         .24147+03
   VALVES+FILTERS+PLUMBING+ULLAGING                              .37000+02
   INSULATION (FIXED AND VARIABLE)                               .76000+02
   METEOROID BUMPER                                              .72000+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .27300+02
 ENGINE SYSTEM                                                              .90000+02
 WEIGHT SUB-TOTAL                                                           .80365+03
 CONTINGENCY 10%                                                            .80365+02
 RESIDUALS                                                                  .81883+02
   PROPELLANT                                                    .66500+02
   VAPOR                                                         .12917+02
   HE GAS                                                        .26664+01
 PERFORMANCE RESERVE (1% dV)                                                .28400+02
IMPULSE PROPELLANTS                                                                        .36200+04
PROPULSION MODULE WEIGHT                                                                   .46140+04
PAYLOAD WEIGHT                                                                             .47792+04
```

Table C-6A

FLOX/CH_4 VENUS ORBITER – COMPUTATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        LAUNCH WEIGHT                         =  .93916+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .26066+01 (  "  )
        VENT PRESSURE                         =  .79500+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .13000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .17079+04 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                                CALCULATION TIME                  =   13.091   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .16731+03 (DEG-R)    COUNTER FOR PENALTIES             =    0

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21133+03 ( PSI )    MAX PRESSURE FOR THIS MISSION     =  .21041+03 ( PSI )
        PROPELLANT TANK                       =  .66708+02 ( LBS )    PROPELLANT TANK SURFACE AREA      =  .35128+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .19578+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16720+01 (FEET )    RESIDUAL                          =  .27216+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .23952-01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .12410+01 ( LBS )    INSULATION                        =  .33665+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .45201-00 (FEET )    PRESSURANT SPHERE WEIGHT          =  .42073+01 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .36684-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .58956-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .16238+01
   NONOPTIMUM TANK(S)   =  2  CH4(F)TANK(S)
        PROPELLANT TANK VOLUME                =  .13070+02 (FT-3 )    PROPELLANT TANK SURFACE AREA      =  .26833+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .14613+01 (FEET )    INSULATION                        =  .25715+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .32532+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .73518-01 ( LBS )
        PROPELLANT TANK                       =  .62367+02 ( LBS )
        TOTAL USED                            =  .00000    ( LBS )    RESIDUAL                          =  .51840+01 ( LBS )
   TOTALS
        PAYLOAD                               =  .45054+04 ( LBS )    LAUNCH WEIGHT                     =  .93916+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY       =  .30021+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73397+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .16639+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .60768-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .59880-01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .39617+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .56413+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .35610-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .37119-01 ( LBS )
        OPERATING PRESSURE                    =  .17116+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .13432+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .72721+04 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .16858+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .21041+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .40138+04 ( LBS )
```

Table C-6b

FLOX/CH_4 VENUS ORBITER – COMPUTATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =      .500 (INCHES)
        LAUNCH WEIGHT                         = .99018+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      = .20019+01 (  %  )
        VENT PRESSURE                         = .47318+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .40000-00 ( LBS )
        MAXIMUM TANK PRESSURE                 = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               = .32578+03 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                    CALCULATION TIME                  =  00.559   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .21098+03 (DEG-R)   COUNTER FOR PENALTIES     =  0

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  CH4 FLTANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .24180+03 ( PSI )   MAX PRESSURE FOR THIS MISSION   = .48051+02 ( PSI )
        PROPELLANT TANK                       = .42347+02 ( LBS )   PROPELLANT TANK SURFACE AREA    = .26833+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .13070+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .19413+01 (FEET )   RESIDUAL                        = .51496+01 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )   CALCULATED RESIDUAL PROPELLANT VAPOR= .74935-01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .54074-00 ( LBS )   INSULATION                      = .26715+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 -  HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .45701-00 (FEET )   PRESSURANT SPHERE WEIGHT        = .42073+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .38484-00 (FT-3 )   PRESSURANT REMAINING IN PRESS SPHR = .39254+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .35182+01
   NONOPTIMUM TANK(S)    =  2   FLOX TANK(S)
        PROPELLANT TANK VOLUME                = .19578+02 (FT-3 )   PROPELLANT TANK SURFACE AREA    = .35128+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16720+01 (FEET )   INSULATION                      = .33665+01 ( LBS )
        INITIAL PROPELLANT LOAD               = .17099+04 ( LBS )   CALCULATED INITIAL PROPELLANT VAPOR = .15571-00 ( LBS )
        PROPELLANT TANK                       = .46706+02 ( LBS )
        TOTAL USED                            = .00000    ( LBS )   RESIDUAL                        = .27214+02 ( LBS )
   TOTALS
        PAYLOAD                               = .95110+04 ( LBS )   LAUNCH WEIGHT                   = .94018+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )   I S PERFORMANCE CONTINGENCY     = .30099+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .75398+04
   BURN #   1
        DELTA VELOCITY OBTAINED               = .50500+02 (FT/SC)   WEIGHT OF PROPELLANT USED       = .31728+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .88730-02 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER= .13578-01 ( LBS )
        OPERATING PRESSURE                    = .20000+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED    = .39660+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               = .17170+02 (FT/SC)   WEIGHT OF PROPELLANT USED       = .10757+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .41256-02 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER= .65003-02 ( LBS )
        OPERATING PRESSURE                    = .48051+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED    = .13446+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               = .72722+04 (FT/SC)   WEIGHT OF PROPELLANT USED       = .32144+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00006    (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .47404+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED    = .40182+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                          13 MAR 69
  PROPELLANTS :        FLOX +CH4(F)
  USABLE WEIGHT        .40268+04
STRUCTURE                                                                  .17720+03
  BASE STRUCTURE                                          .94300+02
  TANK SUPPORTS                                           .49900+02
  ATTACHMENTS                                             .15000+02
  BULKHEAD INSULATION (1**)                               .18000+02
PROPELLANT FEED ASSEMBLY                                                   .37810+03
  TANKS                                                   .25815+03
  VALVES+FILTERS+PLUMBING+ULLAGING                        .38500+02
  INSULATION (FIXED AND EXPENDABLE)                       .31876+02
  METEOROID BUMPER                                        .49569+02
PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .28615+02
ENGINE SYSTEM                                                              .49000+02
INERT SUB-TOTAL                                                            .68191+03
CONTINGENCY 10%                                                            .68191+02
RESIDUALS                                                                  .69164+02
  PROPELLANT                                              .64800+02
  VAPOR                                                   .14230-06
  HE GAS                                                  .42218+01
PERFORMANCE RESERVE (1% AV)                                                .30021+02
IMPULSE PROPELLANTS                                                                    .8838+04
PROPULSION MODULE WEIGHT                                                               .9861+04
PAYLOAD WEIGHT                                                                         .9505+04
```

Table C-7a

OF_2/B_2H_6 VENUS ORBITER – COMPUTATION FOR OF_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =       .500 (INCHS)
        LAUNCH WEIGHT                         =  .94220+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .84992+01 (  %  )
        VENT PRESSURE                         =  .71860+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .40000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .24100+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .15448+04 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                                CALCULATION TIME                  =   05.370   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .24669+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  OF2(C)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21098+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .22809+03 ( PSI )
        PROPELLANT TANK                       =  .69270+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .35296+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .19676+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16747+01 (FEET )      RESIDUAL                          =  .23325+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .82736+00 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .39908+01 ( LBS )      INSULATION                        =  .33777+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 =   HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .74055-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .19103+02 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .17011+01 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .91785-02 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .21109+01
   NONOPTIMUM TANK(S)    =  2   B2H6 TANK(S)
        PROPELLANT TANK VOLUME                =  .19240+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .34723+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16623+01 (FEET )      INSULATION                        =  .33276+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .51490+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .42778+01 ( LBS )
        PROPELLANT TANK                       =  .66496+02 ( LBS )      RESIDUAL                          =  .77750+01 ( LBS )
   TOTALS
        PAYLOAD                               =  .43516+04 ( LBS )      LAUNCH WEIGHT                     =  .94220+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .30492+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73402+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SQ)      WEIGHT OF PROPELLANT USED         =  .15279+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .93632-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .56982-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .40732+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SQ)      WEIGHT OF PROPELLANT USED         =  .51782+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .52643-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .33320-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .13809+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .72726+04 (FT/SQ)      WEIGHT OF PROPELLANT USED         =  .18241+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .40641+04 ( LBS )
```

Table C-7b

OF_2/B_2H_6 VENUS ORBITER – COMPUTATION FOR B_2H_6 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        LAUNCH WEIGHT                           =  .94220+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20011+01 (  %  )
        VENT PRESSURE                           =  .58100+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .36000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .10000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .51990+03 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                          CALCULATION TIME                    =   03.694  ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .30139+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  B2H6 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21254+03 ( PSI )    MAX PRESSURE FOR THIS MISSION       =  .15500+03 ( PSI )
        PROPELLANT TANK                         =  .66496+02 ( LBS )    PROPELLANT TANK SURFACE AREA        =  .34723+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .19240+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16623+01 (FEET )    RESIDUAL                            =  .77750+01 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .86489+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .35841+01 ( LBS )    INSULATION                          =  .33276+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     1)   =   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .74055-00 (FEET )    PRESSURANT SPHERE WEIGHT            =  .19103+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .17011+01 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR  =  .16910-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .23454+01
   NONOPTIMUM TANK(S)   =  2  OF2(C)TANK(S)
        PROPELLANT TANK VOLUME                  =  .19676+02 (FT-3 )    PROPELLANT TANK SURFACE AREA        =  .35246+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16747+01 (FEET )    INSULATION                          =  .35777+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .15948+04 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .11207+01 ( LBS )
        PROPELLANT TANK                         =  .49270+02 ( LBS )    RESIDUAL                            =  .23325+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .43504+04 ( LBS )    LAUNCH WEIGHT                       =  .94220+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1 S PERFORMANCE CONTINGENCY         =  .30484+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73407+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .50914+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .11032-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .14378-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .40731+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .17261+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .19875-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .35131-02 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .13809+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .72730+04 (FT/SC)    WEIGHT OF PROPELLANT USED           =  .60804+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED        =  .40643+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                                  22 MAY 69
      PROPELLANTS :        OF2(C), B2H6
      USABLE WEIGHT        .40682+04
   STRUCTURE                                                                  .17800+03
      BASE STRUCTURE                                          .95000+02
      TANK SUPPORTS                                           .50000+02
      ATTACHMENTS                                             .15000+02
      BULKHEAD INSULATION (1"*)                               .16000+02
   PROPELLANT FEED ASSEMBLY                                                   .40222+03
      TANKS                                                   .27153+03
      VALVES,FILTERS,PLUMBING,ULLAGING                        .36500+02
      INSULATION (FIXED AND VARIABLE)                         .33911+02
      METEOROID BUMPER                                        .58779+02
   PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .58906+02
   ENGINE SYSTEM                                                              .15300+03
   INERT SUB-TOTAL                                                            .79162+03
   CONTINGENCY 10%                                                            .79162+02
   RESIDUALS                                                                  .86986+02
      PROPELLANT                                              .62200+02
      VAPOR                                                   .18687+01
      HE GAS                                                  .16887+02
   PERFORMANCE RESERVE (1% AV)                                                .38472+02
 IMPULSE PROPELLANTS                                                                        .48082+04
 PROPULSION MODULE WEIGHT                                                                   .58704+04
 PAYLOAD WEIGHT                                                                             .43516+04
```

Table C-8a

N_2O_4/A-50 VENUS ORBITER – COMPUTATION FOR N_2O_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        LAUNCH WEIGHT                         =  .94312+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20435+01 (  %  )
        VENT PRESSURE                         =  .14700+04 ( PSI )
        INITIAL PRESSURANT LOAD               =  .17000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .17500+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .15083+04 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                                CALCULATION TIME                =   10.188   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .53092+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  N2O4 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21985+03 ( PSI )      MAX PRESSURE FOR THIS MISSION   =  .17435+03 ( PSI )
        PROPELLANT TANK                       =  .65311+02 ( LBS )      PROPELLANT TANK SURFACE AREA    =  .32459+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .17389+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16072+01 (FEET )      RESIDUAL                        =  .24954+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .14017-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .16076+01 ( LBS )      INSULATION                      =  .31106+01 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # -    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .55540-00 (FEET )      PRESSURANT SPHERE WEIGHT        =  .11843+02 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .71760-00 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .92406-01 ( LBS )
        PRESSURANT SPHERE [illegible]         =  [illegible]
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .22220+01
   NONOPTIMUM TANK(S)   =  2   A-50 TANK(S)
        PROPELLANT TANK VOLUME                =  .18287+02 (FT-3 )      PROPELLANT TANK SURFACE AREA    =  .33567+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16344+01 (FEET )      INSULATION                      =  .32168+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .94270+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .33096-01 ( LBS )
        PROPELLANT TANK                       =  .65891+02 ( LBS )      RESIDUAL                        =  .15596+02 ( LBS )
   TOTALS
        PAYLOAD                               =  .35550+04 ( LBS )      LAUNCH WEIGHT                   =  .94312+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY     =  .33190+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73401+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)      WEIGHT OF PROPELLANT USED       =  .15651+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .15717-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .39235-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED    =  .50866+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)      WEIGHT OF PROPELLANT USED       =  .53021+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .20294-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .12958-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED    =  .17232+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .72724+04 (FT/SC)      WEIGHT OF PROPELLANT USED       =  .14873+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED    =  .48338+04 ( LBS )
```

Table C-8b

N_2O_4/A-50 VENUS ORBITER – COMPUTATION FOR A-50 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        LAUNCH WEIGHT                           = .99308+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20042+01 ( % )
        VENT PRESSURE                           = .14960+04 ( PSI )
        INITIAL PRESSURANT LOAD                 = .17000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .75000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 = .99270+03 ( LBS )

MISSION PARTICULARS
        3 BURN MISSION                                          CALCULATION TIME                = 07.341   ( DEC )
        FINAL PROPELLANT TEMPERATURE            = .56041+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  A-50 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21619+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .15542+03 ( PSI )
        PROPELLANT TANK                         = .65491+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .33567+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .18247+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16344+01 (FEET )     RESIDUAL                        = .15894+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .39265-01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .14974+01 ( LBS )     INSULATION                      = .39168+01 ( LBS )
   PRESSURANT ANALYSIS. PRESSURANT # =    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .55540-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .11843+02 ( LBS )
        PRESSURANT SPHERE VOLUME                = .71762-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .23719-02 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .22220+01
   NONOPTIMUM TANK(S)    =  2   N2O4 TANK(S)
        PROPELLANT TANK VOLUME                  = .17389+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .32459+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16072+01 (FEET )     INSULATION                      = .31104+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15063+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .84704+01 ( LBS )
        PROPELLANT TANK                         = .65311+02 ( LBS )     RESIDUAL                        = .24954+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .35547+04 ( LBS )     LAUNCH WEIGHT                   = .99308+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .33482+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .73407+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .97814+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .42714-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .14958-01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .60864+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .33137+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.58923-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .19268-02 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .17231+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .72730+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .97940+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .74339+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                        17 MAY 69
   PROPELLANTS :      N2O4 , A-50
   USABLE WEIGHT      .48687+04
 STRUCTURE                                                                  .21831+03
   BASE STRUCTURE                                          .11119+03
   TANK SUPPORTS                                           .66118+02
   ATTACHMENTS                                             .15000+02
   BULKHEAD INSULATION (1'')                               .18000+02
 PROPELLANT FEED ASSEMBLY                                                   .37988+03
   TANKS                                                   .26240+03
   VALVES,FILTERS,PLUMBING,ULLAGING                        .26000+02
   INSULATION (FIXED AND VARIABLE)                         .32655+02
   METEOROID BUMPER                                        .52820+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .67571+02
 ENGINE SYSTEM                                                              .15100+03
 INERT SUB-TOTAL                                                            .80476+03
 CONTINGENCY 10%                                                            .80476+02
 RESIDUALS                                                                  .89664+02
   PROPELLANT                                            .81188+02
   VAPOR                                                 .40967-00
   HE GAS                                                .75548+01
 PERFORMANCE RESERVE (1% ΔV)                                                .33196+02
IMPULSE PROPELLANTS                                                                    .48687+04
PROPULSION MODULE WEIGHT                                                               .58762+04
PAYLOAD WEIGHT                                                                         .35550+04
```

Table C-9a

F_2/H_2 LUNAR CARGO – COMPUTATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        LAUNCH WEIGHT                         =  .79745+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20025+01 (  %  )
        VENT PRESSURE                         =  .82280+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .70000-00 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .16909+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                  =   06.376   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .15259+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  LF2(H)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21482+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .32706+02 ( PSI )
        PROPELLANT TANK                       =  .66116+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .33996+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16448+01 (FEET )      RESIDUAL                          =  .30692+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .26802+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .60778-00 ( LBS )      INSULATION                        =  .32580+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .31572-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .12249+01 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .13182-00 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .92223-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .22220+01
   NONOPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        PROPELLANT TANK VOLUME                =  .85930+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .96295+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .30727+01 (FEET )      INSULATION                        =  .18457+02 ( LBS )
        INITIAL PROPELLANT LOAD               =  .28762+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .16744+01 ( LBS )
        PROPELLANT TANK                       =  .10924+03 ( LBS )      RESIDUAL                          =  .51154+01 ( LBS )
   TOTALS
        PAYLOAD                               =  .34345+04 ( LBS )      LAUNCH WEIGHT                     =  .79745+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .26443+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91652+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .40400+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .10773+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .38967-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .38671-01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .23343+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .28280+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .75227+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .27313-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26992-01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .16299+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .32269+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .71139+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .43033-00 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .24883+01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .15414+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .58696+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .95865+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .20771+04 ( LBS )
```

Table C-9b

F_2/H_2 LUNAR CARGO – COMPUTATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =      1.000 (INCHS)
        LAUNCH WEIGHT                         = .79713+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      = .23333+02 (  %  )
        VENT PRESSURE                         = .18776+03 ( PSI )
        INITIAL PRESSURANT LOAD               = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 = .70000+02 ( PSI )
        INITIAL PROPELLANT LOAD               = .28762+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                          CALCULATION TIME                  =   03.524   ( SEC )
        FINAL PROPELLANT TEMPERATURE          = .38861+02 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  1  LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          = .96331+02 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .25290+02 ( PSI )
        PROPELLANT TANK                       = .10926+03 ( LBS )     PROPELLANT TANK SURFACE AREA      = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                = .85930+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .30727+01 (FEET )     RESIDUAL                          = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                     = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .79512+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         = .00000    ( LBS )     INSULATION                        = .18457+02 ( LBS )
   PRESSURANT ANALYSIS- PRESSURANT # -     4 - LH2(F)PRESSURANT SPHERE INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              = .31572-00 (FEET )     PRESSURANT SPHERE WEIGHT          = .12249+01 ( LBS )
        PRESSURANT SPHERE VOLUME              = .13182-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   = .15554+01
   NONOPTIMUM TANK(S)   =  2  LF2(H)TANK(S)
        PROPELLANT TANK VOLUME                = .18639+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .33996+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   = .16449+01 (FEET )     INSULATION                        = .32580+01 ( LBS )
        INITIAL PROPELLANT LOAD               = .16908+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12598-00 ( LBS )
        PROPELLANT TANK                       = .66116+02 ( LBS )     RESIDUAL                          = .30692+02 ( LBS )
   TOTALS
        PAYLOAD                               = .34331+04 ( LBS )     LAUNCH WEIGHT                     = .79713+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY       = .27202+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91679+04
   BURN #   1
        DELTA VELOCITY OBTAINED               = .40400+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .17949+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.60749-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.15202-00 ( LBS )
        OPERATING PRESSURE                    = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .23333+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               = .28280+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .12533+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.39496-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.94619-01 ( LBS )
        OPERATING PRESSURE                    = .20511+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .16293+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               = .32269+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .11852+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.12309+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .24910+01 ( LBS )
        OPERATING PRESSURE                    = .21606+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .18407+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               = .58722+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .15977+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                    = .25290+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .28770+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                          16 MAY 69
   PROPELLANTS :        LF2(H)+LH2(F)
   USABLE WEIGHT        .36316+04
STRUCTURE                                                                  .25070+03
   BASE STRUCTURE                                          .16678+03
   TANK SUPPORTS                                           .50919+02
   ATTACHMENTS                                             .15000+02
   BULKHEAD INSULATION (1**)                               .18000+02
PROPELLANT FEED ASSEMBLY                                                   .35130+03
   TANKS                                                   .24147+03
   VALVES+FITTINGS+PLUMBING+ULLAGING                       .37000+02
   INSULATION (FIXED AND VARIABLE)                         .39973+02
   METEOROID BUMPER                                        .32857+02
PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .26425+02
ENGINE SYSTEM                                                              .98000+02
INERT SUB-TOTAL                                                            .72643+03
CONTINGENCY 10%                                                            .72643+02
RESIDUALS                                                                  .81319+02
   PROPELLANT                                              .66500+02
   VAPOR                                                   .13264+02
   HE GAS                                                  .19559+01
PERFORMANCE RESERVE (1% ΔV)                                                .26423+02
IMPULSE PROPELLANTS                                                                    .36316+04
PROPULSION MODULE WEIGHT                                                               .45383+04
PAYLOAD WEIGHT                                                                         .34345+04
```

Table C-10a

FLOX/CH_4 LUNAR CARGO – COMPUTATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        LAUNCH WEIGHT                         =  .80499+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20039+01 (  %  )
        VENT PRESSURE                         =  .79500+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .70000-00 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .17187+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                =   06.098  ( SEC
        FINAL PROPELLANT TEMPERATURE          =  .15379+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21133+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .32635+02 ( PSI
        PROPELLANT TANK                       =  .66708+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .35128+02 (FT-2
        PROPELLANT TANK VOLUME                =  .19578+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16720+01 (FEET )      RESIDUAL                          =  .27216+02 ( LBS
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .27559+01 ( LBS
        PRESSURANT IN PROPELLANT TANK         =  .63189-00 ( LBS )      INSULATION                        =  .33665+01 ( LBS
   PRESSURANT ANALYSIS, PRESSURANT # =    17 =   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .37673-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .24358+01 ( LBS
        PRESSURANT SPHERE VOLUME              =  .22396-00 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .68169-01 ( LBS
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .17459+01
   NONOPTIMUM TANK(S)    =  2  CH4(F)TANK(S)
        PROPELLANT TANK VOLUME                =  .13070+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .26833+02 (FT-2
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .19613+01 (FEET )      INSULATION                        =  .25715+01 ( LBS
        INITIAL PROPELLANT LOAD               =  .32737+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .64895-01 ( LBS
        PROPELLANT TANK                       =  .62367+02 ( LBS )      RESIDUAL                          =  .51840+01 ( LBS
   TOTALS
        PAYLOAD                               =  .31707+04 ( LBS )      LAUNCH WEIGHT                     =  .80499+04 ( LBS
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .27907+02 ( LBS

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91698+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .40400+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .11919+02 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .91351-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .90281-01 ( LBS
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .27176+02 ( LBS

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .28280+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .79667+01 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .29087-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .28223-01 ( LBS
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .18968+02 ( LBS

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .32269+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .74168+03 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .47372-00 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26376+01 ( LBS
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .17659+04 ( LBS

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .58691+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .95499+03 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS
        OPERATING PRESSURE                    =  .20000+02 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .22738+04 ( LBS
```

Table C-10b

FLOX/CH_4 LUNAR CARGO – COMPUTATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      .500 (INCHS)
        LAUNCH WEIGHT                           = .80422+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .27770+01 ( % )
        VENT PRESSURE                           = .67310+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .40000-00 ( LBS )
        MAXIMUM TANK PRESSURE                   = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .32730+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                          CALCULATION TIME                = 06.536   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .20034+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2 [illegible]F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .25180+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .32773+02 ( PSI )
        PROPELLANT TANK                         = .62367+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .26833+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .13070+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .14615+01 (FEET )     RESIDUAL                        = .51880+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .63658-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .31189-00 ( LBS )     INSULATION                      = .25715+01 ( LBS )
   PRESSURANT ANALYSIS) PRESSURANT # -     17 -   HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .37673-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .24358+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .22396-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .88108-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .30552+01
   NONOPTIMUM TANK(S)   =  2   FLOX TANK(S)
        PROPELLANT TANK VOLUME                  = .19570+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .35128+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16720+01 (FEET )     INSULATION                      = .33665+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .17183+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12593-00 ( LBS )
        PROPELLANT TANK                         = .66700+02 ( LBS )     RESIDUAL                        = .27216+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .31694+04 ( LBS )     LAUNCH WEIGHT                   = .80422+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .29504+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91701+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .40400+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .21721+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .63057-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .92239-02 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .27151+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .28280+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .15161+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .44212-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .64397-02 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .18951+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .32269+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .14115+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .68590-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .58056-00 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .17643+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .58744+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .18187+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .22734+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  18 APR 69
   PROPELLANTS :          [FLOX] ,CH4(F)
   USABLE WEIGHT           .40579+04
 STRUCTURE                                                                .17720+03
   BASE STRUCTURE                                   .49300+02
   TANK SUPPORTS                                    .49900+02
   ATTACHMENTS                                      .15000+02
   BULKHEAD INSULATION (1***)                       .18000+02
 PROPELLANT FEED ASSEMBLY                                                 .55331+03
   TANKS                                            .25815+03
   VALVES,FILTERS,PLUMBING,ULLAGING                 .38500+02
   INSULATION (FIXED AND VARIABLE)                  .31874+02
   METEOROID BUMPER                                 .29789+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                 .25072+02
 ENGINE SYSTEM                                                            .48000+02
 INERT SUB-TOTAL                                                          .65356+03
 CONTINGENCY 10%                                                          .65354+02
 RESIDUALS                                                                .73886+02
   PROPELLANT                                       .69800+02
   VAPOR                                            .66990+01
   HE GAS                                           .29992+00
 PERFORMANCE RESERVE (1% ΔV)                                              .27907+02
IMPULSE PROPELLANTS                                                                .40579+04
PROPULSION MODULE WEIGHT                                                           .94787+04
PAYLOAD WEIGHT                                                                     .31707+04
```

Table C-11a

OF_2/B_2H_6 LUNAR CARGO – COMPUTATION FOR OF_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =       .500 (INCHS)
        LAUNCH WEIGHT                         =  .80289+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .86524+01 (  %  )
        VENT PRESSURE                         =  .71860+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .42000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .24100+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .15441+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                  =  11.779   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .23108+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  OF2(C)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21098+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .16634+03 ( PSI )
        PROPELLANT TANK                       =  .69270+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .35246+02 (FT-I
        PROPELLANT TANK VOLUME                =  .19676+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16747+01 (FEET )      RESIDUAL                          =  .23325+02 ( LBS
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .30271+01 ( LBS
        PRESSURANT IN PROPELLANT TANK         =  .41549+01 ( LBS )      INSULATION                        =  .33777+01 ( LBS
   PRESSURANT ANALYSIS, PRESSURANT # -    17 -  HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .73728-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .18852+02 ( LBS
        PRESSURANT SPHERE VOLUME              =  .16788+01 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .45119-01 ( LBS
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .19839+01
   NONOPTIMUM TANK(S)   =  2   B2H6 TANK(S)
        PROPELLANT TANK VOLUME                =  .19240+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .34723+02 (FT-I
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16623+01 (FEET )      INSULATION                        =  .33276+01 ( LBS
        INITIAL PROPELLANT LOAD               =  .51490+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .42778-01 ( LBS
        PROPELLANT TANK                       =  .66496+02 ( LBS )      RESIDUAL                          =  .77750+01 ( LBS
   TOTALS
        PAYLOAD                               =  .29945+04 ( LBS )      LAUNCH WEIGHT                     =  .80289+04 ( LBS
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .28143+02 ( LBS

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91654+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .40400+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .10417+02 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .63419-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .38685-01 ( LBS
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .27779+02 ( LBS

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .28280+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .72706+01 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .44689-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .27142-01 ( LBS
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .19388+02 ( LBS

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .32269+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .66775+03 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .68347-00 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .24047+01 ( LBS
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .17807+04 ( LBS

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .58698+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .85620+03 ( LBS
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .22832+04 ( LBS
```

Table C-11b

OF_2/B_2H_6 LUNAR CARGO – COMPUTATION FOR B_2H_6 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        LAUNCH WEIGHT                           =  .80300+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20048+01 (  %  )
        VENT PRESSURE                           =  .58100+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .33000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .10000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .51487+03 ( LBS )

MISSION PARTICULARS                                          CALCULATION TIME                =  04.649   ( SEC )
        4 BURN MISSION
        FINAL PROPELLANT TEMPERATURE            =  .32356+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  B2H6 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21256+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   =  .16577+03 ( PSI )
        PROPELLANT TANK                         =  .66996+02 ( LBS )     PROPELLANT TANK SURFACE AREA    =  .34723+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .19240+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16623+01 (FEET )     RESIDUAL                        =  .77750+01 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR=  .92875-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .32167+01 ( LBS )     INSULATION                      =  .  [illegible]+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT #  =    17   =   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .73728-00 (FEET )     PRESSURANT SPHERE WEIGHT        =  .1[illegible]52+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .16788+01 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR  =  .83306-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .25250+01
   NONOPTIMUM TANK(S)   =  2  OF2(C)TANK(S)
        PROPELLANT TANK VOLUME                  =  .19676+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    =  .35246+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16747+01 (FEET )     INSULATION                      =  .35777+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .15491+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR =  .11313+01 ( LBS )
        PROPELLANT TANK                         =  .69270+02 ( LBS )     RESIDUAL                        =  .23325+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .29945+04 ( LBS )     LAUNCH WEIGHT                   =  .80300+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     =  .24934+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91669+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .90900+02 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .34729+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .78583-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .13994-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .27783+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .28280+02 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .24239+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .54779-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .94928-02 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .19391+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .32269+04 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .22261+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .84019-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .81199-04 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .17809+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .56707+04 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .28596+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .22838+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  19 MAY 69
   PROPELLANTS :        OF2(C)+ B2H6
   USABLE WEIGHT        .40829+04
 STRUCTURE                                                                .17800+03
   BASE STRUCTURE                                          .95000+02
   TANK SUPPORTS                                           .50000+02
   ATTACHMENTS                                             .15000+02
   BULKHEAD INSULATION (1**)                               .18000+02
 PROPELLANT FEED ASSEMBLY                                                  .37143+03
   TANKS                                                   .27153+03
   VALVES+FILTERS+PLUMBING+ULLAGING                        .38500+02
   INSULATION (FIXED AND VARIABLE)                         .33411+02
   METEOROID BUMPER                                        .27987+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                  .57904+02
 ENGINE SYSTEM                                                             .15300+03
 INERT SUB-TOTAL                                                           .76033+03
 CONTINGENCY 10%                                                           .76033+02
 RESIDUALS                                                                 .87043+02
   PROPELLANT                                            .62200+02
   VAPOR                                                 .81779+01
   HE GAS                                                .16665+02
 PERFORMANCE RESERVE (1% AV)                                               .28143+02
IMPULSE PROPELLANTS                                                                   .40829+04
PROPULSION MODULE WEIGHT                                                              .50345+04
PAYLOAD WEIGHT                                                                        .29945+04
```

Table C-12a

N_2O_4/A-50 LUNAR CARGO – COMPUTATION FOR N_2O_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHS)
        LAUNCH WEIGHT                         =  .81667+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20004+01 (  %  )
        VENT PRESSURE                         =  .14700+04 ( PSI )
        INITIAL PRESSURANT LOAD               =  .18000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .17500+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .15090+04 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                                CALCULATION TIME                  =   07.989   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .52927+03 (DEG-R)

WEIGHT SUMMARY
    OPTIMUM TANK(S)   =  2  N2O4 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21985+03 ( PSI )      MAX PRESSURE FOR THIS MISSION     =  .15784+03 ( PSI )
        PROPELLANT TANK                       =  .65311+02 ( LBS )      PROPELLANT TANK SURFACE AREA      =  .32459+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .17389+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16072+01 (FEET )      RESIDUAL                          =  .24954+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )      CALCULATED RESIDUAL PROPELLANT VAPOR=  .19170+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .17379+01 ( LBS )      INSULATION                        =  .31106+01 ( LBS )
    PRESSURANT ANALYSIS, PRESSURANT # -    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .56079-00 (FEET )      PRESSURANT SPHERE WEIGHT          =  .12191+02 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .73872-00 (FT-3 )      PRESSURANT REMAINING IN PRESS SPHR =  .62081-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .21603+01
    NONOPTIMUM TANK(S)   =  2   A-50 TANK(S)
        PROPELLANT TANK VOLUME                =  .18287+02 (FT-3 )      PROPELLANT TANK SURFACE AREA      =  .33567+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16344+01 (FEET )      INSULATION                        =  .32168+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .94272+03 ( LBS )      CALCULATED INITIAL PROPELLANT VAPOR =  .33063-01 ( LBS )
        PROPELLANT TANK                       =  .65891+02 ( LBS )      RESIDUAL                          =  .15596+02 ( LBS )
    TOTALS
        PAYLOAD                               =  .23165+04 ( LBS )      LAUNCH WEIGHT                     =  .81667+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )      1 % PERFORMANCE CONTINGENCY       =  .30051+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91656+04
    BURN #   1
        DELTA VELOCITY OBTAINED               =  .40400+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .10848+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.43488-03 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .28699-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .35256+02 ( LBS )

    BURN #   2
        DELTA VELOCITY OBTAINED               =  .28280+02 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .75658+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .43259-02 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .19980-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .24589+02 ( LBS )

    BURN #   3
        DELTA VELOCITY OBTAINED               =  .32269+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .68715+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .60291-00 (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .17865+01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .22332+04 ( LBS )

    BURN #   4
        DELTA VELOCITY OBTAINED               =  .58700+04 (FT/SC)      WEIGHT OF PROPELLANT USED         =  .80160+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)      WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )      TOTAL WEIGHT OF MIXTURE USED      =  .26052+04 ( LBS )
```

Table C-12b

N_2O_4/A-50 LUNAR CARGO – COMPUTATION FOR A-50 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        LAUNCH WEIGHT                           = .81674+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20037+01 ( % )
        VENT PRESSURE                           = .16960+04 ( PSI )
        INITIAL PRESSURANT LOAD                 = .17000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .75000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 = .94272+03 ( LBS )

MISSION PARTICULARS
        4 BURN MISSION                                          CALCULATION TIME                = 04.541   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .61534+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  A-50 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21619+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .16705+03 ( PSI )
        PROPELLANT TANK                         = .65891+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .33567+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .10267+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16344+01 (FEET )     RESIDUAL                        = .15596+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .74469-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .16290+01 ( LBS )     INSULATION                      = .39168+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .56079-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .12191+02 ( LBS )
        PRESSURANT SPHERE VOLUME                = .73872-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .70961-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .22874+01
   NONOPTIMUM TANK(S)   =  2   N2O4 TANK(S)
        PROPELLANT TANK VOLUME                  = .17389+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .32459+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16072+01 (FEET )     INSULATION                      = .31106+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15083+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .44708-01 ( LBS )
        PROPELLANT TANK                         = .65311+02 ( LBS )     RESIDUAL                        = .24959+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .23185+04 ( LBS )     LAUNCH WEIGHT                   = .81674+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .20848+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .91650+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .40400+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .47806+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .52948-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .11277-01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .35259+02 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .28280+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .47290+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .42496-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .77444-02 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .24591+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .37269+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .42950+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .65584-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .69540-00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .22334+04 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .58693+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .50100+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .26052+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  19 MAY 69
   PROPELLANTS :          N2O4 + A-50
   USABLE WEIGHT          .48682+04
 STRUCTURE                                                                .21031+03
   BASE STRUCTURE                                          .11119+03
   TANK SUPPORTS                                           .66118+02
   ATTACHMENTS                                             .15000+02
   BULKHEAD INSULATION (1")                                .18000+02
 PROPELLANT FEED ASSEMBLY                                                 .34947+03
   TANKS                                                   .26240+03
   VALVES+FILTERS+PLUMBING+ULLAGING                        .28000+02
   INSULATION (FIXED AND VARIABLE)                         .32655+02
   METEOROID BUMPER                                        .26910+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                 .68965+02
 ENGINE SYSTEM                                                            .15100+03
 INERT SUB-TOTAL                                                          .77974+03
 CONTINGENCY 10%                                                          .77974+02
 RESIDUALS                                                                .94259+02
   PROPELLANT                                              .81100+02
   VAPOR                                                   .53822+01
   HE GAS                                                  .77770+01
 PERFORMANCE RESERVE (1% ΔV)                                              .30051+02
IMPULSE PROPELLANTS                                                                .48682+04
PROPULSION MODULE WEIGHT                                                           .58503+04
PAYLOAD WEIGHT                                                                     .23165+04
```

Table C-13a

F_2/H_2 JUPITER ORBITER – COMPUTATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        LAUNCH WEIGHT                           =  .65120+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .25025+01 (  %  )
        VENT PRESSURE                           =  .87280+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .00000+00 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .16909+04 ( LBS )

MISSION PARTICULARS
        S BURN MISSION                                          CALCULATION TIME                =   05.081   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .19319+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2 [LF2] MITANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21982+03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .32856+02 ( PSI )
        PROPELLANT TANK                         =  .66116+02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .33996+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16498+01 (FEET )    RESIDUAL                        =  .30692+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .20925+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .73611-00 ( LBS )    INSULATION                      =  .32580+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     19 =   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .33509+00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .13999+01 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .15046-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR  =  .63887-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .22220+01
   NONOPTIMUM TANK(S)    =  1  LH2(F)TANK(S)
        PROPELLANT TANK VOLUME                  =  .85930+02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .96295+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .30727+01 (FEET )    INSULATION                      =  .18957+02 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .33650+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .76180-00 ( LBS )
        PROPELLANT TANK                         =  .15136+03 ( LBS )    RESIDUAL                        =  .51159+01 ( LBS )
   TOTALS
        PAYLOAD                                 =  .17722+04 ( LBS )    LAUNCH WEIGHT                   =  .65120+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    I S PERFORMANCE CONTINGENCY     =  .23618+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12326+05
   BURN #    1
        DELTA VELOCITY OBTAINED                 =  .55302+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .72985+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .69328-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .32099+01 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .20196+04 ( LBS )

   BURN #    2
        DELTA VELOCITY OBTAINED                 =  .50300+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .75922+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .57163-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26162-01 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .16950+02 ( LBS )

   BURN #    3
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .25750+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .17776-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .80955-02 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .85793-01 ( LBS )

   BURN #    4
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .25718+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .11957-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .52289-02 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .85722-01 ( LBS )

   BURN #    5
        DELTA VELOCITY OBTAINED                 =  .67105+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .74699+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .16173+04 ( LBS )
```

Table C-13b

F_2/H_2 JUPITER ORBITER – COMPUTATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      1.000 (INCHS)
        LAUNCH WEIGHT                           = .65100+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .10356+02 (  %  )
        VENT PRESSURE                           = .10770+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .16600+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .33710+03 ( LBS )

MISSION PARTICULARS
        SINGLE MISSION                                                  CALCULATION TIME                = 09.113   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .57447+02 (DEG-R)

WEIGHT SUMMARY
   OPTIMIZED TANK(S)    =  1 [H2](F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .06331+02 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .16574+03 ( PSI )
        PROPELLANT TANK                         = .15392+03 ( LBS )     PROPELLANT TANK SURFACE AREA    = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .85936+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .30727+01 (FEET )     RESIDUAL                        = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .56406+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .00000    ( LBS )     INSULATION                      = .18457+02 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # -     1 - LH2(F)PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .33004+00 (FEET )     PRESSURANT SPHERE WEIGHT        = .13999+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .15066+00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .17776+01
   NONOPTIMIZED TANK(S)    =  2  LF2(F)TANK(S)
        PROPELLANT TANK VOLUME                  = .18630+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .33096+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16648+01 (FEET )     INSULATION                      = .32660+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .16939+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12562-00 ( LBS )
        PROPELLANT TANK                         = .66116+02 ( LBS )     RESIDUAL                        = .30692+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .17668+04 ( LBS )     LAUNCH WEIGHT                   = .65100+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .24442+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12328+05
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .55279+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .15487+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.11112+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .25270+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .20132+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .12652+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.11133+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .20036-00 ( LBS )
        OPERATING PRESSURE                      = .22252+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16447+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .42910-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.10160+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .46226-00 ( LBS )
        OPERATING PRESSURE                      = .37848+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .55783+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .42857-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.35700-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .14733+01 ( LBS )
        OPERATING PRESSURE                      = .15570+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .55714+01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .67149+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .12445+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .16574+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16179+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                    05 JUN 69
        PROPELLANTS :          [H2](F)+LF2(O)
        USABLE WEIGHT          .36343+04
      STRUCTURE                                                          .25070+03
        CASE STRUCTURE                                     .16678+03
        TANK SUPPORTS                                      .50919+02
        ATTACHMENTS                                        .15000+02
        BULKHEAD INSULATION (1**)                          .18000+02
      PROPELLANT FEED ASSEMBLY                                           .49291+03
        TANKS                                              .28615+03
        VALVES+FILTERS+PLUMBING+ULLAGING                   .37000+02
        INSULATION (FIXED AND VARIABLE)                    .39973+02
        METEOROID BUMPER                                   .12979+03
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .26600+02
      ENGINE SYSTEM                                                      .90000+02
      INERT SUB-TOTAL                                                    .86821+03
      CONTINGENCY 10.                                                    .86821+02
      RESIDUALS                                                          .12944+03
        PROPELLANT                                         .66500+02
        VAPOR                                              .61161+02
        HE GAS                                             .17776+01
      PERFORMANCE RESERVE (1% ΔV)                                        .24442+02
    IMPULSE PROPELLANTS                                                                .36343+04
    PROPULSION MODULE WEIGHT                                                           .47932+04
    PAYLOAD WEIGHT                                                                     .17668+04
```

Table C-14a

FLOX/CH_4 JUPITER ORBITER – COMPUTATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      .500 (INCHES)
        LAUNCH WEIGHT                           =  .66072*04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20010*01 (  %  )
        VENT PRESSURE                           =  .79500*03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .60000-00 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .13000*03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .17167*04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                =   07.263   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .16060*03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21133*03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .90577*03 ( PSI )
        PROPELLANT TANK                         =  .66708*02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .35128*03 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .17578*02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16720*01 (FEET )    RESIDUAL                        =  .27216*02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .49919*01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .54352-00 ( LBS )    INSULATION                      =  .53665*01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .38782-00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .26572*01 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .24432-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR  =  .56490-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .22220*01
   NONOPTIMUM TANK(S)   =  2  CH4(F)TANK(S)
        PROPELLANT TANK VOLUME                  =  .13070*02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .26833*03 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .19613*01 (FEET )    INSULATION                      =  .25715*01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .32635*03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .69185-01 ( LBS )
        PROPELLANT TANK                         =  .62367*02 ( LBS )    RESIDUAL                        =  .51800*01 ( LBS )
   TOTALS
        PAYLOAD                                 =  .14614*04 ( LBS )    LAUNCH WEIGHT                   =  .66072*04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    I B PERFORMANCE CONTINGENCY     =  .24173*02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12936*05
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .54419*04 (FT/SE)    WEIGHT OF PROPELLANT USED       =  .90039*03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .74062-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .32973*01 ( LBS )
        OPERATING PRESSURE                      =  .20000*02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .23343*04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .50500*02 (FT/SE)    WEIGHT OF PROPELLANT USED       =  .76704*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .60239*02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .25837-01 ( LBS )
        OPERATING PRESSURE                      =  .20000*02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .18025*03 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .17170*02 (FT/SE)    WEIGHT OF PROPELLANT USED       =  .25667*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .23212*02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .10046-01 ( LBS )
        OPERATING PRESSURE                      =  .20000*02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .61111*01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .17170*02 (FT/SE)    WEIGHT OF PROPELLANT USED       =  .25630*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .29392*02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .12976-01 ( LBS )
        OPERATING PRESSURE                      =  .24190*02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .61073*01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 =  .47095*04 (FT/SE)    WEIGHT OF PROPELLANT USED       =  .72077*03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .24170*02 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .17141*04 ( LBS )
```

Table C-14b

FLOX/CH_4 JUPITER ORBITER – COMPUTATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        LAUNCH WEIGHT                           =  .66046+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .29020+01 (  %  )
        VENT PRESSURE                           =  .67310+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .60000-00 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .32635+03 ( LBS )

MISSION PARTICULARS                                                     CALCULATION TIME                =  86.608  ( SEC )
        5 BURN MISSION
        FINAL PROPELLANT TEMPERATURE            =  .16020+03 (DEG.R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  CH4 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .24180+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   =  .32373+02 ( PSI )
        PROPELLANT TANK                         =  .42567+02 ( LBS )     PROPELLANT TANK SURFACE AREA    =  .26635+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .13070+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .19613+01 (FEET )     RESIDUAL                        =  .51890+01 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR=  .10601+00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .56053-00 ( LBS )     INSULATION                      =  .25715+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 -  HE  PRESSURANT SPHERE. INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .38782-00 (FEET )     PRESSURANT SPHERE WEIGHT        =  .26572+01 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .29432-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR =  .39446-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .22220+01
   NONOPTIMUM TANK(S)    =  2   FLOX TANK(S)
        PROPELLANT TANK VOLUME                  =  .19578+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    =  .35120+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16720+01 (FEET )     INSULATION                      =  .33665+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .17187+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR =  .12462+00 ( LBS )
        PROPELLANT TANK                         =  .66706+02 ( LBS )     RESIDUAL                        =  .27216+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .16591+04 ( LBS )     LAUNCH WEIGHT                   =  .66046+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )     1 S. PERFORMANCE CONTINGENCY    =  .24319+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12434+05
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .56388+04 (FT/SEC)    WEIGHT OF PROPELLANT USED       =  .18658+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .11364+01 (DEG.R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .75238-00 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .23373+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .50500+02 (FT/SEC)    WEIGHT OF PROPELLANT USED       =  .14410+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .86651-02 (DEG.R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .46554-02 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .18022+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SEC)    WEIGHT OF PROPELLANT USED       =  .40001-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .16510-02 (DEG.R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .10242-02 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .61101+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SEC)    WEIGHT OF PROPELLANT USED       =  .40811-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .40295-03 (DEG.R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .22931-03 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .61813+01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 =  .67103+04 (FT/SEC)    WEIGHT OF PROPELLANT USED       =  .13728+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG.R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .17150+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  01 MAY 69
        PROPELLANTS :          FLOX ,CH4(P)
        USABLE WEIGHT          .40544+04
      STRUCTURE                                                          .17720+03
        BASE STRUCTURE                                     .94300+02
        TANK SUPPORTS                                      .99000+02
        ATTACHMENTS                                        .15000+02
        BULKHEAD INSULATION (S***)                         .18000+02
      PROPELLANT FEED ASSEMBLY                                           .91527+03
        TANKS                                              .25015+03
        VALVES,FILTERS,PLUMBING,ULLAGING                   .38500+02
        INSULATION (FIXED AND VARIABLE)                    .31074+03
        METEOROID BUMPER                                   .66794+02
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .25514+02
      ENGINE SYSTEM                                                      .98000+02
      INERT SUB-TOTAL                                                    .71890+03
      CONTINGENCY 10%                                                    .71899+02
      RESIDUALS                                                          .77390+02
        PROPELLANT                                         .64800+02
        VAPOR                                              .99238+01
        HE GAS                                             .26664+01
      PERFORMANCE RESERVE (10 AVS                                        .29173+02
    IMPULSE PROPELLANTS                                                                .40544+04
    PROPULSION MODULE WEIGHT                                                           .99456+04
    PAYLOAD WEIGHT                                                                     .16610+04
```

Table C-15a

OF_2/B_2H_6 JUPITER ORBITER – COMPUTATION FOR OF_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .625 (INCHES)
        LAUNCH WEIGHT                           = .66159*04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .70260*01 (  %  )
        VENT PRESSURE                           = .71860*03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .90000*01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .29100*03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .15982*04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                = 23.224   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .23975*03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  OF2(C)TANK(S)
        MINIMUM BAGE CUTOFF PRESSURE            = .21098*03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .22589*03 ( PSI )
        PROPELLANT TANK                         = .69270*02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .36296*03 (FT-2 )
        PROPELLANT TANK VOLUME                  = .19674*02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16797*01 (FEET )     RESIDUAL                        = .23325*03 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .92809*01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .39298*01 ( LBS )     INSULATION                      = .82221*01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17 =  HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .77468-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .21868*03 ( LBS )
        PRESSURANT SPHERE VOLUME                = .19479*01 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .70220-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .29169*01
   NONOPTIMUM TANK(S)    =  2  B2H6 TANK(S)
        PROPELLANT TANK VOLUME                  = .19240*02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .34723*03 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16423*01 (FEET )     INSULATION                      = .33276*01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .51490*03 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .92778-01 ( LBS )
        PROPELLANT TANK                         = .66496*02 ( LBS )     RESIDUAL                        = .77750*01 ( LBS )
   TOTALS
        PAYLOAD                                 = .19788*04 ( LBS )     LAUNCH WEIGHT                   = .66159*04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .24296*02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12996*05
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .54515*04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .86641*03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .11191*01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .30998*01 ( LBS )
        OPERATING PRESSURE                      = .15500*03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .23438*04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500*02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .46925*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .93687-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .25723-01 ( LBS )
        OPERATING PRESSURE                      = .15500*03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16380*02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170*02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .23367*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .62752-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .12253-01 ( LBS )
        OPERATING PRESSURE                      = .15500*03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .62311*01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .17170*02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .23332*01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .39158-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .10646*01 ( LBS )
        OPERATING PRESSURE                      = .15500*03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .62219*01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .67096*04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .44637*03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500*03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .19237*04 ( LBS )
```

Table C-15b

OF_2/B_2H_6 JUPITER ORBITER – COMPUTATION FOR B_2H_6 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHES)
        LAUNCH WEIGHT                         =  .66169+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20011+01 (  %  )
        VENT PRESSURE                         =  .58100+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .47000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .10000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .51990+03 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                  =   05.591  ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .23017+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  [B2H6] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21254+03 ( PSI )    MAX PRESSURE FOR THIS MISSION     =  .16508+03 ( PSI )
        PROPELLANT TANK                       =  .66496+02 ( LBS )    PROPELLANT TANK SURFACE AREA      =  .34723+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .19290+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16623+01 (FEET )    RESIDUAL                          =  .77750+01 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .34500+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .46951+01 ( LBS )    INSULATION                        =  .33296+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =    17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .77763-00 (FEET )    PRESSURANT SPHERE WEIGHT          =  .22119+02 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .19698+01 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .99338-02 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .20802+01
   NONOPTIMUM TANK(S)   =  2  OF2(C)TANK(S)
        PROPELLANT TANK VOLUME                =  .19676+02 (FT-3 )    PROPELLANT TANK SURFACE AREA      =  .35296+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16797+01 (FEET )    INSULATION                        =  .33777+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .15475+04 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR=  .11188+01 ( LBS )
        PROPELLANT TANK                       =  .69270+02 ( LBS )    RESIDUAL                          =  .23325+02 ( LBS )
   TOTALS
        PAYLOAD                               =  .14809+04 ( LBS )    LAUNCH WEIGHT                     =  .66169+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )    1 S PERFORMANCE CONTINGENCY       =  .24906+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12996+05
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .56526+04 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .24556+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .19193+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .11441+01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .23695+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .22976+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .10467-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .83104-02 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .16381+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .77893-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .11404-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .82954-03 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .62318+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .77779-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .10681-05 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .69722-04 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .62223+01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED               =  .47109+04 (FT/SC)    WEIGHT OF PROPELLANT USED         =  .21549+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED      =  .17239+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  28 MAY 69
        PROPELLANTS :      [OF2](C), B2H6
        USABLE WEIGHT      .40990+04
      STRUCTURE                                                          .17800+03
        BASE STRUCTURE                                   .95000+02
        TANK SUPPORTS                                    .50000+02
        ATTACHMENTS                                      .15000+02
        BULKHEAD INSULATION (1**)                        .10000+02
      PROPELLANT FEED ASSEMBLY                                           .99868+03
        TANKS                                            .27153+03
        VALVES,FILTERS,PLUMBING,ULLAGING                 .38500+02
        INSULATION (FIXED AND VARIABLE)                  .35099+02
        METEOROID BUMPER                                 .10355+03
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .63936+02
      ENGINE SYSTEM                                                      .15300+03
      INERT SUB-TOTAL                                                    .89362+03
      CONTINGENCY 10%                                                    .89362+02
      RESIDUALS                                                          .98417+02
        PROPELLANT                                       .62200+02
        VAPOR                                            .88860+01
        HE GAS                                           .19331+02
      PERFORMANCE RESERVE (1% ΔV)                                        .89296+02
    IMPULSE PROPELLANTS                                                              .40940+04
    PROPULSION MODULE WEIGHT                                                         .51366+04
    PAYLOAD WEIGHT                                                                   .14768+04
```

Table C-16a

N_2O_4/A-50 JUPITER ORBITER – COMPUTATION FOR N_2O_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      5.000 (INCHES)
        LAUNCH WEIGHT                           =  .69611+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20556+01 (  %  )
        VENT PRESSURE                           =  .19700+04 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .20000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .17500+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .15989+04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                  =   21.791   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .98287+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  [N2O4] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21985+03 ( PSI )        MAX PRESSURE FOR THIS MISSION     =  .21985+03 ( PSI )
        PROPELLANT TANK                         =  .65511+02 ( LBS )        PROPELLANT TANK SURFACE AREA      =  .32459+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .17389+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16072+01 (FEET )        RESIDUAL                          =  .29954+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )        CALCULATED RESIDUAL PROPELLANT VAPOR=  .79152-00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .19672+01 ( LBS )        INSULATION                        =  .31106+02 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .59593+00 (FEET )        PRESSURANT SPHERE WEIGHT          =  .19629+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .88697+00 (FT-3 )        PRESSURANT REMAINING IN PRESS SPHR =  .32001-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .23331+01
   NONOPTIMUM TANK(S)    =  2   A-50 TANK(S)
        PROPELLANT TANK VOLUME                  =  .18287+02 (FT-3 )        PROPELLANT TANK SURFACE AREA      =  .33567+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16399+01 (FEET )        INSULATION                        =  .25734+02 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .96750+03 ( LBS )        CALCULATED INITIAL PROPELLANT VAPOR =  .97920-07 ( LBS )
        PROPELLANT TANK                         =  .65891+02 ( LBS )        RESIDUAL                          =  .15896+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .78914+03 ( LBS )        LAUNCH WEIGHT                     =  .69611+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )        I.S. PERFORMANCE CONTINGENCY      =  .29765+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12893+05
   BURN #    1
        DELTA VELOCITY OBTAINED                 =  .60482+04 (FT/SC)        WEIGHT OF PROPELLANT USED         =  .97105+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .21299-01 (DEG-R)        WEIGHT OF PROPELLANT VAPORIZED AFTER=  .11977-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )        TOTAL WEIGHT OF MIXTURE USED      =  .31659+04 ( LBS )

   BURN #    2
        DELTA VELOCITY OBTAINED                 =  .58300+02 (FT/SC)        WEIGHT OF PROPELLANT USED         =  .63198+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.81542-01 (DEG-R)        WEIGHT OF PROPELLANT VAPORIZED AFTER=  .11132-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )        TOTAL WEIGHT OF MIXTURE USED      =  .20523+02 ( LBS )

   BURN #    3
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)        WEIGHT OF PROPELLANT USED         =  .21393+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.44909+02 (DEG-R)        WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26207+01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )        TOTAL WEIGHT OF MIXTURE USED      =  .69526+01 ( LBS )

   BURN #    4
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)        WEIGHT OF PROPELLANT USED         =  .21353+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.29602-01 (DEG-R)        WEIGHT OF PROPELLANT VAPORIZED AFTER=  .27165-02 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )        TOTAL WEIGHT OF MIXTURE USED      =  .69396.01 ( LBS )

   BURN #    5
        DELTA VELOCITY OBTAINED                 =  .67099+04 (FT/SC)        WEIGHT OF PROPELLANT USED         =  .56656+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)        WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )        TOTAL WEIGHT OF MIXTURE USED      =  .18913.04 ( LBS )
```

Table C-16b

N_2O_4/A-50 JUPITER ORBITER – COMPUTATION FOR A-50 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      4.000 (INCHS)
        LAUNCH WEIGHT                           = .69600+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .61473+01 (  %  )
        VENT PRESSURE                           = .16960+04 ( PSI )
        INITIAL PRESSURANT LOAD                 = .22000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .75000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 = .96750+03 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                = 09.581   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .47832+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2   A-50 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21619+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .19007+03 ( PSI )
        PROPELLANT TANK                         = .65691+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .33567+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .18287+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16344+01 (FEET )     RESIDUAL                        = .15596+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .59006-01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .21130+01 ( LBS )     INSULATION                      = .25734+02 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # -    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .59593-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .14629+02 ( LBS )
        PRESSURANT SPHERE VOLUME                = .88647-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .86954-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .21210+01
   NONOPTIMUM TANK(S)    =  2   N2O4 TANK(S)
        PROPELLANT TANK VOLUME                  = .17389+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .32459+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16072+01 (FEET )     INSULATION                      = .31106+02 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15468+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .38163-01 ( LBS )
        PROPELLANT TANK                         = .65311+02 ( LBS )     RESIDUAL                        = .24954+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .78322+03 ( LBS )     LAUNCH WEIGHT                   = .69600+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .24915+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .12843+05
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .60470+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .60671+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.57821-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .93248-01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .31549+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .39466+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.97462-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .93473-03 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .20522+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .13370+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.52191+02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .23796-00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .69524+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .13345+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.30758-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .20091-03 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .69396+01 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .67107+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .35412+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .18414+04 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                    04 JUN 69
   PROPELLANTS :          N2O4 ; A-50
   USABLE WEIGHT          .50069+04
 STRUCTURE                                                                  .21031+03
   BASE STRUCTURE                                                .11117+03
   TANK SUPPORTS                                                 .66118+02
   ATTACHMENTS                                                   .15000+02
   BULKHEAD INSULATION (1**)                                     .16000+02
 PROPELLANT FEED ASSEMBLY                                                   .51784+03
   TANKS                                                         .26240+03
   VALVES,FILTERS,PLUMBING,ULLAGING                              .20000+02
   INSULATION (FIXED AND VARIABLE)                               .13368+03
   METEOROID BUMPER                                              .93786+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                        0          .70717+02
 ENGINE SYSTEM                                                              .15100+03
 INERT SUB-TOTAL                                                            .98787+03
 CONTINGENCY 10%                                                            .95787+02
 RESIDUALS                                                                  .91669+02
   PROPELLANT                                                    .01100+02
   VAPOR                                                         .12568+01
   HE GAS                                                        .93339+01
 PERFORMANCE RESERVE (10.6V)                                                .24755+02
INPUSE PROPELLANTS                                                                  .50069+04
PROPULSION MODULE WEIGHT                                                            .61778+04
PAYLOAD WEIGHT                                                                      .78319+03
```

Table C-17a

F_2/H_2 MARS ORBITER WITH 6,000 LB PROPELLANT – OPTIMIZATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      .500 (INCHES)
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20000+01 (  %  )
        VENT PRESSURE                           = .62280+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .70500+00 ( LBS )
        MAXIMUM TANK PRESSURE                   = .13500+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .14010+04 ( LBS )
        LAUNCH WEIGHT                           = .12865+05 ( LBS )

MISSION PARTICULARS                                                     CALCULATION TIME          =   151390   ( SEC )
        6 BURN MISSION
        FINAL PROPELLANT TEMPERATURE            = .15862+03 (DEG-R)

WEIGHT SUMMARY
    OPTIMUM TANK(S)    =  4  LF2 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .22853+03 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .44273+02 ( PSI )
        PROPELLANT TANK                         = .69095+02 ( LBS )     PROPELLANT TANK SURFACE AREA      = .30039+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .15481+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .15461+01 (FEET )     RESIDUAL                          = .26934+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .46176+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .66173-00 ( LBS )     INSULATION                        = .26787+01 ( LBS )
    PRESSURANT ANALYSIS, PRESSURANT # =    17 -  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .39779-00 (FEET )     PRESSURANT SPHERE WEIGHT          = .24998+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .26365-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .38266-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .49990+01
    NONOPTIMUM TANK(S)   =  1  LH2,P TANK(S)
        PROPELLANT TANK VOLUME                  = .17313+03 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .12239+03 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .39692+01 (FEET )     INSULATION                        = .67993+02 ( LBS )
        INITIAL PROPELLANT LOAD                 = .52622+03 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .22656-05 ( LBS )
        PROPELLANT TANK                         = .13839+03 ( LBS )     RESIDUAL                          = .96962+01 ( LBS )
    TOTALS
        PAYLOAD                                 = .53979+04 ( LBS )     LAUNCH WEIGHT                     = .12865+05 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY       = .92981+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .64935+04
    BURN #   1
        DELTA VELOCITY OBTAINED                 = .29290+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .99963+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .27418-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .15994+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .19267+04 ( LBS )

    BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .92324+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .59404-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .33529-01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .49009+02 ( LBS )

    BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED         = .31314+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .30174-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .17893-01 ( LBS )
        OPERATING PRESSURE                      = .26860+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .33570+02 ( LBS )

    BURN #   4
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED         = .89919+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .65699+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .32024+01 ( LBS )
        OPERATING PRESSURE                      = .27457+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .38965+04 ( LBS )

    BURN #   5
        DELTA VELOCITY OBTAINED                 = .33128+03 (FT/SC)     WEIGHT OF PROPELLANT USED         = .38304+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .23451+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .16598+03 ( LBS )
```

Table C-17b

F_2/H_2 MARS ORBITER WITH 6,000 LB PROPELLANT – OPTIMIZATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      2.875 (INCHES)
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20000+01 ( % )
        VENT PRESSURE                           = .18770+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .10000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .10000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .52827+03 ( LBS )
        LAUNCH WEIGHT                           = .12866+05 ( LBS )

MISSION PARTICULARS                                                     CALCULATION TIME                  =  24.535   ( SEC )
        5 BURN MISSION
        FINAL PROPELLANT TEMPERATURE            = .51335+02 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  1  LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .85446+02 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .97822+02 ( PSI )
        PROPELLANT TANK                         = .13639+03 ( LBS )     PROPELLANT TANK SURFACE AREA      = .12239+03 (FT-2 )
        PROPELLANT TANK VOLUME                  = .12313+03 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .39442+01 (FEET )     RESIDUAL                          = .94562+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .61711+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .00000    ( LBS )     INSULATION                        = .69993+03 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     4 - LH2(F)PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .39779+00 (FEET )     PRESSURANT SPHERE WEIGHT          = .24994+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .26365+00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .31108+01
   NONOPTIMUM TANK(S)    =  4  LF2(H)TANK(S)
        PROPELLANT TANK VOLUME                  = .15481+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .30039+03 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .15441+01 (FEET )     INSULATION                        = .24797+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .19010+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .10921+00 ( LBS )
        PROPELLANT TANK                         = .69045+02 ( LBS )     RESIDUAL                          = .24938+03 ( LBS )
   TOTALS
        PAYLOAD                                 = .53904+04 ( LBS )     LAUNCH WEIGHT                     = .12866+05 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY       = .91807+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .99903+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .24290+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .14822+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.30115-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .21553+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .16268+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+03 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .30777+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.55192-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.85053-01 ( LBS )
        OPERATING PRESSURE                      = .20830+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .80010+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17130+02 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .10938+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.35610-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= -.11451+00 ( LBS )
        OPERATING PRESSURE                      = .67275+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .13570+03 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .29979+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.29739+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .37450+02 ( LBS )
        OPERATING PRESSURE                      = .79186+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .53416+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .37810+03 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .12649+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .97822+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .14441+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  20 MAY 69
      PROPELLANTS :        LF2(H)-LH2(F)
      USABLE WEIGHT        .5999+04
    STRUCTURE                                                                 126 .8[illegible]
      BASE STRUCTURE                                          .21906+03
      TANK SUPPORTS                                       77) .6[illegible]
      ATTACHMENTS                                             .15590+02
      BULKHEAD INSULATION (I**)                               .18000+02
    PROPELLANT FEED ASSEMBLY                                                  .67350+03
      TANKS                                                   .38457+03
      VALVES,FILTERS,PLUMBING,ULLAGING                        .94000+02
      INSULATION (FIXED AND VARIABLE)                         .10396+03
      METEOROID BUMPER                                        .33097+03
    PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                      .29650+02
    ENGINE SYSTEM                                                                 .94000+02
    INERT SUB-TOTAL                                                               .11553+04
    CONTINGENCY 10%                                                           112 .1[illegible]
    RESIDUALS                                                                     .21665+03
      PROPELLANT                                              .12590+03
      VAPOR                                                 . .98087+02
      HE GAS                                                  .31108+01
    PERFORMANCE RESERVE (1% ΔV)                                                   .92981+02
    IMPULSE PROPELLANTS                                                                       .59999+04
    PROPULSION MODULE WEIGHT                                                          749) .7[illegible]
    PAYLOAD WEIGHT                                                                    F110 .5[illegible]
```

Table C-18a

FLOX/CH_4 MARS ORBITER WITH 6,000 LB PROPELLANT – OPTIMIZATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20000+01 (  %  )
        VENT PRESSURE                           = .79500+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .13000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .25485+04 ( LBS )
        LAUNCH WEIGHT                           = .12083+05 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                = 08.843  ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .15870+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  [FLOX] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .20000+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .43018+02 ( PSI )
        PROPELLANT TANK                         = .86517+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .43349+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .28961+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .80618-00 (FEET )     RESIDUAL                        = .34230+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .11259+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .12585+01 ( LBS )     INSULATION                      = .41543+01 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # =    17 =  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .44394-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .39058+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .36648-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .41540-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .15383+01
   NONOPTIMUM TANK(S)    =  2  CH4(F)TANK(S)
        PROPELLANT TANK VOLUME                  = .19223+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .34702+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16618+01 (FEET )     INSULATION                      = .33256+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .49542+03 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .83466-01 ( LBS )
        PROPELLANT TANK                         = .66485+02 ( LBS )     RESIDUAL                        = .65200+01 ( LBS )
   TOTALS
        PAYLOAD                                 = .50405+04 ( LBS )     LAUNCH WEIGHT                   = .12083+05 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .41688+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .89387+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .19190+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .69925+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .23224-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .24589+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16649+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .18458+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .62370-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .65453-01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .43948+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .62580+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .31204-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .33927-01 ( LBS )
        OPERATING PRESSURE                      = .25776+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .14900+02 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .17420+04 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .71402+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .55646+01 ( LBS )
        OPERATING PRESSURE                      = .26427+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .41477+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .33145+03 (FT/SC)     WEIGHT OF PROPELLANT USED       = .71390+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .22178+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16998+03 ( LBS )
```

Table C-18b

FLOX/CH_4 MARS ORBITER WITH 6,000 LB PROPELLANT – OPTIMIZATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .20000+01 (  %  )
        VENT PRESSURE                           =  .67310+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .60000-00 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .98491+03 ( LBS )
        LAUNCH WEIGHT                           =  .12082+05 ( LBS )

MISSION PARTICULARS                                          CALCULATION TIME                =   12.690   ( SEC )
        5 BURN MISSION
        FINAL PROPELLANT TEMPERATURE            =  .19610+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  CH4 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21262+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   =  .34514+02 ( PSI )
        PROPELLANT TANK                         =  .66486+02 ( LBS )     PROPELLANT TANK SURFACE AREA    =  .34703+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .19223+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16610+01 (FEET )     RESIDUAL                        =  .66200+01 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR=  .16969+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .50310-00 ( LBS )     INSULATION                      =  .33257+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17 -  HE  PRESSURANT SPHERE INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .95201-00 (FEET )     PRESSURANT SPHERE WEIGHT        =  .92073+01 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .39684-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR  =  .96903-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .35182+01
   NONOPTIMUM TANK(S)    =  2  FLOX TANK(S)
        PROPELLANT TANK VOLUME                  =  .20960+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    =  .93399+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .40616+00 (FEET )     INSULATION                      =  .91593+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .25484+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR =  .18406-03 ( LBS )
        PROPELLANT TANK                         =  .86516+02 ( LBS )     RESIDUAL                        =  .38230+02 ( LBS )
   TOTALS
        PAYLOAD                                 =  .50402+04 ( LBS )     LAUNCH WEIGHT                   =  .12082+05 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     =  .91940+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .89372+04
   BURN #   1
        DELTA VELOCITY OBTAINED                 =  .19190+04 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .13316+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .35720-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .56619-00 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .16498+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 =  .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .35156+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .99528-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .14832-01 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .93946+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .11919+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .39580-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .59999-02 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .14899+02 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 =  .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .33180+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .70306+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .94503-00 ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .91976+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 =  .33196+03 (FT/SC)     WEIGHT OF PROPELLANT USED       =  .13616+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    =  .17022+03 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  01 MAY 69
   PROPELLANTS 1          FLOX/CH4(F)
   USABLE WEIGHT          .59997+04
 STRUCTURE                                                                        .22753+03
   BASE STRUCTURE                                                .13452+03
   TANK SUPPORTS                                                 .60011+02
   ATTACHMENTS                                                   .15000+02
   BULKHEAD INSULATION (1")                                      .18000+02
 PROPELLANT FEED ASSEMBLY                                                         .45439+03
   TANKS                                                         .30600+03
   VALVES+FILTERS+PLUMBING+ULLAGING                              .38500+02
   INSULATION (FIXED AND VARIABLE)                               .34960+02
   METEOROID BUMPER                                              .74929+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                         .28172+02
 ENGINE SYSTEM                                                                    .98000+02
 INERT SUB-TOTAL                                                                  .80809+03
 CONTINGENCY 10%                                                                  .80809+02
 RESIDUALS                                                                        .11240+03
   PROPELLANT                                                    .81508+02
   VAPOR                                                         .26903+02
   HE GAS                                                        .39996+01
 PERFORMANCE RESERVE (1% ΔV)                                                      .41688+02
IMPULSE PROPELLANTS                                                                          .59997+04
PROPULSION MODULE WEIGHT                                                                     .70427+04
PAYLOAD WEIGHT                                                                               .50405+04
```

Table C-19a

F_2/H_2 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR F_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      .625 (INCHES)
        LAUNCH WEIGHT                           = .69494+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20025+01 ( % )
        VENT PRESSURE                           = .02280+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .70000-00 ( LBS )
        MAXIMUM TANK PRESSURE                   = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .16904+04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                      CALCULATION TIME                  =  08.701   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .15887+03 (DEGKR)

WEIGHT SUMMARY
    OPTIMUM TANK(S)    =  2  (H2 INITANKS)
        MINIMUM GAGE CUTOFF PRESSURE            = .21982+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .94012+02 ( PSI )
        PROPELLANT TANK                         = .66116+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .33996+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .18639+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIMS     = .16990+01 (FEET )     RESIDUAL                        = .30692+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR = .81304+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .66877-00 ( LBS )     INSULATION                      = .90724+01 ( LBS )
    PRESSURANT ANALYSIS, PRESSURANT = (IF HE, PRESSURANT SPHERE OUTSIDE PROPELLANT TANK)
        PRESSURANT SPHERE RADIUS                = .31572-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .12249+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .13182-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .31229-01 ( LBS )
        GAS FACTOR (OTHER TANK = RESIDUALS)     = .22220+01
    NONOPTIMUM TANK(S)    =  1  (LH2 PTANKS)
        PROPELLANT TANK VOLUME                  = .05930+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .96298+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIMS     = .30727+01 (FEET )     INSULATION                      = .93038+02 ( LBS )
        INITIAL PROPELLANT LOAD                 = .33930+03 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .70580-00 ( LBS )
        PROPELLANT TANK                         = .12923+03 ( LBS )     RESIDUAL                        = .81196+01 ( LBS )
    TOTALS
        PAYLOAD                                 = .22590+04 ( LBS )     LAUNCH WEIGHT                   = .69494+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 S PERFORMANCE CONTINGENCY     = .24941+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .11091+05
    BURN #   1
        DELTA VELOCITY OBTAINED                 = .40709+04 (FT/SEC)    WEIGHT OF PROPELLANT USED       = .76495+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .97840+00 (DEGKR)     WEIGHT OF PROPELLANT VAPORIZED AFTER = .26706+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16663+04 ( LBS )
    BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SEC)    WEIGHT OF PROPELLANT USED       = .89273+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .57945+02 (DEGKR)     WEIGHT OF PROPELLANT VAPORIZED AFTER = .32086+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .19545+02 ( LBS )
    BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SEC)    WEIGHT OF PROPELLANT USED       = .30275+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .29031+02 (DEGKR)     WEIGHT OF PROPELLANT VAPORIZED AFTER = .17176+01 ( LBS )
        OPERATING PRESSURE                      = .26494+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .65505+01 ( LBS )
    BURN #   4
        DELTA VELOCITY OBTAINED                 = .44205+04 (FT/SEC)    WEIGHT OF PROPELLANT USED       = .84944+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .53730+01 (DEGKR)     WEIGHT OF PROPELLANT VAPORIZED AFTER = .26816+01 ( LBS )
        OPERATING PRESSURE                      = .27239+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16030+04 ( LBS )
    BURN #   5
        DELTA VELOCITY OBTAINED                 = .33149+03 (FT/SEC)    WEIGHT OF PROPELLANT USED       = .97854+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEGKR)     WEIGHT OF PROPELLANT VAPORIZED AFTER = .00000    ( LBS )
        OPERATING PRESSURE                      = .23716+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .00286+02 ( LBS )
```

Table C-19b

F_2/H_2 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR H_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      2.375 (INCHES)
        LAUNCH WEIGHT                           = .69436+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .97472+01 ( % )
        VENT PRESSURE                           = .18770+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .10006+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .12750+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .33950+03 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                                  CALCULATION TIME                = 07.049   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .54209+02 (DEG-R)

WEIGHT SUMMARY
    OPTIMUM TANK(S)    =  1  LH2(F)TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .96331+02 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .12451+03 ( PSI )
        PROPELLANT TANK                         = .12923+03 ( LBS )     PROPELLANT TANK SURFACE AREA    = .96295+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .85930+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .30727+01 (FEET )     RESIDUAL                        = .51154+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .59699+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .00000    ( LBS )     INSULATION                      = .43834+02 ( LBS )
    PRESSURANT ANALYSIS: PRESSURANT # -     4 - LH2(F)PRESSURANT SPHERE INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .31572-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .12249+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .13182-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .10000+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .15554+01
    NONOPTIMUM TANK(S)    =  2  LF2(H)TANK(S)
        PROPELLANT TANK VOLUME                  = .18639+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .33996+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16448+01 (FEET )     INSULATION                      = .32580+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .16908+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12598-00 ( LBS )
        PROPELLANT TANK                         = .66116+02 ( LBS )     RESIDUAL                        = .30692+02 ( LBS )
    TOTALS
        PAYLOAD                                 = .22557+04 ( LBS )     LAUNCH WEIGHT                   = .69436+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .25354+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .11094+05
    BURN #   1
        DELTA VELOCITY OBTAINED                 = .40702+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .12737+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.91923-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .19549+01 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .16558+04 ( LBS )

    BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .14877+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.05153-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .10038+00 ( LBS )
        OPERATING PRESSURE                      = .23102+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .19341+02 ( LBS )

    BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .50459-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.53116-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .29328-00 ( LBS )
        OPERATING PRESSURE                      = .89023+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .65596+01 ( LBS )

    BURN #   4
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .14489+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.24383+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .32146+02 ( LBS )
        OPERATING PRESSURE                      = .99048+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .18836+04 ( LBS )

    BURN #   5
        DELTA VELOCITY OBTAINED                 = .33527+03 (FT/SC)     WEIGHT OF PROPELLANT USED       = .62456+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .12651+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .81193+02 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                                  18 JUN 69
    PROPELLANTS :          LH2(M),LH2(F)
    USABLE WEIGHT          .36218+04
  STRUCTURE                                                                  .25870+03
    BASE STRUCTURE                                          .16670+03
    TANK SUPPORTS                                           .50919+02
    ATTACHMENTS                                             .15000+02
    BULKHEAD INSULATION (1**)                               .16050+02
  PROPELLANT FEED ASSEMBLY                                                   .99266+03
    TANKS                                                   .26144+03
    VALVES,FILTERS,PLUMBING,ULLAGING                        .37809+02
    INSULATION (FIXED AND VARIABLE)                         .62979+02
    METEOROID BUMPER                                        .77215+02
  PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                   .26925+02
  ENGINE SYSTEM                                                              .90000+02
  INERT SUB-TOTAL                                                            .81274+03
  CONTINGENCY 10%                                                            .81274+02
  RESIDUALS                                                                  .19084+03
    PROPELLANT                                              .66800+02
    VAPOR                                                   .78989+02
    HE GAS                                                  .15859+01
  PERFORMANCE RESERVE (10 %)                                                 .24461+02
IMPULSE PROPELLANTS                                                                   .36218+04
PROPULSION MODULE WEIGHT                                                              .96899+04
PAYLOAD WEIGHT                                                                        .22540+04
```

Table C-20a

FLOX/CH_4 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR FLOX SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .428 (INCHES)
        LAUNCH WEIGHT                         =  .70303+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20000+01 (  %  )
        VENT PRESSURE                         =  .79500+03 ( PSI )
        INITIAL PRESSURANT LOAD               =  .00000+00 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .17107+04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                        CALCULATION TIME          =   12.754   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .16486+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  FLOX TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21133+03 ( PSI )   MAX PRESSURE FOR THIS MISSION     =  .50631+02 ( PSI )
        PROPELLANT TANK                       =  .66708+02 ( LBS )   PROPELLANT TANK SURFACE AREA      =  .38128+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .19570.02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16778+01 (FEET )   RESIDUAL                          =  .27216+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )   CALCULATED RESIDUAL PROPELLANT VAPOR=  .10307+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .79392-00 ( LBS )   INSULATION                        =  .92081+01 ( LBS )
PRESSURANT ANALYSIS, PRESSURANT # =    1 ,  HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .37673-00 (FEET )   PRESSURANT SPHERE WEIGHT          =  .29356+01 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .22396-00 (FT-3 )   PRESSURANT REMAINING IN PRESS SPHR =  .45896-02 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUAL)    =  .15276+01
   NONOPTIMUM TANK(S)   =  2  CHAMP TANK(S)
        PROPELLANT TANK VOLUME                =  .13870+02 (FT-3 )   PROPELLANT TANK SURFACE AREA      =  .26833+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .19613+01 (FEET )   INSULATION                        =  .25719+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .32367+03 ( LBS )   CALCULATED INITIAL PROPELLANT VAPOR =  .85885-01 ( LBS )
        PROPELLANT TANK                       =  .42367+02 ( LBS )   RESIDUAL                          =  .51848+01 ( LBS )
TOTALS
        PAYLOAD                               =  .21271+04 ( LBS )   LAUNCH WEIGHT                     =  .70303+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )   1 % PERFORMANCE CONTINGENCY       =  .25506+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .11194+04
   BURN #   1
        DELTA VELOCITY OBTAINED               =  .41343+04 (FT/SEC)  WEIGHT OF PROPELLANT USED         =  .80738+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .51920-00 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER=  .27516+01 ( LBS )
        OPERATING PRESSURE                    =  .20000+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED      =  .14222+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED               =  .50500.02 (FT/SEC)  WEIGHT OF PROPELLANT USED         =  .70507+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .45978-02 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER=  .32049-01 ( LBS )
        OPERATING PRESSURE                    =  .20000.02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED      =  .21546+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SEC)  WEIGHT OF PROPELLANT USED         =  .30484+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .37899-02 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER=  .11186-01 ( LBS )
        OPERATING PRESSURE                    =  .31770+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED      =  .73086+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED               =  .46205+04 (FT/SEC)  WEIGHT OF PROPELLANT USED         =  .98916+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .61121+01 (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER=  .31621+01 ( LBS )
        OPERATING PRESSURE                    =  .32975+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED      =  .20339+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED               =  .33197+03 (FT/SEC)  WEIGHT OF PROPELLANT USED         =  .38008+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)   WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .29227+02 ( PSI )   TOTAL WEIGHT OF MIXTURE USED      =  .83346+02 ( LBS )
```

Table C-20b

FLOX/CH_4 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR CH_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =      .500 (INCHES)
        LAUNCH WEIGHT                           = .70297+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .28150+01 (  %  )
        VENT PRESSURE                           = .67310+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .30000+00 ( LBS )
        MAXIMUM TANK PRESSURE                   = .13000+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .32644+03 ( LBS )

MISSION PARTICULARS
        $ BURN MISSION                                      CALCULATION TIME          =  97.187   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .19838+03 (DEG-R)

WEIGHT SUMMARY
    OPTIMUM TANK(S)    =  2  [CH4] FITANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .24180+03 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .35427+02 ( PSI )
        PROPELLANT TANK                         = .62367+02 ( LBS )     PROPELLANT TANK SURFACE AREA      = .26833+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .13070+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .19613+01 (FEET )     RESIDUAL                          = .51840+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .12448+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .27188+00 ( LBS )     INSULATION                        = .25718+01 ( LBS )
    PRESSURANT ANALYSIS, PRESSURANT # =     17  HE  PRESSURANT SPHERE, INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .38782+00 (FEET )     PRESSURANT SPHERE WEIGHT          = .26673+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .29432+00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .28128+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .99998+01
    NONOPTIMUM TANK(S)    =  2   FLOX TANK(S)
        PROPELLANT TANK VOLUME                  = .19578+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .35128+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16720+01 (FEET )     INSULATION                        = .92081+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .17187+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .12462+00 ( LBS )
        PROPELLANT TANK                         = .66708+02 ( LBS )     RESIDUAL                          = .27216+02 ( LBS )
    TOTALS
        PAYLOAD                                 = .21200+04 ( LBS )     LAUNCH WEIGHT                     = .70297+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 S PERFORMANCE CONTINGENCY       = .25913+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .71195+04
    BURN #   1
        DELTA VELOCITY OBTAINED                 = .41339+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .15378+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .78210+00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .62960-00 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .19218+04 ( LBS )

    BURN #   2
        DELTA VELOCITY OBTAINED                 = .50600+02 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .17236+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .90191-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .71881-02 ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .21547+02 ( LBS )

    BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .58448-00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .34294-02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .27546-02 ( LBS )
        OPERATING PRESSURE                      = .20100+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .73058+01 ( LBS )

    BURN #   4
        DELTA VELOCITY OBTAINED                 = .46205+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .16246+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .49649+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .97329-00 ( LBS )
        OPERATING PRESSURE                      = .20138+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .20336+04 ( LBS )

    BURN #   5
        DELTA VELOCITY OBTAINED                 = .33313+03 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .67008+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .20000+02 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .63758+02 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  29 APR 69
        PROPELLANTS 1        CH4(P), FLOX
        USABLE WEIGHT        .70428+04
      STRUCTURE                                                          .17728+03
        BASE STRUCTURE                                       .99300+02
        TANK SUPPORTS                                        .99900+02
        ATTACHMENTS                                          .15000+02
        BULKHEAD INSULATION (1**)                            .18000+02
      PROPELLANT FEED ASSEMBLY                                           .38226+03
        TANKS                                                .25815+03
        VALVES,FILTERS,PLUMBING,ULLAGING                     .38500+02
        INSULATION (FIXED AND VARIABLE)                      .33559+02
        METEOROID BUMPER                                     .52047+02
      PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           .25514+02
      ENGINE SYSTEM                                                      .98000+02
      INERT SUB-TOTAL                                                    .68297+03
      CONTINGENCY 10%                                                    .68297+02
      RESIDUALS                                                          .90818+02
        PROPELLANT                                           .64000+02
        VAPOR                                                .23081+02
        HE GAS                                               .26649+01
      PERFORMANCE RESERVE (10 AV)                                        .25913+02
    IMPULSE PROPELLANTS                                                                .70428+04
    PROPULSION MODULE WEIGHT                                                           .49097+03
    PAYLOAD WEIGHT                                                                     .21200+04
```

Table C-21a

OF_2/B_2H_6 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR OF_2 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHES)
        LAUNCH WEIGHT                           =  .70404+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        =  .80921+01 (  %  )
        VENT PRESSURE                           =  .71660+03 ( PSI )
        INITIAL PRESSURANT LOAD                 =  .34000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   =  .24100+03 ( PSI )
        INITIAL PROPELLANT LOAD                 =  .15557+04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                                  CALCULATION TIME                =   06.083   ( SEC )
        FINAL PROPELLANT TEMPERATURE            =  .24986+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  OF2 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            =  .21098+03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .21605+03 ( PSI )
        PROPELLANT TANK                         =  .69270+02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .35246+02 (FT-2 )
        PROPELLANT TANK VOLUME                  =  .19676+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16797+01 (FEET )    RESIDUAL                        =  .23325+02 ( LBS )
        BOIL OFF (IF ANY)                       =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .12829+02 ( LBS )
        PRESSURANT IN PROPELLANT TANK           =  .33333+01 ( LBS )    INSULATION                      =  .33777+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17 -   HE  PRESSURANT SPHERE  INSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                =  .23399+00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .10600+02 ( LBS )
        PRESSURANT SPHERE VOLUME                =  .16564+01 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR  =  .46651-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     =  .29181+01
   NONOPTIMUM TANK(S)    =  2   B2H6 TANK(S)
        PROPELLANT TANK VOLUME                  =  .19290+02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .34723+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     =  .16623+01 (FEET )    INSULATION                      =  .33276+01 ( LBS )
        INITIAL PROPELLANT LOAD                 =  .51490+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .92778-01 ( LBS )
        PROPELLANT TANK                         =  .64496+02 ( LBS )    RESIDUAL                        =  .77750+01 ( LBS )
   TOTALS
        PAYLOAD                                 =  .19451+04 ( LBS )    LAUNCH WEIGHT                   =  .70404+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY     =  .25695+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .71195+05
   BURN #    1
        DELTA VELOCITY OBTAINED                 =  .41799+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .73559+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .77418-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26047+01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .19416+04 ( LBS )

   BURN #    2
        DELTA VELOCITY OBTAINED                 =  .80500+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .82334+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .99357-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .31601-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .21956+02 ( LBS )

   BURN #    3
        DELTA VELOCITY OBTAINED                 =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .27912+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .79396-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .26833-01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .74933+01 ( LBS )

   BURN #    4
        DELTA VELOCITY OBTAINED                 =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .76592+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .79586-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .35704+01 ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .20411+04 ( LBS )

   BURN #    5
        DELTA VELOCITY OBTAINED                 =  .33166+03 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .31485+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                      =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .84388+02 ( LBS )
```

Table C-21b

OF_2/B_2H_6 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR B_2H_6 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .000 (INCHES)
        LAUNCH WEIGHT                           = .70400+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20311+01 ( % )
        VENT PRESSURE                           = .18100+03 ( PSI )
        INITIAL PRESSURANT LOAD                 = .40000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .16500+03 ( PSI )
        INITIAL PROPELLANT LOAD                 = .51470+03 ( LBS )

MISSION PARTICULARS
        2 BURN MISSION                                          CALCULATION TIME                  =  86.720   ( DPC )
        FINAL PROPELLANT TEMPERATURE            = .27252+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2   B2H6 TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21254+03 ( PSI )     MAX PRESSURE FOR THIS MISSION     = .16503+03 ( PSI )
        PROPELLANT TANK                         = .44474+03 ( LBS )     PROPELLANT TANK SURFACE AREA      = .34733+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .19757+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16633+01 (FEET )     RESIDUAL                          = .77710+01 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .36937+00 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .19741+01 ( LBS )     INSULATION                        = .33276+01 ( LBS )
   PRESSURANT ANALYSIS, PRESSURANT # =     17   HE   PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .73394+00 (FEET )     PRESSURANT SPHERE WEIGHT          = .38608+01 ( LBS )
        PRESSURANT SPHERE VOLUME                = .16864+01 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .35863+01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUAL)      = .20853+01
   NONOPTIMUM TANK(S)   =  2   OPTIC1TANK(S)
        PROPELLANT TANK VOLUME                  = .19674+02 (FT-3 )     PROPELLANT TANK SURFACE AREA      = .35296+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16393+01 (FEET )     INSULATION                        = .32272+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15657+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .70644+01 ( LBS )
        PROPELLANT TANK                         = .69370+02 ( LBS )     RESIDUAL                          = .23326+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .19451+04 ( LBS )     LAUNCH WEIGHT                     = .70400+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1.0 PERFORMANCE CONTINGENCY       = .20280+03 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .71196+05
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .31759+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .31520+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .08434+00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .61848+00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .19616+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50000+02 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .27445+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .10212+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .10141+01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .21964+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .72853+00 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .11440+02 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .10853+01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .94434+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .44706+04 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .25519+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .20241+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .21347+00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .26411+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .33172+03 (FT/SEC)    WEIGHT OF PROPELLANT USED         = .10550+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED      = .04484+02 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                        BY JOHN AV
      PROPELLANTS :        OF2 C1, B2H6
      USABLE WEIGHT          .40956+04
    STRUCTURE                                                                   .17000+03
      BASE STRUCTURE                                          .95000+02
      TANK SUPPORTS                                           .50000+02
      ATTACHMENTS                                             .15000+02
      BULKHEAD INSULATION (1**)                               .10000+02
    PROPELLANT FEED ASSEMBLY                                                    .40641+03
      TANKS                                                   .27153+03
      VALVES,FILTERS,PLUMBING,ULLAGING                        .36500+02
      INSULATION (FIXED AND VARIABLE)                         .33911+02
      METEOROID BUMPER                                        .63972+02
    PRESSURIZATION SYSTEM (PLUMBING + TANKS)                                    .57101+02
    ENGINE SYSTEM                                                               .15500+03
    INERT SUB-TOTAL                                                             .77481+03
    CONTINGENCY 10%                                                             .77481+02
    RESIDUALS                                                                   .10847+03
      PROPELLANT                                              .62200+02
      VAPOR                                                   .25823+02
      HE GAS                                                  .16943+02
    PERFORMANCE RESERVE (10.0%)                                                 .25849+02
  IMPULSE PROPELLANTS                                                                       .40956+04
  PROPULSION MODULE WEIGHT                                                                  .80952+04
  PAYLOAD WEIGHT                                                                            .19451+04
```

Table C-22a

N_2O_4/A-50 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR N_2O_4 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                  =        .500 (INCHES)
        LAUNCH WEIGHT                         =  .72250+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME      =  .20068+01 (  %  )
        VENT PRESSURE                         =  .14700+04 ( PSI )
        INITIAL PRESSURANT LOAD               =  .20000+01 ( LBS )
        MAXIMUM TANK PRESSURE                 =  .17500+03 ( PSI )
        INITIAL PROPELLANT LOAD               =  .15089+04 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME          =  15.879   ( SEC )
        FINAL PROPELLANT TEMPERATURE          =  .48324+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)   =  2  [N2O4] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE          =  .21986+03 ( PSI )    MAX PRESSURE FOR THIS MISSION   =  .16592+03 ( PSI )
        PROPELLANT TANK                       =  .65311+02 ( LBS )    PROPELLANT TANK SURFACE AREA    =  .32459+02 (FT-2 )
        PROPELLANT TANK VOLUME                =  .17389+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16072+01 (FEET )    RESIDUAL                        =  .24954+02 ( LBS )
        BOIL OFF (IF ANY)                     =  .00000    ( LBS )    CALCULATED RESIDUAL PROPELLANT VAPOR=  .11402+01 ( LBS )
        PRESSURANT IN PROPELLANT TANK         =  .19916+01 ( LBS )    INSULATION                      =  .31106+01 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # -    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS              =  .60979-00 (FEET )    PRESSURANT SPHERE WEIGHT        =  .15674+02 ( LBS )
        PRESSURANT SPHERE VOLUME              =  .94979-00 (FT-3 )    PRESSURANT REMAINING IN PRESS SPHR =  .84199-02 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)   =  .24997+01
   NONOPTIMUM TANK(S)   =  2   A-50 TANK(S)
        PROPELLANT TANK VOLUME                =  .18287+02 (FT-3 )    PROPELLANT TANK SURFACE AREA    =  .33567+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.   =  .16344+01 (FEET )    INSULATION                      =  .32188+01 ( LBS )
        INITIAL PROPELLANT LOAD               =  .94270+03 ( LBS )    CALCULATED INITIAL PROPELLANT VAPOR =  .33096-01 ( LBS )
        PROPELLANT TANK                       =  .65891+02 ( LBS )    RESIDUAL                        =  .15596+02 ( LBS )
   TOTALS
        PAYLOAD                               =  .13240+04 ( LBS )    LAUNCH WEIGHT                   =  .72250+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)      =  .00000    ( LBS )    1 % PERFORMANCE CONTINGENCY     =  .28562+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .11327+05
   BURN #  1
        DELTA VELOCITY OBTAINED               =  .43067+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .77704+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .74359-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .20242+01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .25254+04 ( LBS )

   BURN #  2
        DELTA VELOCITY OBTAINED               =  .50500+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .77091+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.10757-02 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .22441-01 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .25347+02 ( LBS )

   BURN #  3
        DELTA VELOCITY OBTAINED               =  .17170+02 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .26421+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE          = -.32326-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .56207-02 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .85468+01 ( LBS )

   BURN #  4
        DELTA VELOCITY OBTAINED               =  .66205+04 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .60444+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .70737-00 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .98180-00 ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .22569+04 ( LBS )

   BURN #  5
        DELTA VELOCITY OBTAINED               =  .35254+03 (FT/SC)    WEIGHT OF PROPELLANT USED       =  .25930+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE          =  .00000    (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER=  .00000    ( LBS )
        OPERATING PRESSURE                    =  .15500+03 ( PSI )    TOTAL WEIGHT OF MIXTURE USED    =  .84272+02 ( LBS )
```

Table C-22b

N_2O_4/A-50 MARS ORBITER WITH TITAN IIID ONLY – OPTIMIZATION FOR A-50 SYSTEM

```
PARAMETERS BEING OPTIMIZED
        INSULATION THICKNESS                    =        .500 (INCHS)
        LAUNCH WEIGHT                           = .7226o+04 ( LBS )
        ULLAGE PERCENTAGE OF TANK VOLUME        = .20165+01 (  %  )
        VENT PRESSURE                           = .16960+04 ( PSI )
        INITIAL PRESSURANT LOAD                 = .22000+01 ( LBS )
        MAXIMUM TANK PRESSURE                   = .75000+02 ( PSI )
        INITIAL PROPELLANT LOAD                 = .94260+03 ( LBS )

MISSION PARTICULARS
        5 BURN MISSION                                          CALCULATION TIME                = 89.849   ( SEC )
        FINAL PROPELLANT TEMPERATURE            = .48213+03 (DEG-R)

WEIGHT SUMMARY
   OPTIMUM TANK(S)    =  2  [A-50] TANK(S)
        MINIMUM GAGE CUTOFF PRESSURE            = .21619+03 ( PSI )     MAX PRESSURE FOR THIS MISSION   = .16551+03 ( PSI )
        PROPELLANT TANK                         = .65891+02 ( LBS )     PROPELLANT TANK SURFACE AREA    = .33567+02 (FT-2 )
        PROPELLANT TANK VOLUME                  = .18287+02 (FT-3 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16344+01 (FEET )     RESIDUAL                        = .15596+02 ( LBS )
        BOIL OFF (IF ANY)                       = .00000    ( LBS )     CALCULATED RESIDUAL PROPELLANT VAPOR= .97324-01 ( LBS )
        PRESSURANT IN PROPELLANT TANK           = .21422+01 ( LBS )     INSULATION                      = .32168+01 ( LBS )
   PRESSURANT ANALYSIS: PRESSURANT # =    17 -   HE  PRESSURANT SPHERE OUTSIDE PROPELLANT TANK
        PRESSURANT SPHERE RADIUS                = .61869-00 (FEET )     PRESSURANT SPHERE WEIGHT        = .16371+02 ( LBS )
        PRESSURANT SPHERE VOLUME                = .99200-00 (FT-3 )     PRESSURANT REMAINING IN PRESS SPHR = .57768-01 ( LBS )
        GAS FACTOR (OTHER TANK + RESIDUALS)     = .23735+01
   NONOPTIMUM TANK(S)    =  2   N2O4 TANK(S)
        PROPELLANT TANK VOLUME                  = .17389+02 (FT-3 )     PROPELLANT TANK SURFACE AREA    = .32459+02 (FT-2 )
        PROPELLANT TANK CHARACTERISTIC DIM.     = .16072+01 (FEET )     INSULATION                      = .31106+01 ( LBS )
        INITIAL PROPELLANT LOAD                 = .15390+04 ( LBS )     CALCULATED INITIAL PROPELLANT VAPOR = .82856-01 ( LBS )
        PROPELLANT TANK                         = .65311+02 ( LBS )     RESIDUAL                        = .24954+02 ( LBS )
   TOTALS
        PAYLOAD                                 = .13215+04 ( LBS )     LAUNCH WEIGHT                   = .72260+04 ( LBS )
        STRUCTURAL (VARIABLE + CONSTANT)        = .00000    ( LBS )     1 % PERFORMANCE CONTINGENCY     = .26403+02 ( LBS )

SUMMARY OF BURN DATA - TOTAL DELTA VELOCITY = .11328+05
   BURN #   1
        DELTA VELOCITY OBTAINED                 = .43375+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .48579+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .84794-00 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .80970-00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .25261+04 ( LBS )

   BURN #   2
        DELTA VELOCITY OBTAINED                 = .50500+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .48747+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .44120+01 (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= -.48382-01 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .25349+02 ( LBS )

   BURN #   3
        DELTA VELOCITY OBTAINED                 = .17170+02 (FT/SC)     WEIGHT OF PROPELLANT USED       = .16514+01 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.64560-01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .60966-03 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .85873+01 ( LBS )

   BURN #   4
        DELTA VELOCITY OBTAINED                 = .66205+04 (FT/SC)     WEIGHT OF PROPELLANT USED       = .43405+03 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = -.24881+01 (DEG-R)    WEIGHT OF PROPELLANT VAPORIZED AFTER= .10733+00 ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .22571+04 ( LBS )

   BURN #   5
        DELTA VELOCITY OBTAINED                 = .33189+03 (FT/SC)     WEIGHT OF PROPELLANT USED       = .16176+02 ( LBS )
        DELTA PROPELLANT TEMPERATURE            = .00000    (DEG-R)     WEIGHT OF PROPELLANT VAPORIZED AFTER= .00000    ( LBS )
        OPERATING PRESSURE                      = .15500+03 ( PSI )     TOTAL WEIGHT OF MIXTURE USED    = .84115+02 ( LBS )
```

WEIGHT SUMMARY

```
                                                                                  02 JUN 69
   PROPELLANTS :        [N2O4] + A-50
   USABLE WEIGHT        .48739+04
 STRUCTURE                                                          .21031+03
   BASE STRUCTURE                                   .11119+03
   TANK SUPPORTS                                    .66118+02
   ATTACHMENTS                                      .15000+02
   BULKHEAD INSULATION (1**)                        .18000+02
 PROPELLANT FEED ASSEMBLY                                           .37984+03
   TANKS                                            .26240+03
   VALVES+FILTERS+PLUMBING+ULLAGING                 .28000+02
   INSULATION (FIXED AND VARIABLE)                  .32655+02
   METEOROID BUMPER                                 .56782+02
 PRESSURIZATION SYSTEM (PLUMBING + TANKS)                           ~~.83487+03~~  78.716
 ENGINE SYSTEM                                                      .15100+03
 INERT SUB-TOTAL                                                    ~~.52736+03~~  819.4
 CONTINGENCY 10%                                                    .82405+02
 RESIDUALS                                                          .94042+02
   PROPELLANT                                       .81100+02
   VAPOR                                            .29429+01
   HE GAS                                           .99990+01
 PERFORMANCE RESERVE (1% ΔV)                                        .26562+02
IMPULSE PROPELLANTS                                                              .48739+04
PROPULSION MODULE WEIGHT                                                         ~~.90010+04~~  5057
PAYLOAD WEIGHT                                                                   ~~.13200+04~~  1128
```

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