K-19-68-6 • 30 AUGUST 1968

FINAL REPORT

PROPELLANT SELECTION FOR SPACECRAFT PROPULSION SYSTEMS

CONTRACT NASw-1644

VOLUME I

RESULTS, CONCLUSIONS, AND RECOMMENDATIONS

PREPARED FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
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FOREWORD

This report was prepared by the Lockheed Missiles & 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 Contract NASw-1644, Propellant Selection for Spacecraft Propulsion Systems. The report is printed in three volumes:

- Volume I – Results, Conclusions, and Recommendations
- Volume II – Missions and Vehicles
- Volume III – Thermodynamics and Propulsion

ACKNOWLEDGMENTS

The Space Storable Propellant Selection study was performed under the direction of a NASA Management Committee comprising the following individuals:

Jack A. Suddreth, Committee Chairman, NASA Headquarters
Don Nored, Lewis Research Center
Robert Breshears, Jet Propulsion Laboratory
Robert Polifka, Manned Spacecraft Center
Keith Coates, Marshall Space Flight Center

Additional NASA participants were

Robert Levine, NASA Headquarters
Charles Yodzis, Manned Spacecraft Center
Clarke Covington, Manned Spacecraft Center
Sid Saucier, Marshall Space Flight Center
Ed Gomersall, Ames Research Center, Mission Analysis Division
C. Bendersky, Bellcomm
A. Marks, Bellcomm

Propulsion company support was provided in the form of data related to characteristics and properties of the candidate propellants and to performance characteristics of rocket engines utilizing these propellants. Three companies, Aerojet-General Corporation, Pratt & Whitney Aircraft, and Rocketdyne, provided close support throughout the study, first with extensive parametric data and later with point designs corresponding to the propulsion requirements of stages selected for detailed analysis in Task II – Stage Investigations. This support was vital to providing current, realistic data for use in the analysis.

Lockheed Missiles & Space Company personnel responsible for major contributions to the study included:

James E. Piper, Study Manager
Frank M. Friedlaender, Assistant Study Manager and Systems Integration
Don D. Smith, Technical Leader, Thermodynamics/Pressurization
Yukio Yoshikawa, Thermodynamic Analysis
B. D. Neff, Pressurization Systems Analysis
Dave Morton, Thermodynamic Analysis
John Gries, Thermodynamic Analysis
Sheldon Kay, Thermodynamic Analysis
Karl Tollo, Design and Structures
Mark Tyson, Design and Structures
Jack Hief, Propulsion
Jack Elias, Propulsion
Sid Everett, Propulsion
Robert Gorman, Ground Operations
Paul Van Zytveld, Flight Mechanics

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Section 1
SUMMARY

A detailed comparison of the storability and performance of high-energy liquid propellants was performed for a wide range of near-earth and interplanetary missions. Emphasis was placed on detailed thermodynamic, pressurization, and structural analyses of representative vehicle systems. Preliminary analysis was conducted on the broad range of missions, and was followed by a refined analysis of two systems. The two systems investigated in depth were the propulsion stage for an unmanned Mars Orbiter and the ascent stage for a manned Mars Excursion Module (MEM) with fixed payloads. Specific propellants investigated were F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/CH_4, F_2/NH_3, OF_2/B_2H_6, ClF_5/MHF-5, and N_2O_4/A-50.

Major conclusions of this study are as follows:

- All candidate propellants can be stored nonvented for the Mars Orbiter and MEM missions
- Space storables yield propulsion vehicle systems 20 percent or more lighter than earth storables for the missions analyzed
- Fluorine/hydrogen yields the lightest weight pump-fed propulsion vehicle system (approximately 25 to 30 percent lighter than earth storables)
- OF_2/B_2H_6 yields the lightest weight pressure-fed propulsion vehicle system (approximately 20 percent lighter than earth storables)
- All of the propellants are relatively insensitive to variations in mission duration, propellant tank surface coating, insulation degradation, and spacecraft orientation. H_2 is affected more than space storables.

Section 2
INTRODUCTION

This report presents the results of work performed by the Lockheed Missiles & Space Company for Headquarters, NASA-OART, under Contract NASw-1644 during the period 5 July 1967 to 5 July 1968. The Contract was directed by a Management Committee representing NASA Headquarters, Lewis Research Center, Marshall Space Flight Center, Manned Space Center, and the Jet Propulsion Laboratory.

For the purposes of this study, three general classes of liquid propellants were defined as follows:

- Earth storable: In the liquid state at earth ambient temperatures and pressures
- Cryogenic or deep cryogenic: Propellants using liquid hydrogen as the fuel
- Space storable: In the liquid state at temperatures below earth ambient but higher than liquid hydrogen

The study was directed toward evaluating the performance of space-storable propellants in comparison with earth-storables and cryogens and was biased toward identifying technology development requirements related to thermal storage of propellants having attractive performance. Missions and stages analyzed were selected as items of convenience, and were not intended to establish configurations for any specific NASA program requirement.

2.1 STUDY APPROACH

The work was performed in three phases: The first phase (Task I) consisted of a mission investigation to identify potential applications for space-storable propellants and to identify attractive candidates for a detailed stage investigation; the second phase (Task II) consisted of a detailed thermal, structural, and operational analysis of two propulsion stages selected by NASA using alternate earth-storable, space-storable, and cryogenic propellants with emphasis on investigation of propellant storage requirements; the third phase (Task III) included application of the results of phases one and two in developing propellant selection factors in identifying areas of propulsion systems commonality, and in identifying major problem areas and technology development requirements resulting from the use of space-storable propellants. A summary of the study approach is presented in Fig. 1. Assumptions used for the preliminary (Task I) and refined (Task II) basic analyses are presented in Table 1.

Table 1

BASIC ANALYSIS ASSUMPTIONS

<table>
<tr><th>Parameter</th><th>Preliminary Analysis Assumptions</th><th>Refined Analysis Assumptions</th></tr>
<tr><td>Missions
Stage Inert Weight
Engine Weight
Tank Heat Input
Stage Optimization</td><td>Mission Spectrum
Scaling Law Weights
Scaling Law Weights
Converted to Boiloff
Boiloff and Insulation Only</td><td>Mars Orbiter and MEM
Calculated Weights
Engine Company Data
Considered in Optimization
Full System Optimization</td></tr>
<tr><td>Superinsulation Conductivity</td><td colspan="2">$H_2 - 2.5 \times 10^{-5}$ Btu/hr-ft-°R
O_2, FLOX, OF_2, F_2, $CH_4 - 5.0 \times 10^{-5}$ Btu/hr-ft-°R
$NH_3 - 10.0 \times 10^{-5}$</td></tr>
<tr><td>Surface Coatings (Nominal)</td><td colspan="2">H_2, O_2, F_2, FLOX, CH_4, OF_2 – Optical Solar Reflector, $\alpha_s/\epsilon = 0.05/0.80$
NH_3 – Degraded White Thermatrol, $\alpha_s/\epsilon = 0.30/0.95$
N_2O_4/A-50, ClF_5, MHF-5 – Degraded White Skyspar, $\alpha_s/\epsilon = 0.60/0.91$</td></tr>
</table>

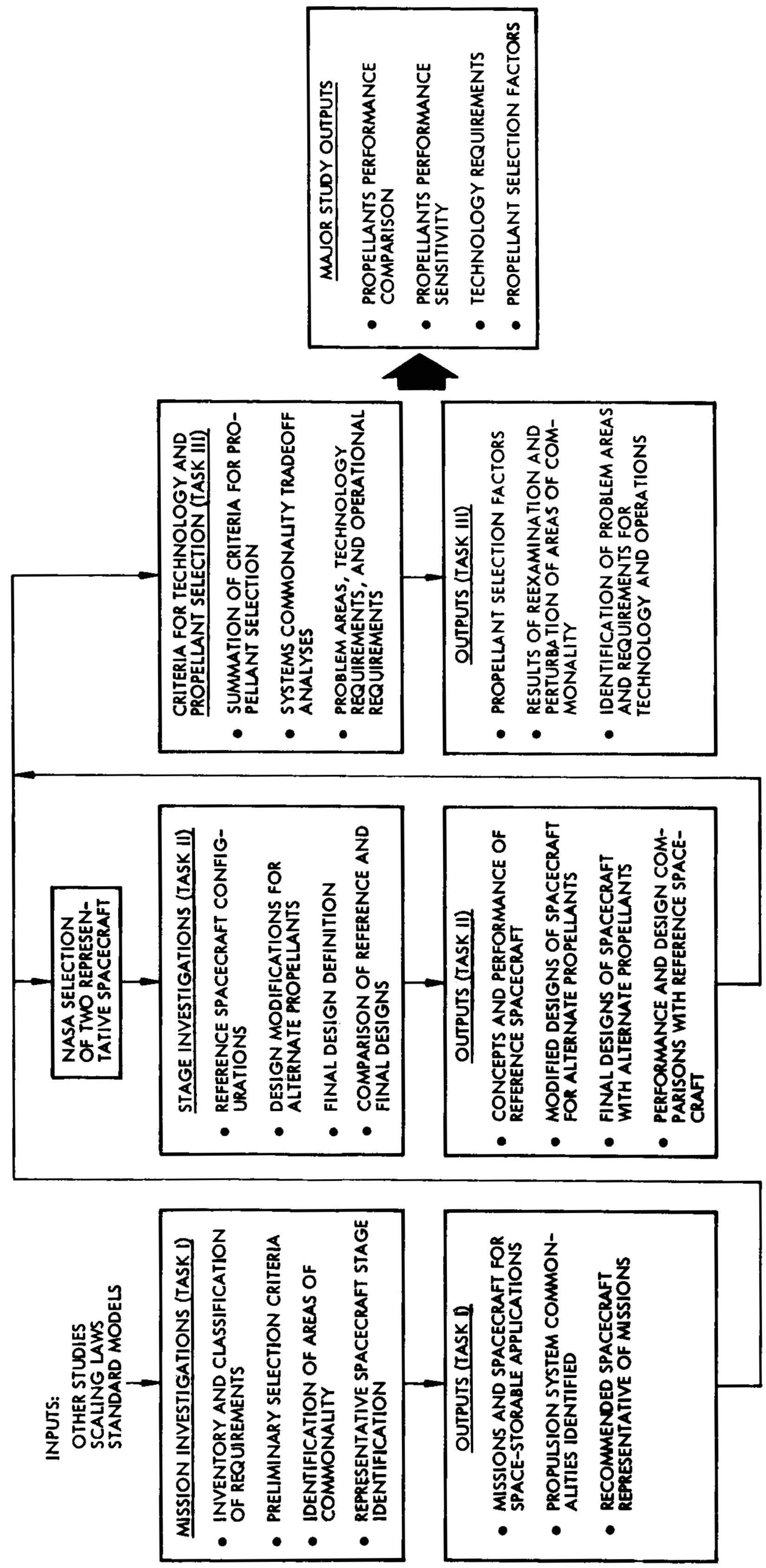


Fig. 1 Summary of Study Approach

2.2 STUDY GROUND RULES AND CONSTRAINTS

The following ground rules were used in the study:

- To achieve a common technical basis, as well as to expedite the analysis, the following scaling relationships, environmental models, and parametric mission models were used in Task I, Mission Analysis:
  - Scaling laws for propellant tank inert weights and any desirable propellant density
  - Thermal and micrometeorite environmental models
  - Thermal insulation and boiloff penalties
  - Meteoroid shielding penalties
  - Characteristic velocity requirements as a function of trip times
- The propellants used in Task I were representative of the cryogenics, space storables, and earth storables. Specifically, they were F_2/H_2, O_2/H_2, $FLOX/CH_4$, F_2/NH_3, and N_2O_4/A-50.
- Evaluation procedures used in Task II included realistic calculations of heat leak through insulation, radiation interchange between surfaces, and conducting through insulation penetrations, down supports, and through the structure.

The following study constraints were specified by NASA:

- The space-storable propellants to be considered to the extent necessary in Task II include:

  OF_2/CH_4
  $FLOX/CH_4$ (82.5 Percent F_2/17.5 Percent O_2)
  OF_2/B_2H_6 (Mars Orbiter Only)
  F_2/NH_3
  O_2/H_2 and Subcooled O_2/H_2
  F_2/H_2
  N_2O_4/A-50 (Mars Orbiter Only)
  ClF_5/MHF-5

- Only pressure-fed engine systems are considered for OF_2/B_2H_6. Both pressure-fed and pump-fed engine systems (regeneratively cooled, ablatively cooled, or transpiration cooled) may be considered for the other fuels.
- All propulsion systems are assumed to have equal reliability.
- Sterilization criteria are not considered.
- Mission parameters and nonpropulsive portions of spacecraft vehicles are not studied in detail; these items are adopted from the original study for the various missions.
- Use of refrigeration systems is not considered.

2.3 LIMITATIONS OF ANALYSIS

The following limitations of the analyses are presented as an aid in understanding the study results:

- The study was propulsion-technology oriented.
- The analyses were heavily weighted toward propellant storage requirements, detailed thermal/structural/pressurization optimization, and performance comparisons.
- All designs featured separate fuel and oxidizer tanks. No analyses were made of tanks with common bulkheads.
- Oxidizer and fuel tanks were optimized separately as to pressure, tank weight, and insulation weight. This implies a need for mixture ratio control for pressure-fed systems.
- Subsystems, except for solar panels on the Mars Orbiter, were assumed separated from the propulsion module by a thermal barrier. Designs requiring integration of spacecraft equipment and propellant tankage would pose a more severe thermal storage problem for the space-storable and cryogenic propellants.
- Detailed analysis of propellant tank exposure was limited to extremes of exposure to the sun and complete shielding behind the Mars Orbiter capsule. Local shadow shielding was examined in lesser detail.

- No shadow shield studies were made for the MEM on the surface of Mars. Stage I tanks were assumed exposed to the sun.
- Launch-pad thermal control requirements and problems were not studied in detail.
- No general agreement could be found among rocket engine companies as to problems or merits of one propellant combination over another when used in an engine system, or as to the desirability and potential reliability of (1) pump-fed systems as compared to pressure-fed systems, and of (2) regenerative cooling versus transpiration cooling versus film and ablative cooling.
- No analysis was made of meteoroid shields discarded prior to stage firing since the stages analyzed required multiple firings.
- Analysis of differential boiloff of fluorine from FLOX was limited to ullage volume since fluorinated oxidizers were not vented.

Section 3
RESULTS

3.1 MISSION ANALYSIS – TASK I

During Task I, a systematic analysis was made in which a mission matrix was established, a preliminary screening was conducted, and the selected missions were then evaluated through a preliminary thermodynamic and propulsion system analysis. The evaluation procedure is illustrated in Fig. 2. Scaling laws were used to conduct the performance analysis, which was accomplished using a computer program entitled Rapid Analysis of Propellants for Initial Design (RAPID).

The mission analysis was conducted for a broad spectrum of space missions. All of the propulsive steps in each mission were considered, and those having a $\Delta V \geq 3,500$ ft/sec were selected for analysis since they provided the possibility of increasing the stage performance by 10 percent or more when space-storable propellants were substituted for the earth storable N_2O_4/A-50. The missions and stages selected for analysis are listed in Table 2, together with important mission parameters.

Representative cryogenic, space-storable, and earth-storable propellants were selected for the initial mission screening analysis. The propulsion assumptions used for Task I are shown in Table 3. These data were based on information available at the start of the study and were modified for Task II, as described in subsequent paragraphs. In Table 4, the propulsion stage initial weights for the various missions are normalized using N_2O_4/A-50 as the reference propellant. From Table 4 it is evident that, based on the scaling law analysis, a FLOX/CH_4 or F_2/NH_3 stage is from 20 to 45 percent lighter than the earth-storable stage to perform a specified mission. An O_2/H_2 stage is slightly heavier than FLOX/CH_4 for all but the large planet-departure stages, while F_2/H_2 results in the lightest stage for all but the long-duration

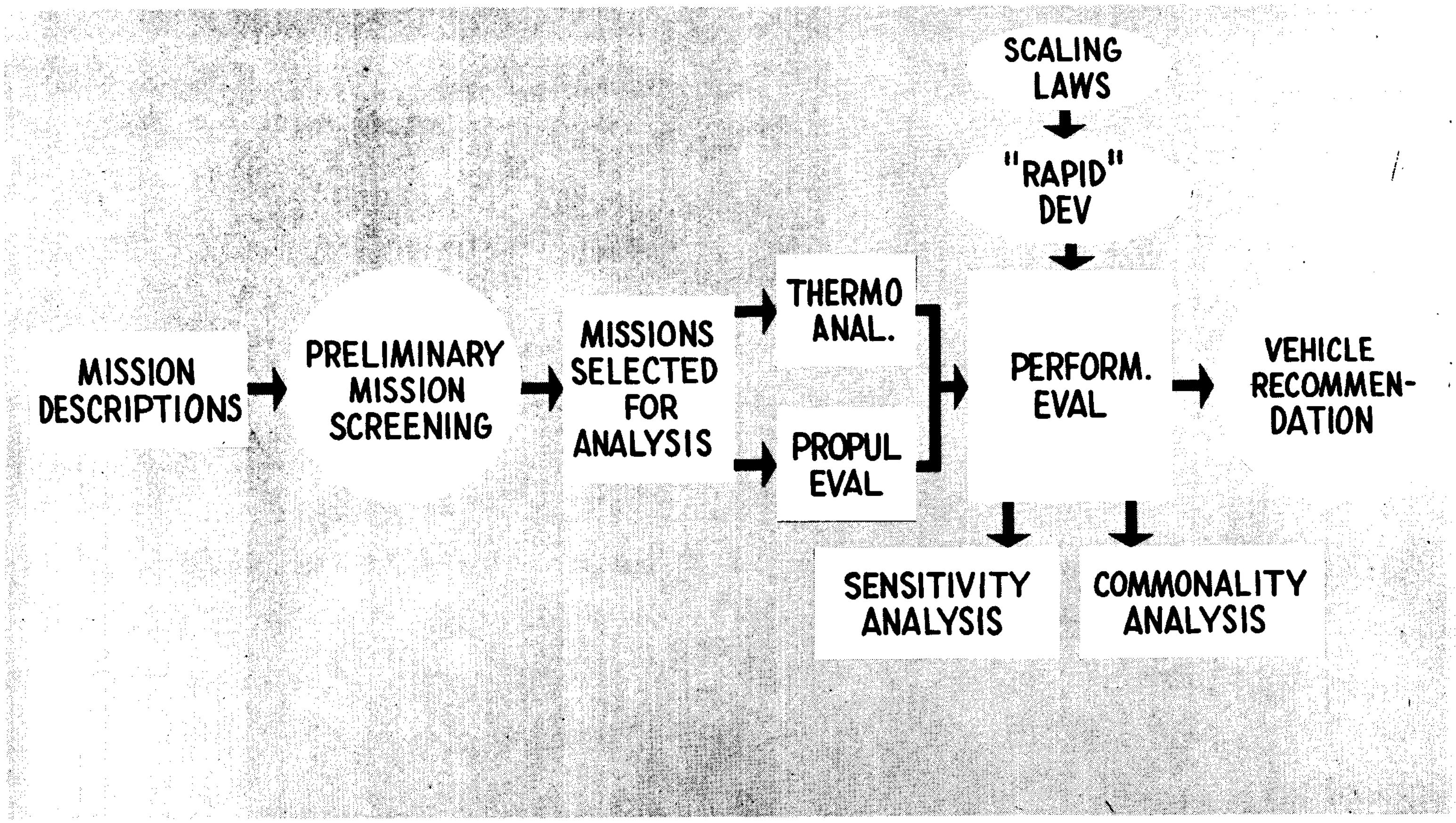


Fig. 2 Mission Evaluation Procedure

Table 2

SELECTED MISSIONS AND MISSION PARAMETERS

Mission	Payload (lb)	ΔV (ft/sec)	Trip Time (days)	Thrust (lb)
Saturn Unmanned Orbiter	2,000	6,000	1,450	2,000
Jupiter Unmanned Orbiter	2,000	7,600	650	2,000
Venus Manned Flyby – Orbiter Probe	1,500	13,000	115	4,000
Mars Manned Flyby – Orbiter Probe	1,000	21,000	150	4,000
Mars Manned Lander – MEM Ascent Stage	5,260	16,000	220	30,000
Venus Unmanned Orbiter	7,000	13,500	140	8,000
Mars Unmanned Orbiter	8,143	6,950	195	8,000
Lunar Manned Surface Station – Return Stage	19,340	9,186	178	15,000
Earth Manned Synchronous Orbiter – Descent Stage	13,000	9,750	120	20,000
Mars Manned Lander – Planet Departure Stage	92,000	16,000	220/300	100,000
Venus Manned Orbiter – Planet Departure Stage	92,000	14,000	173	100,000
Mars Manned Lander – Earth Departure Stage	110,000	12,900	60	100,000

Table 3

PROPULSION ASSUMPTIONS FOR TASK I

Parameter	Propellent				
	F_2/H_2	O_2/H_2	N_2O_4/A-50	FLOX/CH_4	F_2/NH_3
I_{sp} (sec)	461	446	310	405	407
Mixture Ratio (O/F)	9:1	5:1	1.6:1	5.75:1	3.2:1

Table 4

PROPELLANT PERFORMANCE BY MISSION – STAGE WEIGHT COMPARED WITH N_2O_4/A-50 STAGE

Mission	Stage Weight (% of N_2O_4/A-50)			
	F_2/H_2	O_2/H_2	FLOX/CH_4	F_2/NH_3
Saturn Unmanned Orbiter	95	107	80	79
Jupiter Unmanned Orbiter	81	92	76	75
Venus Manned Flyby – Orbiter Probe	63	71	67	66
Mars Manned Flyby – Orbiter Probe	50	58	55	54
Mars Manned Lander – MEM Ascent Stage	56	65	62	60
Venus Unmanned Orbiter	59	66	64	63
Mars Unmanned Orbiter	70	75	73	73
Lunar Manned Surface Station – Return Stage	66	73	71	69
Earth Manned Synchronous Orbiter – Descent Stage	62	68	68	67
Mars Manned Lander – Planet Departure Stage	53	59	61	60
Venus Manned Orbiter – Planet Departure Stage	54	60	62	61
Mars Manned Lander – Earth Departure Stage	55	60	64	63

missions to Saturn and Jupiter. The validity of the Task I scaling law analysis as a tool for preliminary evaluations was borne out by the detailed Task II analysis.

At the completion of the mission analysis task, the four stages listed in Table 5 were recommended for further analysis. The NASA Management Committee selected the Unmanned Mars Orbiter–Orbit Injection Stage as defined in the TRW Voyager studies and the Mars Excursion Module–Ascent Stage as defined by North American-Rockwell Corporation as the two stages to be evaluated in detail in the Stage II investigation phase of the study. These two missions and stages are described in the following paragraphs.

Table 5

STAGES RECOMMENDED FOR ANALYSIS

Stage	Thrust (lb)	Payload (lb)	ΔV (ft/sec)	Duration (days)
Unmanned Saturn Orbiter – Orbit Injection	2,000	2,000	6,000	1,450
Unmanned Mars Orbiter – Orbit Injection	8,000	8,143	6,950	195
Lunar Manned Station – Return Stage	15,000	19,340	9,186	178
Mars Excursion Module – Ascent Stage	30,000	5,260	16,000	220

3.1.1 Baseline Mars Orbiter

The baseline Mars Orbiter has an 8,000-lb-thrust propulsion system used to insert a spacecraft into an eccentric orbit about Mars. This vehicle is shown in Fig. 3. Nominal parameters for the mission and stage are as follows:

- Payload: 8,143 lb
- Mission Duration: 205 days
- ΔV Total: 6,950 ft/sec
  - 1st Midcourse: 164 ft/sec at 2 days
  - 2nd Midcourse: 164 ft/sec at 165 days
  - Orbit Insertion: 6,294 ft/sec at 195 days
  - Orbit Trim 328 ft/sec at 205 days
- Three-axis stabilization with propulsion system facing the sun

3.1.2 Baseline Mars Excursion Module

The baseline MEM ascent stage has a 30,000-lb-thrust pump-fed propulsion system used to return a four-man capsule from the surface of Mars to a 500-km circular

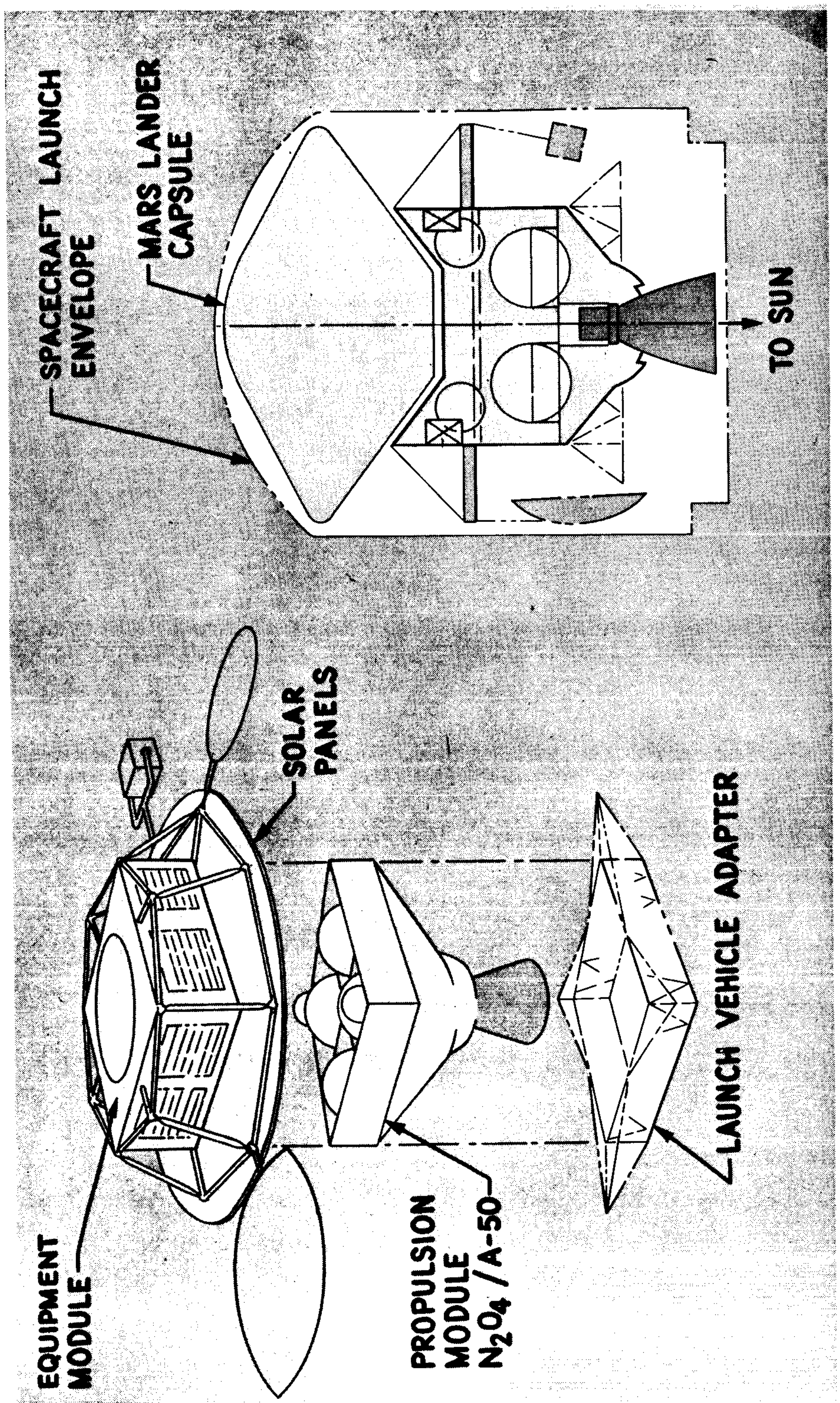


Fig. 3 Baseline Orbiter – TRW Voyager

orbit. This vehicle is shown in Fig. 4. Nominal parameters for this mission and stages are:

• Payload:	5,260 lb
• Mission Duration:	220 days
Earth orbit:	30 days
Enroute to Mars:	160 days
Mars surface:	30 days
• ΔV Total:	16,000 ft/sec*

3.2 PROPULSION DATA SELECTION

Propellant and propulsion systems data were obtained from the supporting engine companies for point designs with systems of 8,000 lb and 30,000 lb thrust. These thrust levels were specified in the Mars Orbiter and MEM Ascent Stage studies, respectively. These data were reviewed and selections made by Lockheed for use in Task II.

The nominal specific impulse values selected are considered to be realistically optimistic for the 1975 time period. Since the degree of optimism is believed to be uniform across the matrix of propellants, the comparison of one propellant with any other is not affected.

Propellant and propulsion systems characteristics assumed for the 8,000-lb-thrust Mars Orbiter stage and for the 30,000-lb-thrust Mars Excursion Module Ascent Stage are presented in Tables 6 and 7, respectively.

*16,000 ft/sec is the nominal mission, not the basic design mission assumed by North American-Rockwell.

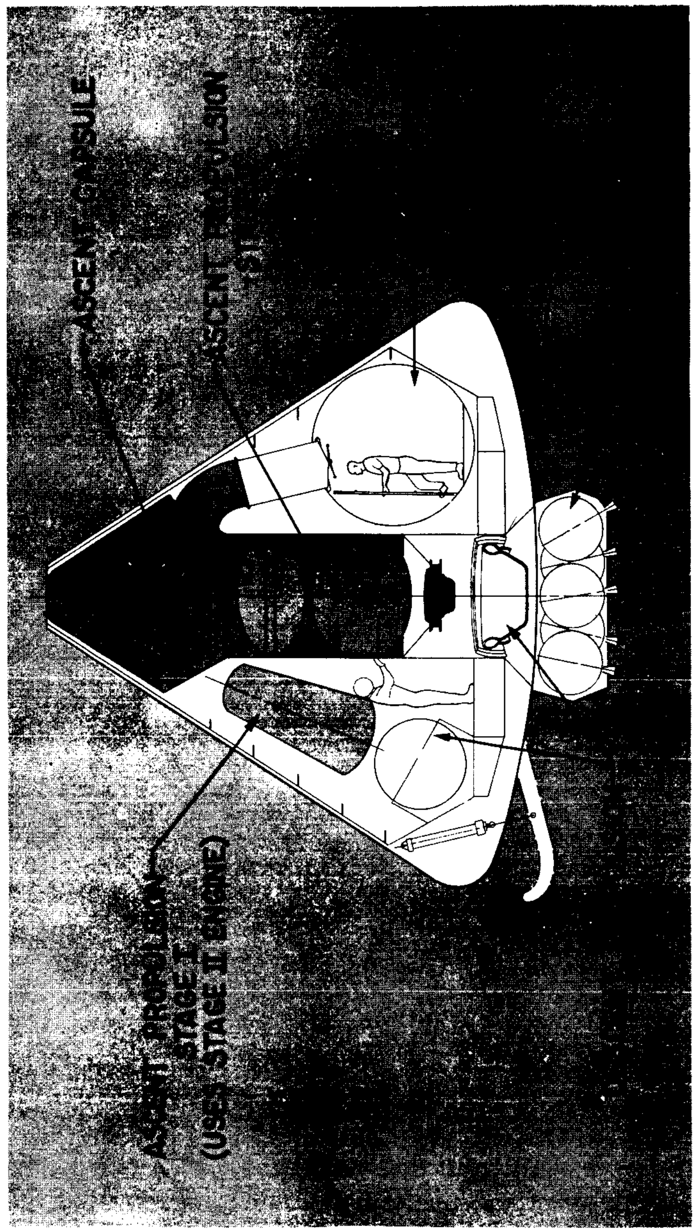


Fig. 4 Baseline Mars Excursion Module

Table 6

MARS ORBITER PROPULSION SYSTEM CHARACTERISTICS
(Bell Nozzle 8,000-lb Thrust)

Propellant	Mixture Ratio (O/F)	Pump-Fed Systems			Pressure-Fed Systems			Cooling
		Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	
F_2/H_2	13	900	468	173 (b)	100	442	375	Regenerative
O_2/H_2	6	900	451	173 (b)	100	445	380	Regenerative
FLOX/CH_2	5.75	600	410	152	100	387	375	Regenerative
OF_2/CH_4	5.3	600	410	152	100	396	380	Regenerative
OF_2/B_2H_6	3.82	-	-	-	100	414	384	Ablative
F_2/NH_3	3.3	1,500	408	192	100	386	375	Regenerative
N_2O_4/A-50	2.0	750	335	158	100	328	330	Ablative
ClF_5/MHF-5	2.4	750	342	167	100	330	384	Ablative

(a) Based on ϵ = 100
(b) Extendable bell nozzle

Table 7

MEM PROPULSION SYSTEM CHARACTERISTICS
(AEROSPIKE NOZZLE – 30,000-LB THRUST)

Propellant	Mixture Ratio (O/F)	Chamber Pressure (psia)	Expansion Ratio ϵ	I_{sp} (sec)	Engine Weight (lb)	Cooling
F_2/H_2	13	750	75	463	440	Regenerative
O_2/H_2	6	750	100	449	520	Regenerative
$FLOX/CH_4$	5.7	750	75	400	440	Regenerative
OF_2/CH_4	5.3	750	75	406	460	Regenerative
F_2/NH_3	3.3	750	75	397	440	Regenerative
ClF_5/MHF-5	2.4	750	100	336	475	Ablative

3.3 SYSTEMS DESCRIPTIONS

3.3.1 Propellants

The propellants of interest can be grouped into cryogens, space storables, and earth storables. The cryogens, using liquid hydrogen as fuel, are characterized by low bulk density, high specific impulse, and nonoverlapping liquidus range. Earth storables are in the liquid state at earth ambient temperatures, and are characterized by high bulk density, low specific impulse, and overlapping liquidus range. The space storables exhibit moderately high specific impulse, high bulk density, and may or may not have an overlapping liquidus range.

The propellants selected for the Task II vehicle analysis are shown in Fig. 5, together with the liquidus temperature range of each. The propellant OF_2/B_2H_6 was not analyzed for the MEM because only pump-fed systems were considered, and OF_2/B_2H_6 was not recommended for pump-fed application. N_2O_4/A-50 was also not considered

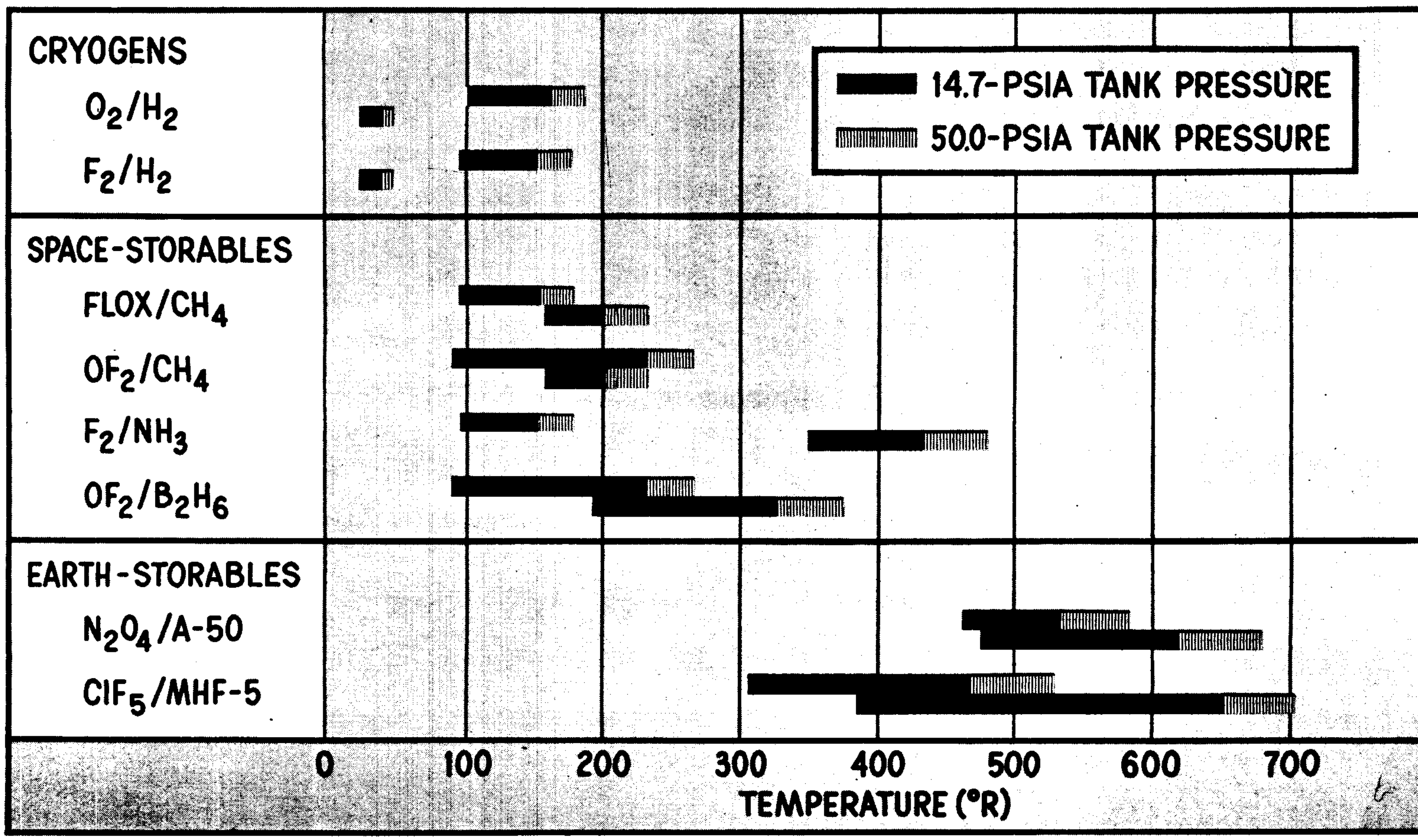


Fig. 5 Propellant Liquid Temperature Range

since the MEM was basically a $FLOX/CH_4$ system, and only the best performing earth storable, ClF_5/MHF-5, was used for comparison. For the Mars Orbiter vehicle, all of the propellants were evaluated for pressure-fed systems, and all but OF_2/B_2H_6 were evaluated for the pump-fed systems.

3.3.2 Engine Systems

For the purpose of this study, the engine systems were classified by geometry, engine feed, and cooling technique. The Mars Orbiter vehicle utilizes a bell-shaped nozzle with either a fixed skirt or an extendable skirt where that is an advantage. The MEM vehicle is of such a design that toroidal engines become mandatory if a single engine is used. In this study, Aerospike engines were assumed for the MEM application. For the Mars Orbiter, both pump-fed and pressure-fed engine systems were analyzed. For the MEM, only pump-fed systems were investigated since pressure-fed engines were larger than available space would accommodate. Regenerative cooling was assumed for all MEM engines except for those that use ClF_5/MHF-5. For the Mars Orbiter, all of the propellant combinations except OF_2/B_2H_6, N_2O_4/A-50, and ClF_5/MHF-5 have regeneratively cooled engines for both pump- and pressure-fed systems. OF_2/B_2H_6, N_2O_4/A-50, and ClF_5/MHF-5 were used only in ablatively cooled engines.

3.3.3 Engine Fluid Systems

A typical space-storable engine fluid system schematic is shown in Fig. 6. The operation of the system is described as follows.

Oxidizer and fuel loading is accomplished by supplying liquid from a ground source to the fill lines. Connection to the ground source is provided by quick-disconnect couplings. The tanks are back filled from the bottom after completion of purging and passivation. Purge gas is introduced through the fill lines and is vented by means of the ground-controlled vent valve. During ground hold, topping is continued as required, and vapors are vented via the ground control vent. The fill/drain line shutoff valves are open during prelaunch operations.

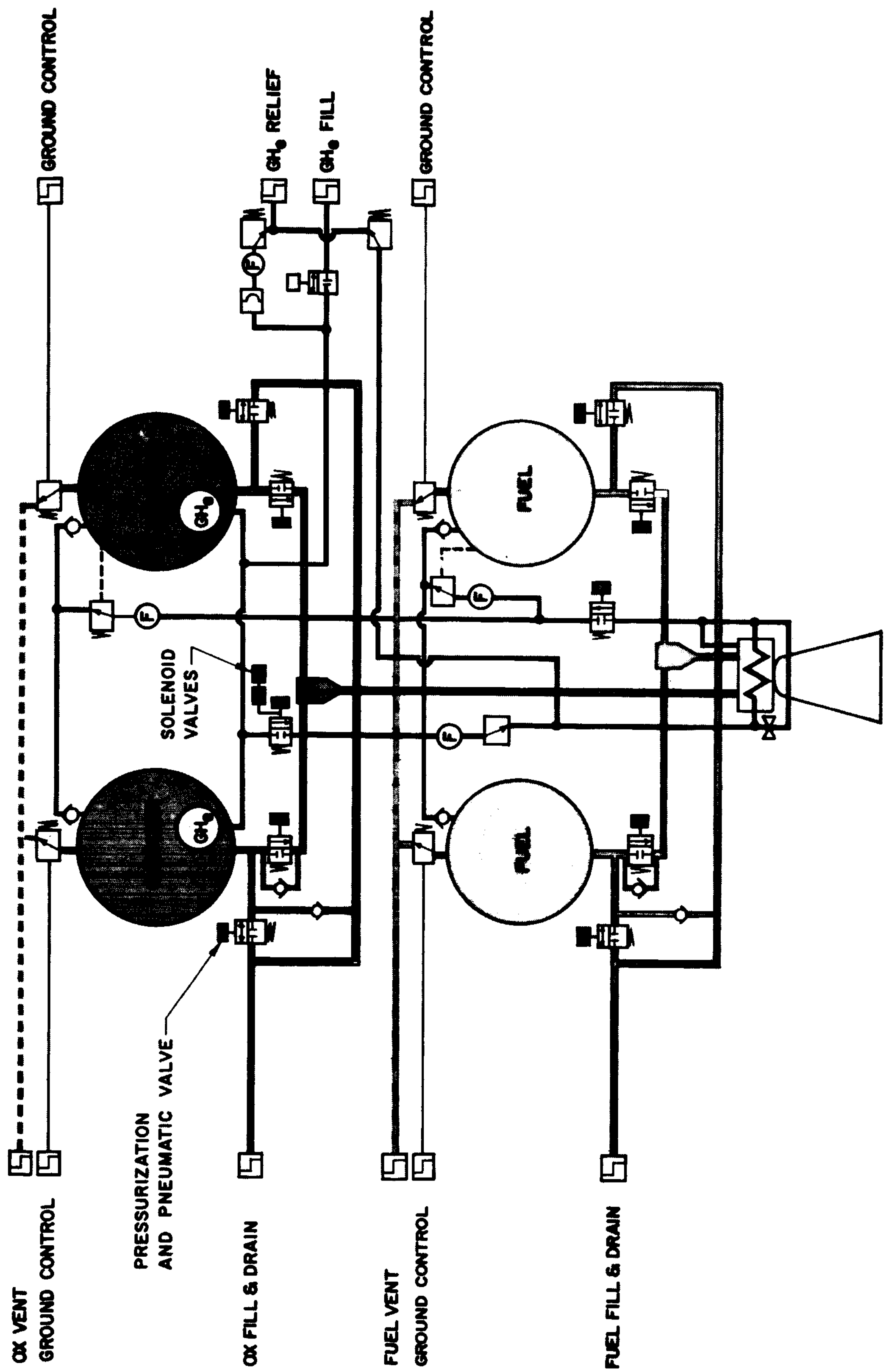


Fig. 6 Fluid System Schematic for a Typical Space-Storable Engine

After completion of fuel and oxidizer fill, the helium storage spheres located within either the oxidizer or fuel tanks are charged from a high-pressure ground source. After helium loading (enough time is allowed for the gas to cool to the propellant temperature), the helium fill line valve is closed.

In the event of a need to drain the system, the liquid oxidizer and fuel are transferred out the fill/drain lines by gravity head. In the event that propellant vapor (as well as liquid) must be removed, the tanks can be pressurized with helium, or a new purge cycle can be initiated after the liquid is completely drained. Helium pressure is reduced to 3,000 psi or less to accommodate expansion as the tank temperature increases. Prior to launch, all fill-line shutoff valves are closed, and trapped propellant in the lines is drained; the lines are then purged and the quick-disconnects separated.

Before engine start, the feed lines are cooled by opening a feed-line shutoff valve (fuel, for most propellant combinations) that allows preflow to the engine. Next, the oxidizer feed-line valve is opened and the engine started. Pressurization of the tanks is accomplished with helium that is heated prior to injection into the tanks. The helium shutoff valve is opened so that the gas flows through a regulator that reduces the pressure from the storage level of 4,500 psia maximum to about 500 psia. At this pressure level, the gas flows through an engine heat exchanger where it is heated to the desired temperature (controlled with a calibrated bypass). The gas flows to a low-range pressure regulator set to the desired tank total pressure level, and then through a check valve into the tank. The check valve prevents propellant from flowing into the pressurization line during zero-g coast. Gaseous helium remains in the line upstream of the check valve. Subsequent to engine burn, the helium supply is shut off, the engine is cooled by fuel postflow, and the feedline valves are closed.

3.4 COMPARISON OF FINAL DESIGNS

The Mars Orbiter and MEM Ascent Stage designs were optimized in Task II by minimizing the combined weights of propellant, structure, insulation, pressurization systems, and engine system. Weights of structure, insulation, tanks, and pressurization system were minimized through the use of point designs, carefully constructed models, and detailed thermal/structural/pressurization optimizations analyses performed on a large digital computer. Design and analysis were performed only on the propulsion module of the baseline Mars Orbiter vehicle (Fig. 3) in order to modify it for use of alternate propellants. Modified Orbiter stage designs using earth storables, space storables, and cryogenics are discussed in the following paragraphs.

Earth-Storable Mars Orbiter. The earth-storable configuration for pump- and pressure-fed systems for N_2O_4/A-50 and ClF_5/MHF-5 propellants is shown in Fig. 7. This is basically the TRW configuration modified to the study groundrules and mission criteria.

Space-Storable Mars Orbiter. The space-storable configuration for FLOX/CH_4, OF_2/CH_4, and F_2/NH_3 pump-fed systems is shown in Fig. 8. The corresponding pressure-fed configurations are shown in Fig. 9, adding OF_2/B_2H_6 to the propellant combinations.

Cryogenic Mars Orbiter. The cryogenic configuration for F_2/H_2 and O_2/H_2 pump- and pressure-fed systems is shown in Fig. 10. The pump-fed cryogenic configuration and all pressure-fed systems were assumed to use an extendable nozzle to avoid exceeding the basic dynamic envelope by an unacceptable length. Alternate approaches could include engines with decreased thrust and lower expansion ratios.

MEM Ascent Stage. The space-storable ascent stage configuration is shown in Fig. 11, with data for earth-storable and deep cryogenic propellants tabulated. All configurations are pump fed and feature an Aerospike-type nozzle design.

3.5 PERFORMANCE COMPARISON

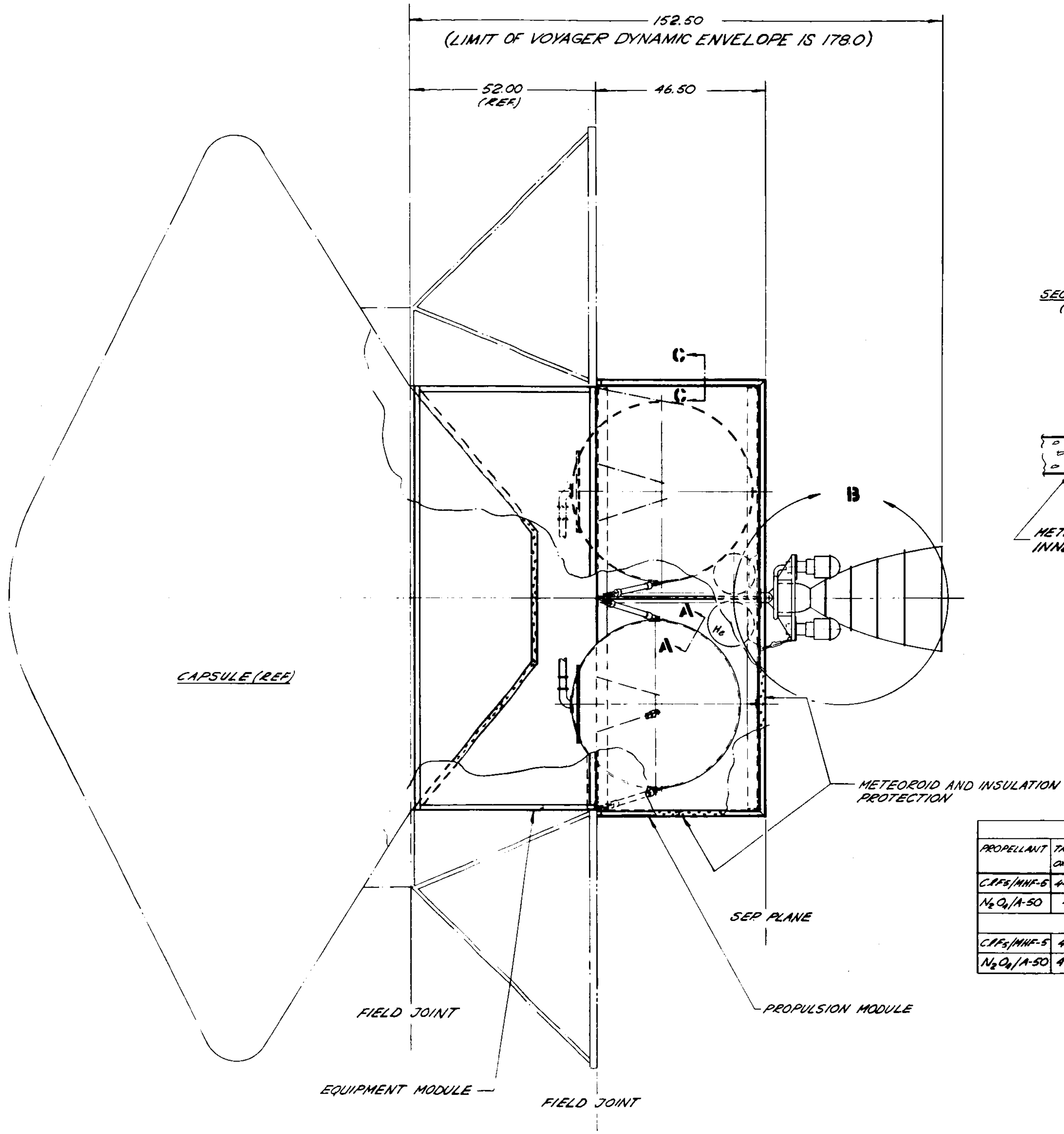
A performance comparison of the Mars Orbiter and MEM Ascent Stages is presented in the following paragraphs.

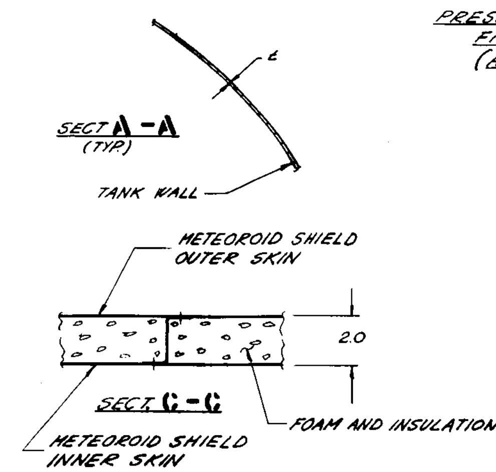
Mars Orbiter. The basic Mars Orbiter propulsion stage, using the LM descent engine burning N_2O_4/A-50, was estimated by TRW Systems to weigh 13,453 lb to orbit a payload of 8,143 lb. Using the same propellant and basic stage design, the system was revised to use a new, pump-fed N_2O_4/A-50 engine with delivered specific impulse increased from 305 to 335 sec. The resulting stage weight was reduced from 13,453 to 9,535 lb to form a new "baseline" orbiter. Figure 12 shows the stage weights for all propellant combinations for both pump-fed and pressure-fed systems. The pump-fed systems outperform the pressure-fed systems by from 11 percent for F_2/NH_3 to 21 percent for F_2/H_2. Figure 13 shows the stage weights for an alternate spacecraft orientation and the results of venting H_2. Sun-shading the cryogens and space storables improved performance about 7 percent and 2 percent, respectively. Earth storables were best when not shaded. Venting H_2 improved performance about 5 percent. The space-storable and cryogenic stages outperform the N_2O_4/A-50 stages in all cases by inserting the fixed 8,143-lb payload with a lighter propulsion stage. The percent reduction in stage weight over that of the new "baseline" N_2O_4/A-50 stages is presented in Table 8 for each propellant combination, feed system, and spacecraft orientation. The lightest pump-fed stage is achieved with F_2/H_2, and the lightest pressure-fed stage with OF_2/B_2H_6.

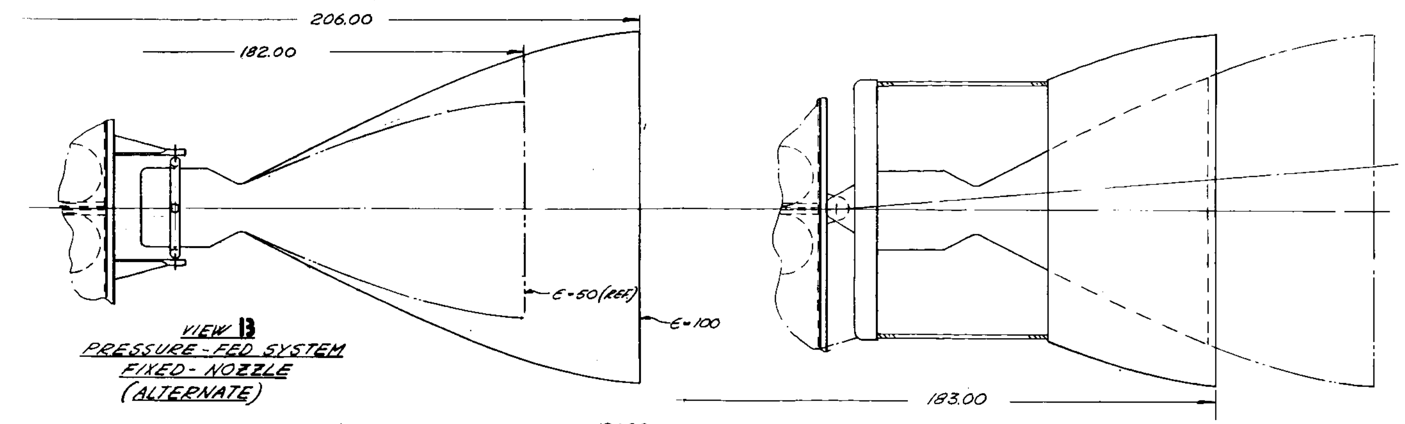
Table 8

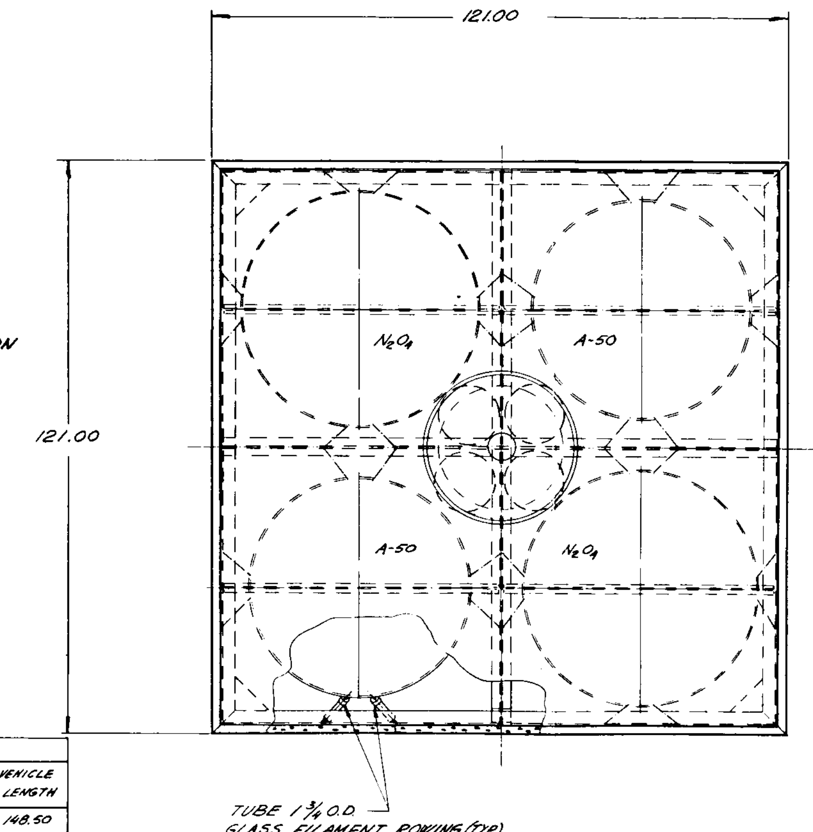
PERCENT REDUCTION IN MARS ORBITER STAGE WEIGHT OVER N_2O_4/A-50

Orientation	Propellant						
	F_2/H_2	O_2/H_2	$FLOX/CH_4$	OF_2/CH_4	F_2NH_3	OF_2/B_2H_6	$C\ell F_5$/MHF-5
Pump-Fed (Sun on Tanks)	24	11	16	17	16	–	3
Pump-Fed (Tanks in shade)	29	18	19	19	18	–	3
Pressure-Fed (Sun on Tanks)	12	4	11	14	14	20	2









PROPELLANT	TANK SIZE OXIDIZER	TANK SIZE FUEL	MIXTURE RATIO	I_{SP}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH
PUMP-FED SYSTEM							
ClF_5/MHF-5	44.40 I.D.	[illegible] I.D.	2.40/1	342	750	100	148.50
N_2O_4/A-50	48.20	44.70	2.0	335	750	100	152.50
PRESSURE-FED SYSTEM							
ClF_5/MHF-5	45.50	41.80	2.4	330	100	100	179
N_2O_4/A-50	49.50	46.70	2.0	328	100	100	183

TANK DATA:
MATERIAL AL 2021

METEOROID PROTECTION & INSULATION
DUAL WALL AL SKIN AND FOAM BETWEEN

EARTH-STORABLE PROPELLANTS

BASELINE IS N_2O_4/A-50, PUMP FED

Fig. 7 Mars Orbiter Earth-Storable Stage Details

FOLDOUT FRAME 1

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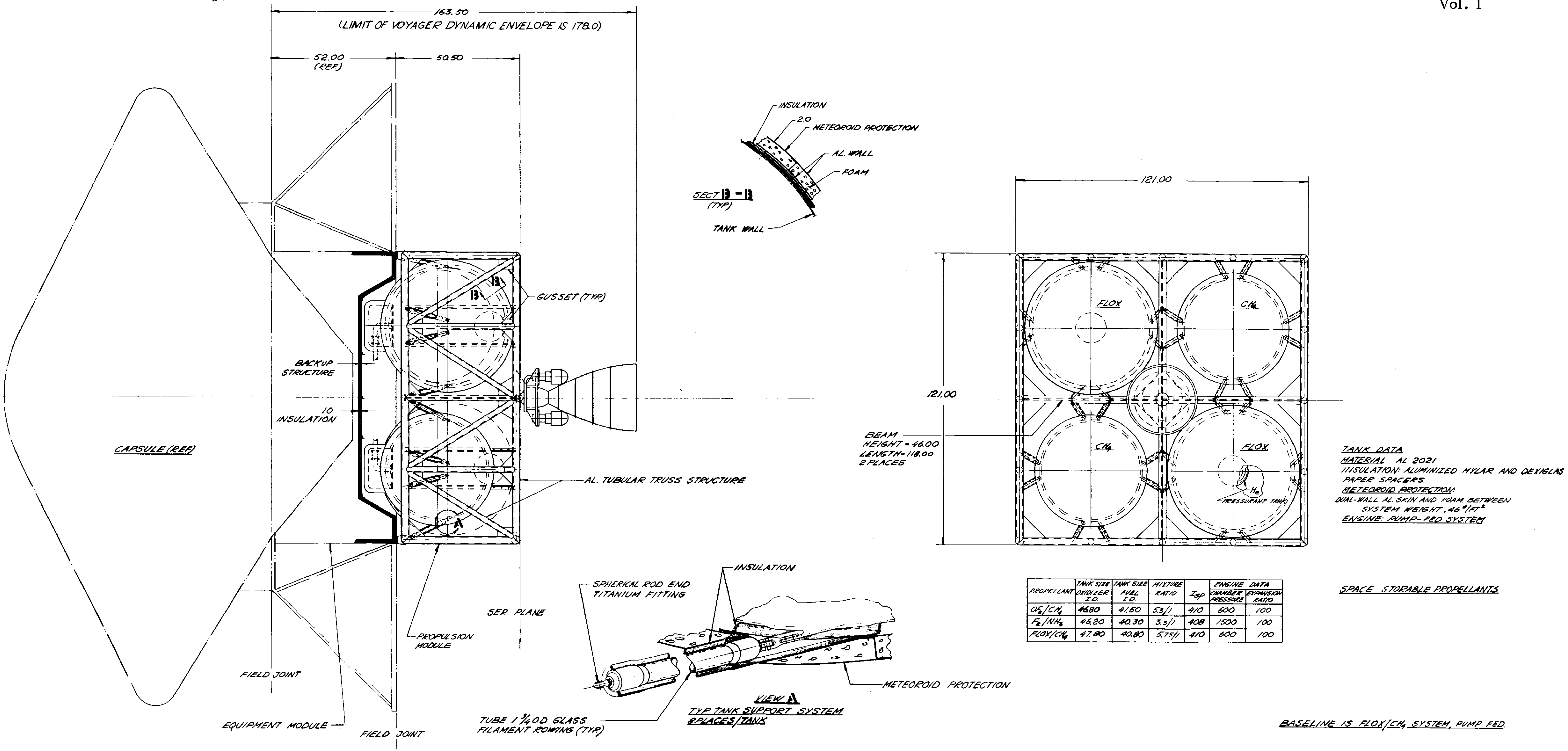
FOLDOUT FRAME 2



PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
OF_2/CH_4	46.80	41.60	5.5/1	410	600	100
F_2/NH_3	46.20	40.30	3.3/1	408	1500	100
$FLOX/CH_4$	47.80	40.80	5.75/1	410	600	100

Fig. 8 Mars Orbiter Space-Storable Pump-Fed Stage Details

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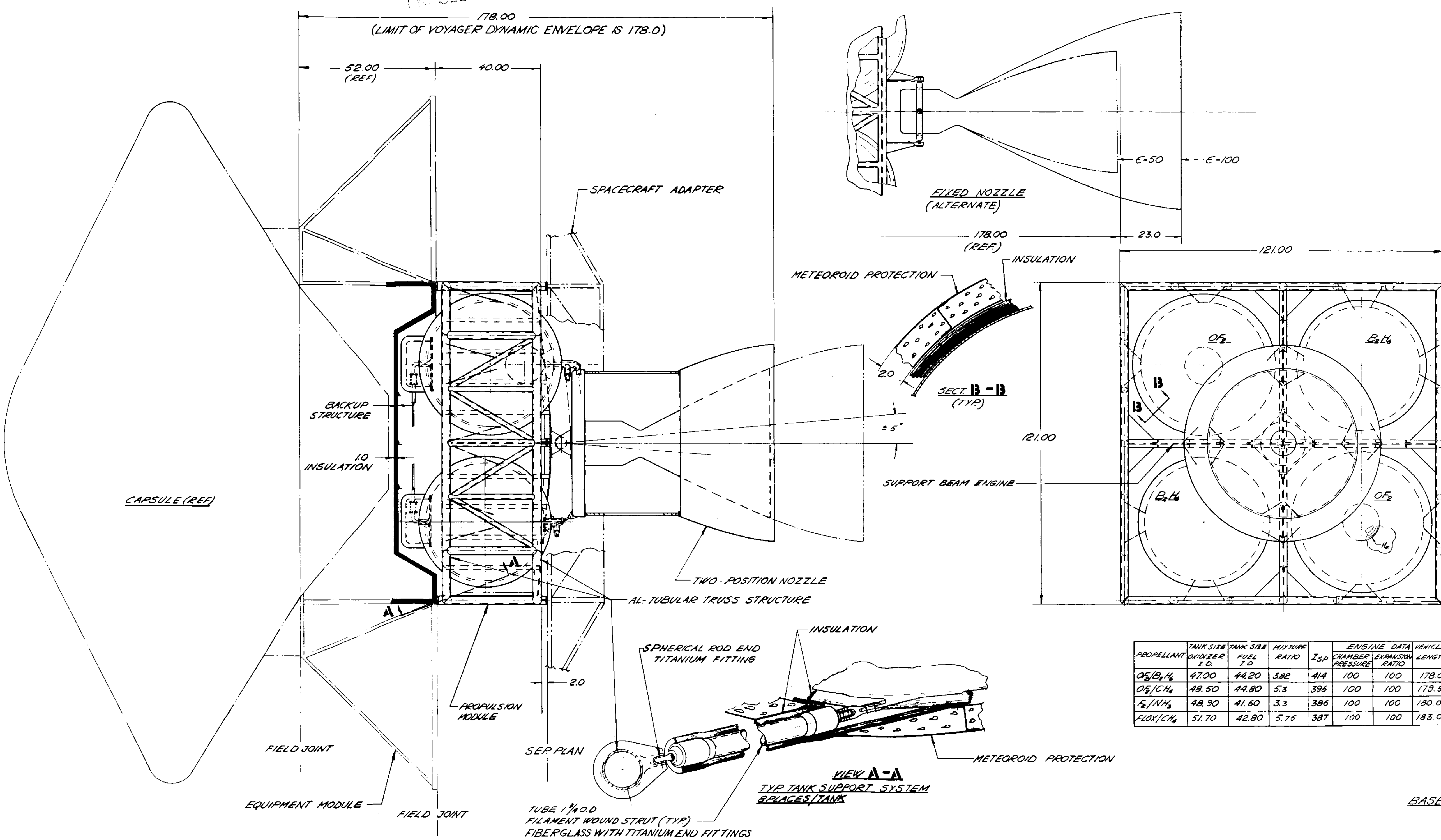

TANK DATA
MATERIAL AL 2021
INSULATION: ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
METEOROID PROTECTION:
TWO AL SKIN AND FOAM BETWEEN
SYSTEM WEIGHT .48 #/FT.²

PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{SP}	ENGINE DATA CHAMBER PRESSURE	EXPANSION RATIO	VEHICLE LENGTH
OF_2/B_2H_6	47.00	44.20	3.82	414	100	100	178.0
OF_2/CH_4	48.50	44.80	5.3	396	100	100	179.5
F_2/NH_3	48.90	41.60	3.3	386	100	100	180.0
$FLOX/CH_4$	51.70	42.80	5.75	387	100	100	183.0

SPACE STORABLE PROPELLANTS

BASELINE IS OF_2/B_2H_6 SYSTEM, PRESSURE FED

Fig. 9 Mars Orbiter Space Storable Pressure-Fed Stage Details

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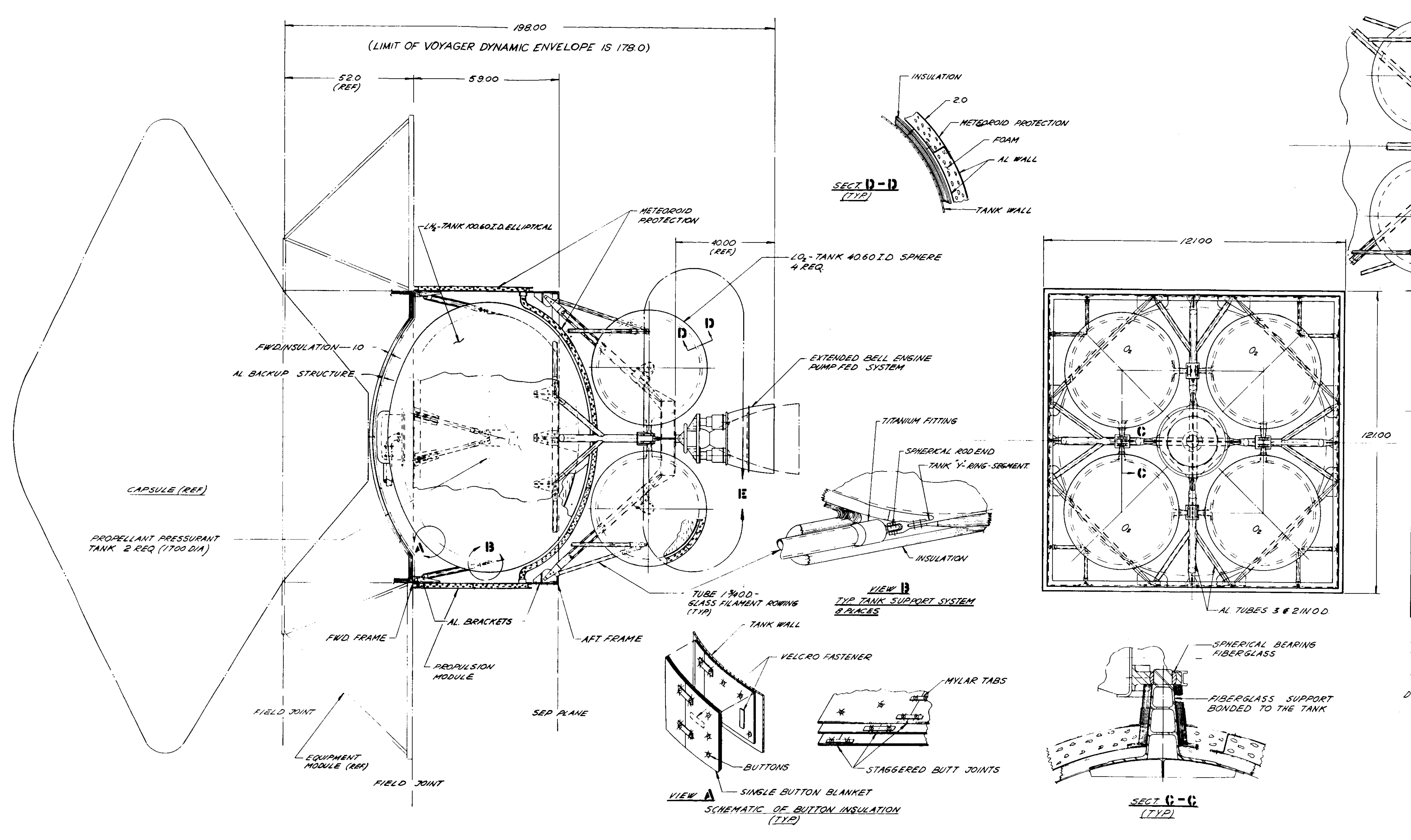

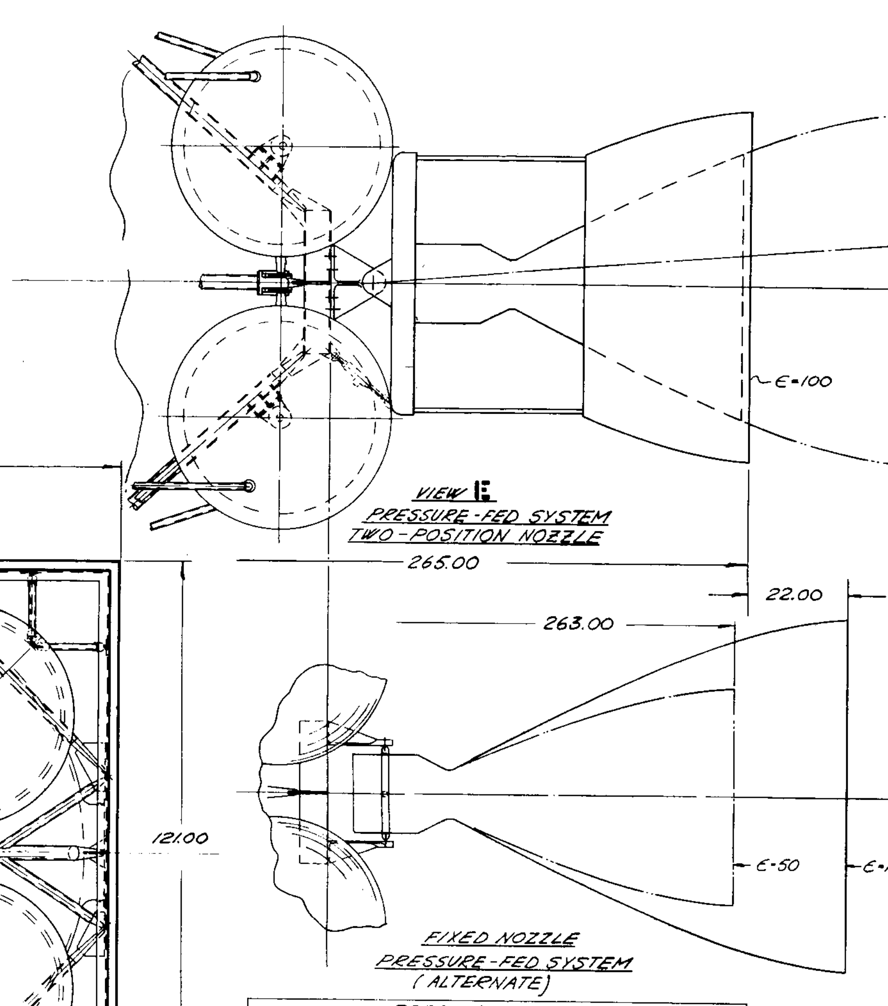

PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH IN
PRESSURE-FED SYSTEM							
O_2/H_2	41.80	100/92	6	445	100	100	265
F_2/H_2	39.60	90/64	13	442	100	100	245
PUMP-FED SYSTEM							
O_2/H_2	40.60	104.3/74.6	6	451	900	100	198
F_2/H_2	46.70	82.6/59	13	468	900	100	187

TANK DATA
MATERIAL AL 2021
INSULATION: ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
METEOROID PROTECTION:
DUAL WALL AL SKIN AND FOAM BETWEEN
SYSTEM WEIGHT .49 #/FT²

DEEP CRYOGENIC PROPELLANTS

BASELINE IS O_2/H_2, PUMP FED

Fig. 10 Mars Orbiter Cryogenic Stage Details

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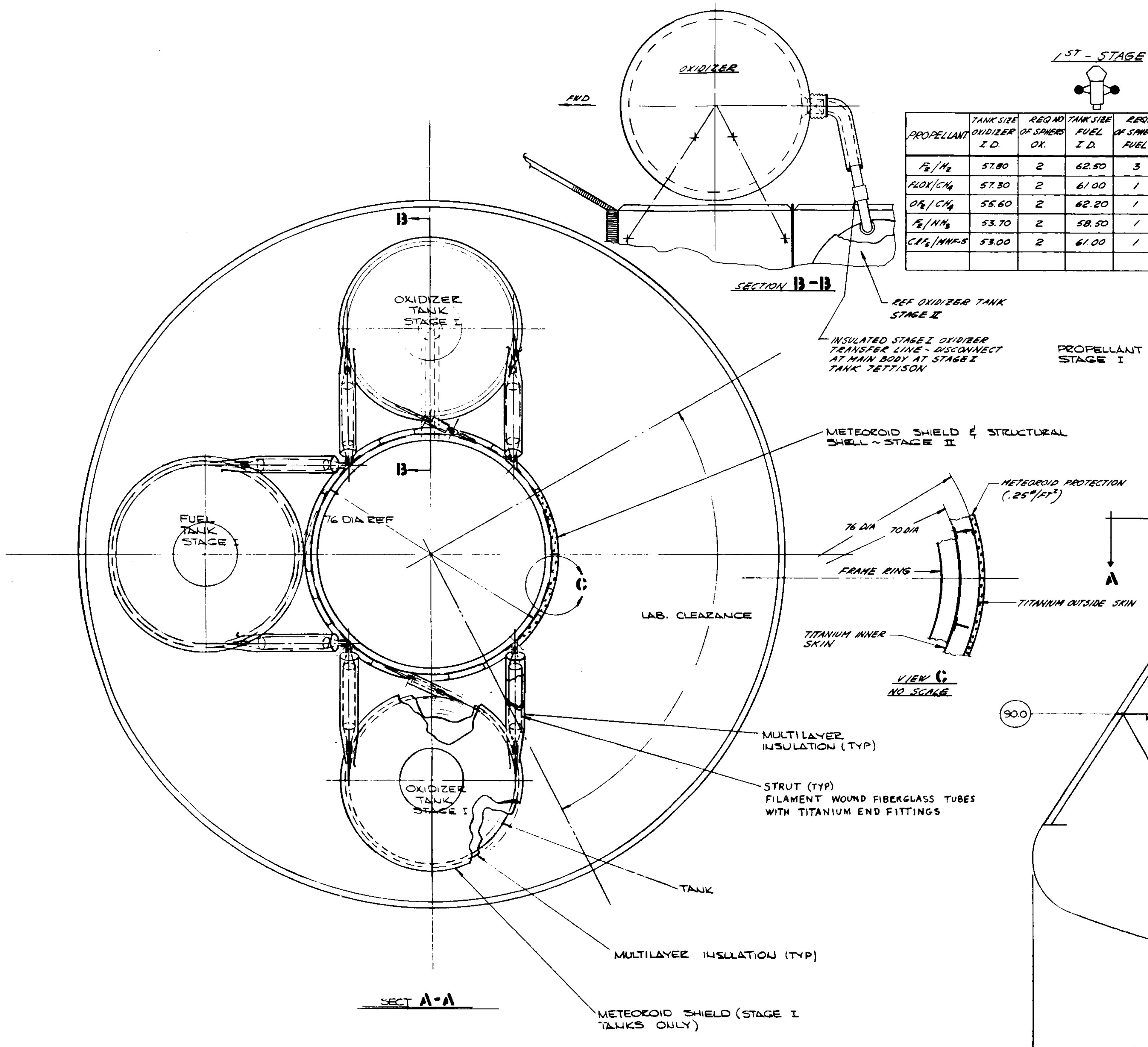
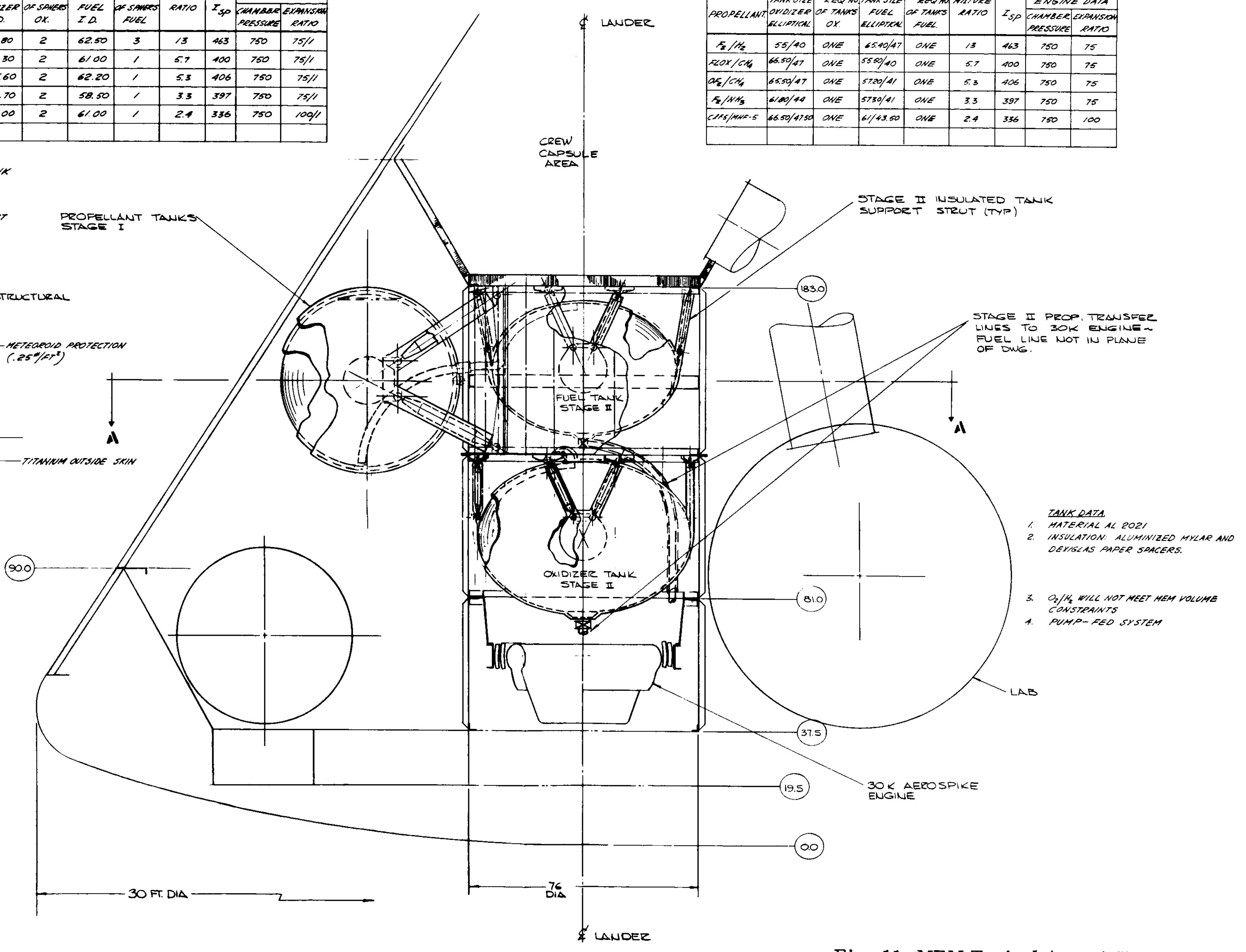


1ST - STAGE

PROPELLANT	TANK SIZE OXIDIZER I.D.	REQ NO OF SPHERS OX.	TANK SIZE FUEL I.D.	REQ. NO. OF SPHERS FUEL	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
F_2/H_2	57.80	2	62.50	3	13	463	750	75/1
$FLOX/CH_4$	57.30	2	61.00	1	5.7	400	750	75/1
OF_2/CH_4	55.60	2	62.20	1	5.3	406	750	75/1
F_2/NH_3	53.70	2	58.50	1	3.3	397	750	75/1
$ClF_5/MHF\text{-}5$	53.00	2	61.00	1	2.4	336	750	100/1

2ND - STAGE

PROPELLANT	TANK SIZE OXIDIZER ELLIPTICAL	REQ NO OF TANKS OX.	TANK SIZE FUEL ELLIPTICAL	REQ NO OF TANKS FUEL	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
F_2/H_2	55/40	ONE	65.40/47	ONE	13	463	750	75
$FLOX/CH_4$	66.50/47	ONE	55.50/40	ONE	5.7	400	750	75
OF_2/CH_4	65.50/47	ONE	57.20/41	ONE	5.3	406	750	75
F_2/NH_3	61.80/44	ONE	57.30/41	ONE	3.3	397	750	75
$ClF5/MHF\text{-}5$	66.50/47.50	ONE	61/43.50	ONE	2.4	336	750	100



TANK DATA
1. MATERIAL AL 2021
2. INSULATION ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
3. O_2/H_2 WILL NOT MEET MEM VOLUME CONSTRAINTS
4. PUMP-FED SYSTEM

Fig. 11 MEM Typical Ascent Stage Design

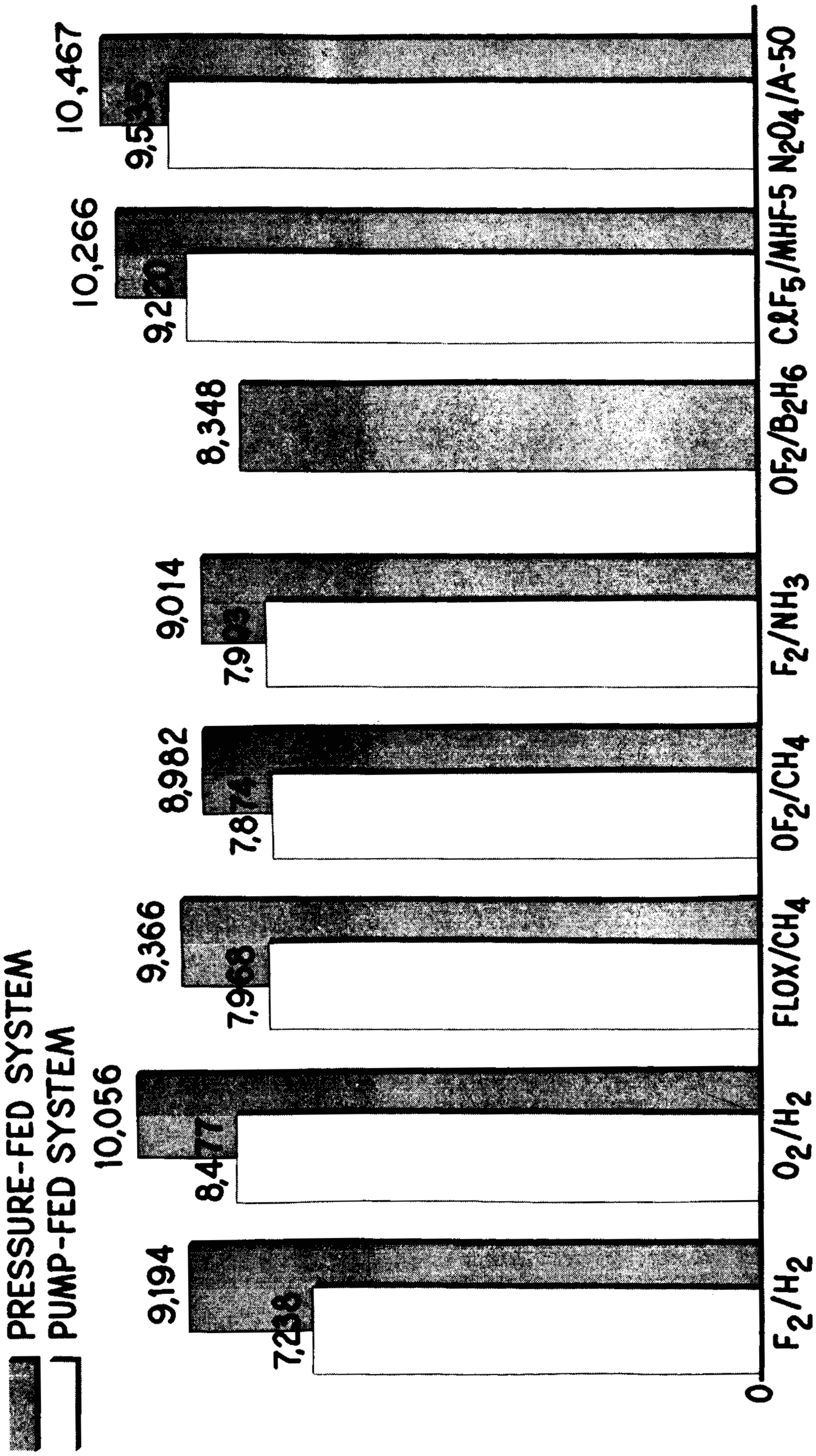


Fig. 12 Mars Orbiter Propulsion Stage Weights – Pressure-Fed and Pump-Fed Systems Comparison

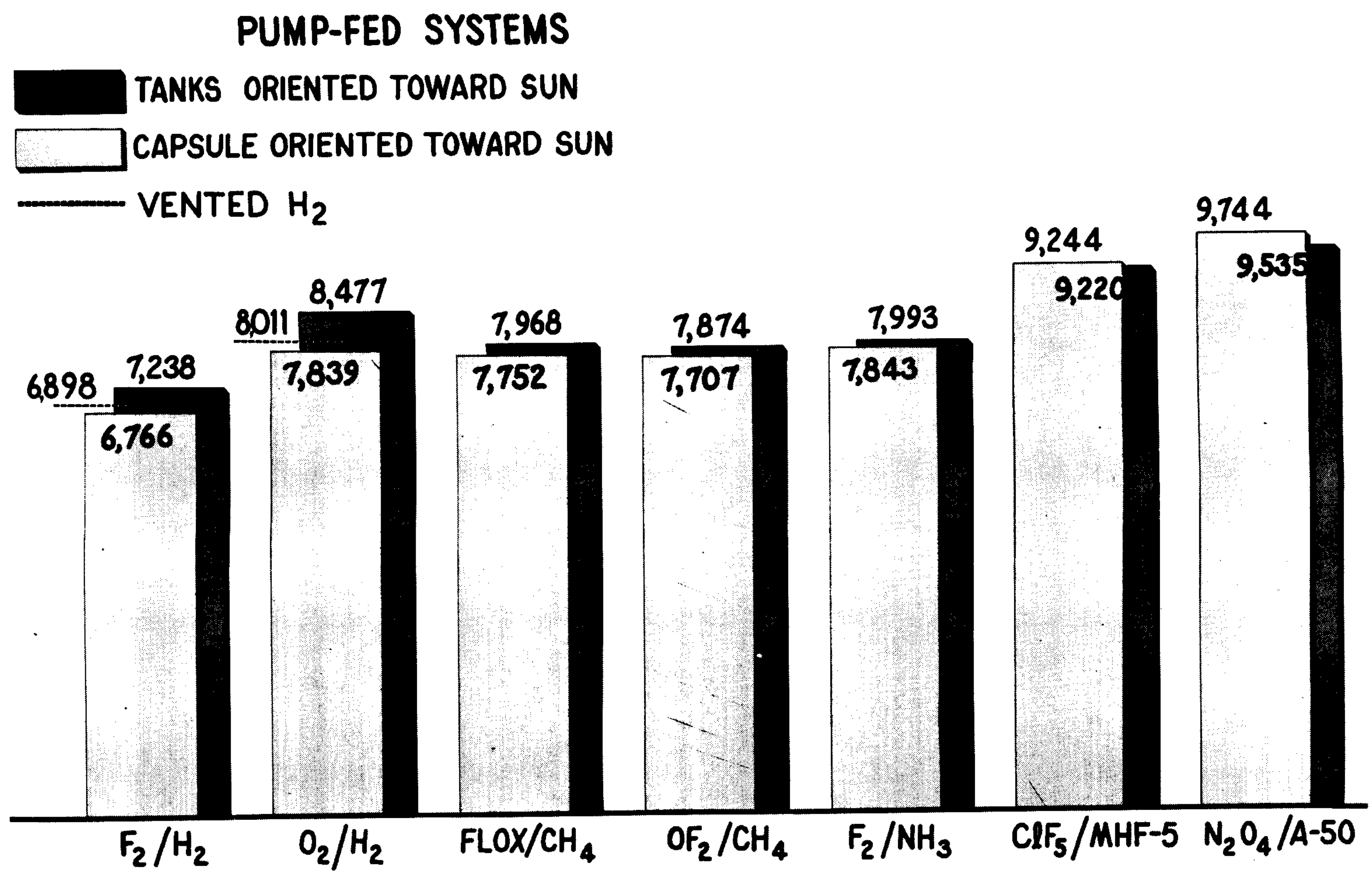


Fig. 13 Mars Orbiter Propulsion Stage Weights – Effects of Orientation and Venting

MEM Ascent Stage. The basic MEM Ascent Stage, burning FLOX/CH_4, was estimated by North American-Rockwell to weigh 24,600 lb in order to launch and orbit a payload of 5,250 lb, providing a total ΔV of 16,000 ft/sec. Using the same propellant with a modified stage design, the MEM was revised to a nominal diameter of 30 ft, spherical propellant tanks, and an increase in specific impulse from 383 to 400 sec. The resulting ascent stage weight was reduced from 24,600 lb to 21,301 lb to form a new "baseline." Figure 14 shows the stage weights for all propellant combinations (except O_2/H_2 which exceeded the volume limits).

The percent change in the weight over that of the new "baseline" FLOX/CH_4 stage is presented in Table 9 for each propellant combination. The lightest weight stage is achieved with F_2/H_2.

Table 9

PERCENT CHANGE IN MEM ASCENT STAGE WEIGHT FROM FLOX/CH_4

Propellant	F_2/H_2	OF_2/CH_4	F_2/NH_3	$C\ell F_5$/MHF-5
Percent Change	-11	-3	-1	+32

Additional performance comparisons are described in the following Sections, including the results of detailed analysis of performance sensitivity to mission and stage variables.

3.6 SENSITIVITY ANALYSES

An assessment was made to determine the system effects of varying several of the design parameters for the Mars Orbiter pump-fed vehicle. The parameters varied with mission length, surface coatings, meteoroid flux, specific impulse, insulation conductivity, propellant initial condition, vent versus nonvent, vehicle orientation to the sun, and a "worst-on-worst" combination.

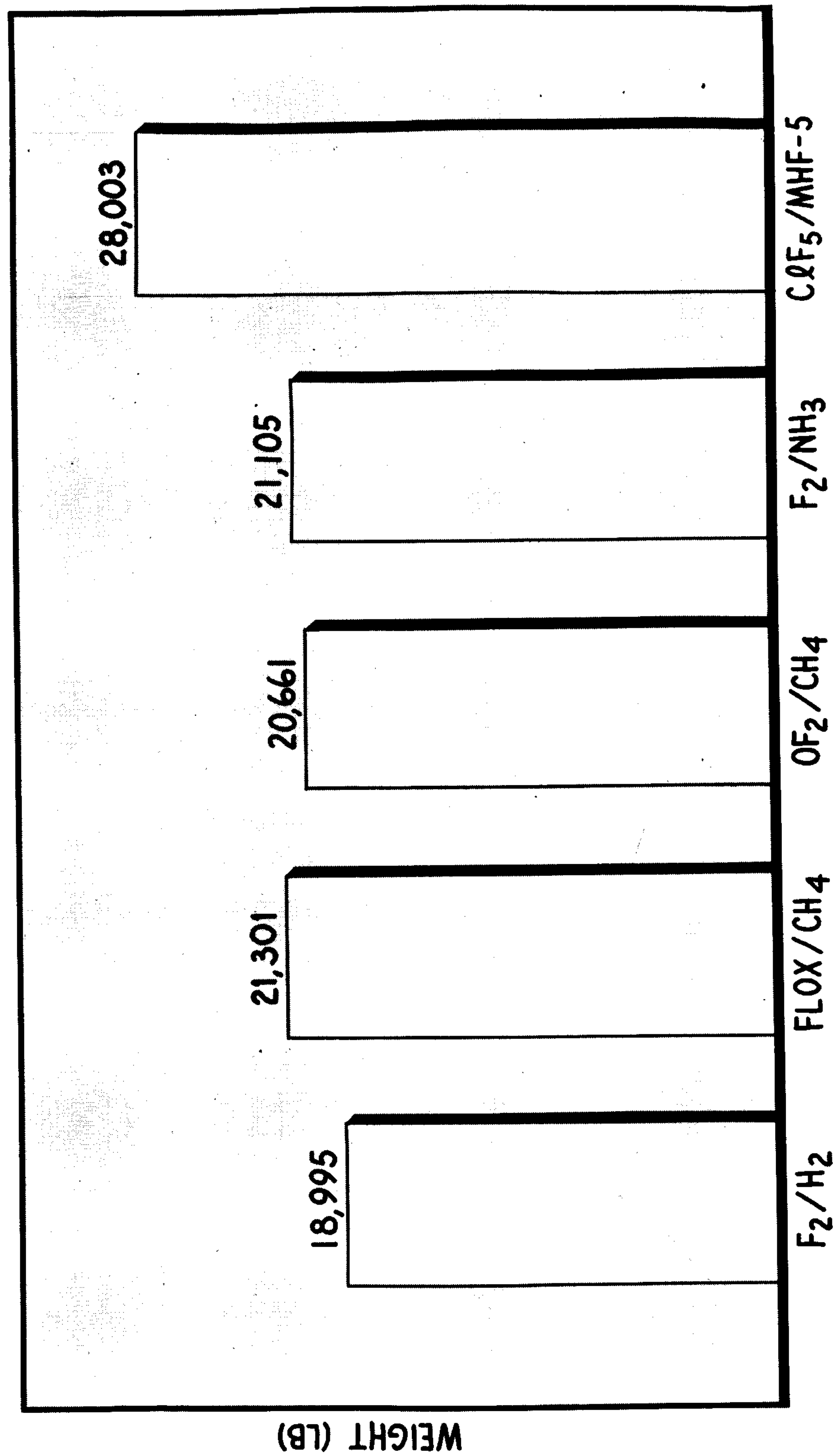


Fig. 14 MEM Ascent Stage Weights

3.6.1 Mission Length

The first investigation was made by extending the mission duration to a total of 300 days from 205 days. All mission sequences and velocity steps were kept constant, except for the interplanetary transit, which was extended by 95 days. The tank design pressure, insulation thickness, and total stage weights for the baseline sun-on-tank, nonvented systems are presented in Table 10. The indicated weight changes vary from about 0.5 percent for the space storables to 2.9 percent for O_2/H_2. Three additional mission lengths were investigated assuming that the final orbit trim burn was performed by a secondary system. The new mission lengths were 195 days, 290 days, and 650 days. The 195- and 290-day missions are the previously discussed Mars Orbiter mission, and the 650-day mission utilizes the same spacecraft on a trip to Jupiter. The tank operating pressure, insulation thickness, and total stage weights for the various propellant combinations are listed in Table 11. The longer transit Mars mission represents a 2 to 3 percent weight penalty when cryogens are used, and essentially no penalty for space storables and earth storables. For the 650-day Jupiter mission, there is a 3 to 5 percent penalty for the cryogens, and essentially no penalty for the remaining propellants. For the earth storables there is an interim tank pressure rise that is well below the tank minimum-gage limit. The pressure also drops toward the end of the mission so that the pressurant residuals do not affect the system weights.

3.6.2 Surface Coatings

To assess the sensitivity to surface characteristics, the various propellant combinations were evaluated with both silver and aluminum-backed Optical Solar Reflector (OSR), white thermatrol paint, and white skyspar paint. Table 12 presents the stage weights resulting from using these coatings for the sun-on-tank vehicle configuration, and shows the insensitivity to surface finish. For the cryogens and space storables the silver-backed OSR yielded the lightest weight system, and white paint provided the lightest weight system for the earth storables. The maximum effect of alternate surface coating was about a 1 percent increase in stage weight for the cryogens with white thermatrol paint as compared with silver-backed OSR.

Table 10

MARS ORBITER SENSITIVITY TO MISSION DURATION
(Sun-On-Tank, Nonvented, 205-Day vs. 300-Day Mission)

Propellant	Surface Finish (α/ϵ ratio)	205-Day Mission			300-Day Mission			Weight Change (%)
		Max. Oper. Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Max. Oper. Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	
F_2	0.05/0.80	40	1-1/8	7,327	50	1-1/4	7,442	1.6
H_2	0.05/0.80	130	4-5/8		150	5-3/4		
O_2	0.05/0.80	58	3/4	8,477	80	1	8,721	2.9
H_2	0.05/0.80	96	4-5/8		114	5-1/4		
FLOX	0.05/0.80	57	1-1/2	7,968	75	1-5/8	8,023	0.7
CH_4	0.05/0.80	107	3/4		140	1		
OF_2	0.05/0.80	45	1-1/8	7,874	56	1-1/4	7,921	0.6
CH_4	0.05/0.80	107	3/4		140	1		
F_2	0.05/0.80	59	1-1/8	7,984	74	1-5/8	8,009	0.3
NH_3	0.60/0.91	<15	1/4		<20	1/4		

Note: N_2O_4/A-50 and ClF_5/MHF-5 are relatively insensitive to mission duration.

Table 11

MARS ORBITER SENSITIVITY TO MISSION DURATION
(Sun-On-Tank, Pump-Fed, Nonvented, Optimum α/ϵ, No Orbit Trim)

PROPELLANT	195-DAY MISSION			290-DAY MISSION			650-DAY JUPITER MISSION		
	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE (LB)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)
F_2	80	*	7,080	100	5/8	7,224	110	1/2	7,299
H_2	120	4-1/4		140	5-1/4		190	5-3/4	
O_2	100	*	8,206	160	1/2	8,382	168	1/2	8,615
H_2	86	3-3/4		106	4-1/4		174	4-1/2	
FLOX	150	5/8	7,796	175	5/8	7,811	175	3/4	7,809
CH_4	164	*		180	5/8		210	1/2	
OF_2	120	*	7,723	123	1/2	7,750	133	*	7,730
CH_4	164	*		180	5/8		210	1/2	
F_2	74	1/2	7,825	183	5/8	7,858	183	3/4	7,865
NH_3	<15	**		<20	**		<30	**	
N_2O_4	<15	**	9,527	<20	**	9,527	<30	**	9,527
A-50	<15	**		<20	**		<30	**	
ClF_5	<15	**	9,218	<20	**	9,218	<30	**	9,218
MHF-5	<15	**		<20	**		<30	**	

*MINIMUM VALUE 1/2-IN. SUPERINSULATION.
**MINIMUM VALUE 1/4-IN. FOAM.

Table 12

MARS ORBITER SENSITIVITY TO TANK SURFACE FINISH
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission)

Propellant	OSR Silver Backed α/ϵ = 0.05/0.80			OSR Aluminum Backed α/ϵ = 0.10/0.80			White Thermatrol Paint (Degraded) α/ϵ = 0.30/0.95			White Skyspar Paint (Degraded) α/ϵ = 0.60/0.91		
	P	T	WT	P	T	WT	P	T	WT	P	T	WT
F_2	40	1-1/8	7,238				75	2	7,327			
H_2	130	4-5/8					150	4-3/4				
O_2	58	3/4	8,477				85	1	8,576			
H_2	96	4-5/8					115	4-3/4				
FLOX	57	1-1/2	7,968	65	1-5/8	7,997	70	2	8,063			
CH_4	107	3/4		110	7/8		160	1-1/8				
OF_2	45	1-1/8	7,874	52	1-1/4	7,887	58	1-1/2	7,943			
CH_4	107	3/4		110	7/8		160	1-1/8				
F_2	59	1-3/8	7,993	68	1-1/2	8,006	70	2	8,030			
NH_3	<15	MIN		<15	MIN		16	MIN				
N_2O_4	<15	1	9,639	<15	3/4	9,615	<15	1/2	9,567	<15	MIN	9,535
A-50	<15	1-3/8		<15	1-1/4		<15	3/4		<15	MIN	
ClF_5	<15	MIN	9,227	<15	MIN	9,220	<15	MIN	9,220	<15	MIN	9,220
MHF-5	<15	3/8		<15	1/4		<15	MIN		<15	MIN	

P = Tank maximum operating pressure (psia)
T = Insulation thickness (in.)
WT = Propulsion Module Weight (lb)

3.6.3 Meteoroid Flux

The meteoroid flux was increased by a factor of ten to evaluate the effect of a very severe change in this environment. This effect more than doubled the actual weight of the meteoroid shield. The effects on system weight are shown in Table 13. The cryogens are most sensitive, with stage weight penalties of about 4 percent as compared to 2 percent for space storables and 3 percent for earth storables.

Table 13
MARS ORBITER SENSITIVITY TO METEOROID FLUX
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission)

Propellant	Propulsion Module Weight (lb)		Weight Change (%)
	Basic Flux	10X Basic Flux	
F_2/H_2	7,238	7,503	3.7
O_2/H_2	8,477	8,885	4.8
$FLOX/CH_4$	7,968	8,141	2.2
OF_2/CH_4	7,874	8,047	2.2
F_2/NH_3	7,993	8,193	2.5
N_2O_4/A-50	9,535	9,811	2.9
ClF_5/MHF-5	9,220	9,476	2.8

3.6.4 Specific Impulse

The effect of varying the specific impulse by ±3 percent was also evaluated. The actual values of specific impulse used and the propulsion module weights are shown in Table 14, together with the percentage change in weights from nominal. F_2/H_2 is the least sensitive at about 30 percent weight change, followed by O_2/H_2 and the space storables at 3.3 percent, and the earth storables at 3.5 percent.

Table 14

MARS ORBITER SENSITIVITY TO SPECIFIC IMPULSE
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

Propellant	- 3%			Nominal		+ 3%		
	I_{sp} (sec)	Propulsion Module Weight (lb)	% Wt Change From Nominal	I_{sp} (sec)	Propulsion Module Weight (lb)	I_{sp} (sec)	Propulsion Module Weight (lb)	% Wt Change From Nominal
F_2/H_2	453.96	7,455	3.0	468	7,238	482.04	7,035	-2.8
O_2/H_2	437.47	8,757	3.3	451	8,477	464.53	8,214	-3.2
$FLOX/CH_4$	397.70	8,231	3.3	410	7,968	422.30	7,721	-3.2
OF_2/CH_4	397.70	8,134	3.3	410	7,874	422.30	7,638	-3.1
F_2/NH_3	395.76	8,257	3.3	408	7,993	420.24	7,753	-3.1
N_2O_4/A-50	324.95	9,865	3.5	335	9,535	345.05	9,226	-3.3
ClF_5/MHF-5	331.74	9,544	3.5	342	9,220	352.26	8.925	-3.3

3.6.5 Insulation Conductivity

Insulation conductivity sensitivity was analyzed wherein the baseline insulation conductivities of 2.5×10^{-5} Btu/hr-ft-°R used for hydrogen tanks, 5×10^{-5} for oxygen, fluorine, and the space storables, and 10×10^{-5} for NH_3 and the earth storables were degraded by a factor of two. Table 15 lists the operating pressure, insulation thickness, and propulsion module weight for the baseline case and for the systems with assumed degraded insulation. Weight penalties for degraded insulation were 0.6 percent for OF_2/CH_4 and F_2/NH_3, 2.9 percent for FLOX/CH_4, 3.5 percent for F_2/H_2, and 5.1 percent for O_2/H_2.

3.6.6 Propellant Initial Condition

An analysis was also made of the effect of the initial condition of hydrogen and the venting of hydrogen for the cryogenic systems. The comparison was made with the sun-on-tank orientation comparing (1) saturated, (2) triple-point, (3) 50-percent slush hydrogen, and (4) venting the hydrogen. With the vented hydrogen case, the oxidizer is cooled by passing the vented hydrogen through a heat exchanger in the oxidizer tank.

Table 16 lists the operating pressure, insulation thickness, and propulsion module weights for the cases studied. The percent weight reductions from the nominal for F_2/H_2 are: Triple point 1.5 percent, slush 2.4 percent, and vented 4.7 percent. The corresponding reductions for O_2/H_2 are: 3.0 percent, 4.0 percent, and 5.5 percent. Combinations of triple-point or slush with venting were not examined.

3.6.7 Vehicle Orientation

The effect of orienting the vehicle so that its propellant tanks are exposed to the sun or shielded from the sun provided the most significant effect in terms of insulation thickness, operating pressure, and system weight. Table 17 presents these data. It is significant that the hydrogen tank pressure can be reduced from 130 to 80 psi, the

Table 15

MARS ORBITER SENSITIVITY TO INSULATION CONDUCTIVITY
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

PROPELLANT	BASELINE				DEGRADED				WEIGHT INCREASE (%)
	INSUL CONDUCT., K (BTU/HR-FT-°R)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	INSUL CONDUCT., K (BTU/HR-FT-°R)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	
F_2 H_2	5×10^{-5} 2.5×10^{-5}	40 130	1-1/8 4-5/8	7,238	10×10^{-5} 5×10^{-5}	58 187	1-3/4 6-1/4	7,492	3.5
O_2 H_2	5×10^{-5} 2.5×10^{-5}	58 96	3/4 4-5/8	8,477	10×10^{-5} 5×10^{-5}	65 140	1-1/2 6-3/8	8,910	5.1
FLOX CH_4	5×10^{-5} 5×10^{-5}	57 107	1-1/2 3/4	7,968	10×10^{-5} 10×10^{-5}	83 165	2-3/8 1-1/4	8,198	2.9
OF_2 CH_4	5×10^{-5} 5×10^{-5}	45 105	1-1/8 3/4	7,874	10×10^{-5} 10×10^{-5}	69 165	1-1/2 1-1/4	7,921	0.6
F_2 NH_3	5×10^{-5} 10×10^{-5}	59 16	1-3/8 *	7,993	10×10^{-5} 20×10^{-5}	78 18	2-1/2 *	8,078	0.6

*MINIMUM VALUE 1/4-IN. FOAM.

Table 16

MARS ORBITER SENSITIVITY TO PROPELLANT INITIAL CONDITION AND VENTING
(Sun-On-Tank, Pump-Fed, 205-Day Mission, Optimum α/ϵ = 0.05/0.80)

PROPELLANT	NONVENTED									VENTED			
	SATURATED H_2			TRIPLE-POINT H_2			50% SLUSH H_2			SATURATED H_2			
	P	T	WT	P	T	WT	P	T	WT	P	T	WT	BOILOFF (LB)
F_2	40	1-1/8	7238	40	1-1/8	7126	40	1-1/8	7067	15	1/2	6898	100
H_2	130	4-5/8	(REF)	95	3-3/4	(-1.5%)	82	3-1/2	(-2.4%)	76	2	(-4.7%)	
O_2	58	3/4	8477	58	3/4	8226	58	3/4	8141	15	1/2	8011	140
H_2	96	4-5/8	(REF)	66	3-3/4	(-3%)	66	3	(-4.0%)	64	2	(-5.5%)	

P = TANK MAXIMUM OPERATING PRESSURE (PSIA)
T = INSULATION THICKNESS (IN.)
WT = PROPULSION MODULE WEIGHT (LB)

Table 17

MARS ORBITER SENSITIVITY TO ORIENTATION
(Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

Propellant	Sun on Tank			Sun on Capsule			Weight Change (%)
	P[a]	T[b]	WT[c]	P	T	WT	
F_2 H_2	40 130	1-1/8 4-5/8	7,238	<15 80	Min 1-3/4	6,766	-7.0
O_2 H_2	58 96	3/4 4-5/8	8,477	<15 72	Min 1-3/4	7,839	-8.1
FLOX CH_4	57 107	1-1/2 3/4	7,968	32 <15	Min Min	7,752	-2.8
OF_2 CH_4	45 107	1-1/8 3/4	7,874	<15 <15	Min Min	7,707	-2.2
F_2 NH_3	59 16	1-3/8 Min	7,993	<15 <15	Min Min	7,843	-1.9
N_2O_4 A-50	<15 <15	Min Min	9,535	<15 <15	2 2	9,744	+2.2
ClF_5 MHF-5	<15 <15	Min Min	9,220	<15 <15	3/8 3/8	9,244	+0.3

(a) Tank maximum operating pressure (psia)
(b) Insulation thickness (in.)
(c) Propulsion module weight (lb)

insulation thickness reduced from 4-5/8 to 1-3/4 in., and the system weight for this F_2/H_2 propulsion module reduced from 7,238 to 6,766 lb for a 7-percent weight saving by orienting the vehicle so that the propellant tanks are shaded. This effect varies from about 2 percent for the space storables to 8 percent for O_2/H_2. For the earth storables, sun-facing tanks are desired because a sun-shielded orientation requires greater insulation thickness to prevent the propellants from freezing.

3.6.8 Worst-On-Worst Analysis

As a summary to the sensitivity analysis, a combination of some adverse factors was considered. The insulation conductivity was doubled, the heat leaks were doubled, only white paint surfaces were assumed available, and the helium pressurization tanks were man-rated. This combination of adverse design conditions was entitled "worst-on-worst."

To evaluate the Mars Orbiter worst-on-worst requirements, the following specific conditions were analyzed:

- Vehicle Orientation
  - Sun on the capsule for all propellants except the earth storables
  - Sun on the tanks for N_2O_4/A-50 and ClF_5/MHF-5 using $\alpha/\epsilon = 0.6/0.91$ and $\alpha/\epsilon = 0.3/0.95$
  - Sun on tanks and capsule for F_2/NH_3 using an $\alpha/\epsilon = 0.3/0.95$
- High insulation conductivity (values of two times the baseline)
  - $k = 5.0 \times 10^{-5}$ Btu/hr-ft-°R for H_2
  - $k = 10.0 \times 10^{-5}$ Btu/hr-ft-°R for O_2, F_2, FLOX, CH_4, and OF_2
  - $k = 20.0 \times 10^{-5}$ Btu/hr-ft-°R for NH_3, N_2O_4, A-50, ClF_5, and MHF-5
- Double the heat leaks (values of two times the baseline)
  - Half the penetration (propellant feed and pressurant lines) thermal resistance
  - Half the support strut thermal resistance

The F_2/NH_3 propellant combination was analyzed for both the sun-on-capsule condition and sun on the tanks with an α/ϵ of 0.30/0.95 (white paint) to determine the optimum orientation. Also, all the sun-on-tank cases were analyzed with an α/ϵ of 0.3/0.95 and an α/ϵ of 0.6/0.91 to determine the optimum surface finish. For the F_2/NH_3 propellant combination, the sun-on-capsule condition resulted in the minimum system weight and is, therefore, the only one presented. This occurs even with 2-1/2 in. of insulation for NH_3 in order to prevent freezing because, with the sun on the tanks, the F_2 requires over 3 in. of insulation and the F_2 tank is slightly larger than the NH_3 tank. The N_2O_4/A-50 propellant combination results in minimum system weight with an α/ϵ of 0.6/0.91, whereas the ClF_5/MHF-5 combination optimizes with the α/ϵ of 0.3/0.95 because of the lower freezing points.

Many of the propellants experience a net heat loss during the mission, resulting in minimum insulation thicknesses. The propulsion module weights, operating pressures, and insulation thicknesses are presented in Table 18. It is significant that, having chosen the most desirable propulsion system orientation to the sun, the system is very insensitive to degradation of insulation, nonoptimum surface finishes, and man-rated helium tanks. The weight penalties varied between 0.6 and 1.7 percent with no clear distinction between classes of propellants.

3.7 PROPELLANT SELECTION FACTORS

Factors that can influence the choice of a propellant include system performance, volumetric constraints, system sensitivities to environment or to off-optimum operations, system complexity/reliability, development time/availability, cost, compatibility, and commonality. The actual selection of propellants for a specific application will depend on many additional factors peculiar to the requirements and circumstances present in the program. The final choice must rest with the vehicle program office.

An approach to use of some of the factors of importance in selecting a propellant is outlined in Fig. 15. In following the paths shown, a considerable amount of personal judgment and knowledge will be required.

Table 18

MARS ORBITER – WORST-ON-WORST ANALYSIS

Propellant	Baseline			Worst-On-Worst				Orientation
	Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Weight Increase (%)	
F_2 H_2	< 15 80	Min[a] 1-3/4	6,766	18 59	1/2 2	6,839	0.8	Sun on capsule
O_2 H_2	< 15 72	Min 1-3/4	7,839	18 52	1/2 2	7,930	1.4	Sun on capsule
FLOX CH_4	32 < 15	Min Min	7,752	52 22	1/2 1/2	7,845	1.2	Sun on capsule
OF_2 CH_4	< 15 < 15	Min Min	7,707	19 22	1/2 1/2	7,790	1.1	Sun on capsule
F_2 NH_3	< 15 < 15	Min Min	7,843	52 18	1/2 2-1/2	7,964	1.5	Sun on capsule
N_2O_4 A-50	< 15 < 15	Min Min	9,535	25 15	5/8 5/8	9,641	1.7	Sun on tank
ClF_5 MHF-5	< 15 < 15	Min Min	9,220	20 15	3/8 3/8	9,278	0.6	Sun on tank

(a) Min represents less than 1/2-in. of multilayer insulation required.

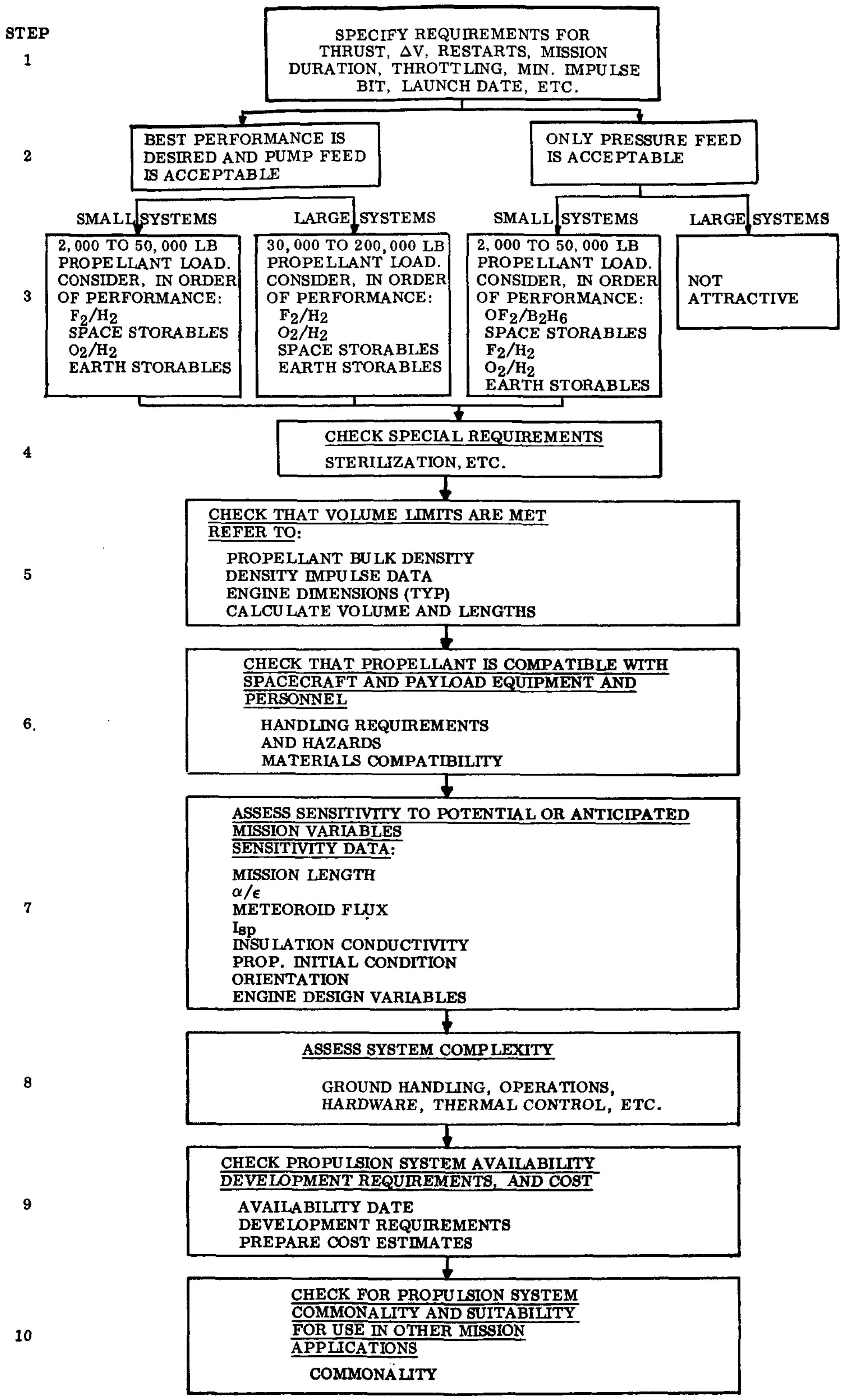


Fig. 15 Propellant Selection Factors to be Considered

In Step (1), the requirements placed on the propulsion stage for thrust, ΔV, restarts, mission duration, throttling, minimum impulse bit, launch date, etc., are stated in detail. A decision must then be made at Step (2) as to whether pump or pressure-feed will be specified, recognizing that pump feed will generally result in the lowest weight and most compact system.

In Step (3), the size of the required stage is estimated and the best performing propellant selected for further analysis. In this step, space storables include FLOX/CH_4, FLOX/C_3H_8, OF_2/CH_4, OF_2/C_3H_8, OF_2/MMH, and F_2/NH_3. Each of these propellants was found to be space storable for the missions analyzed with no overriding storage reason found for choosing one over another, although not all of these combinations were studied in detail. The specific impulse for pump-fed applications is comparable (± 1 percent), except for OF_2/C_3H_8, which provides a specific impulse about 2.5 percent below the mean. Pressure-fed data were obtained only for OF_2/CH_4, FLOX/CH_4, and F_2/NH_3, with the OF_2/CH_4 providing about 2.5 percent higher specific impulse than the latter two. OF_2/B_2H_6 is treated separately from the other space storables, since its use is restricted to pressure-fed applications and it provides a higher specific impulse than other pressure-fed space storables.

In Step (4), a check must be made to determine that mission-peculiar requirements, such as need for propellant or total stage sterilization, can be met.

In Step (5), a check is made to ensure that volume and dimensional limits for the spacecraft and launch system are met. Propellant and engine data are provided as an aid in this assessment.

In Step (6), a check is made to ensure that the selected propellant is compatible with spacecraft and payload equipment and personnel. Some general guides are provided.

In Step (7), a check is made of sensitivity to potential or anticipated mission variables, such as spacecraft orientation, venting requirements, meteoroid flux, propellant tank surface coating properties, and specific impulse tolerance. Sensitivity data for a typical mission, a Mars Orbiter, are presented as a guide.

In Step (8), the overall complexity of the selected system is evaluated to assess, on a preliminary basis, the probability of meeting reliability requirements.

In Step (9), the development status and requirements, availability date, and estimated development and procurement costs are assessed.

In Step (10), a check is made for propulsion system commonality and suitability for use in alternate mission applications.

If at any step the propellant under consideration is found unacceptable, or is judged not a good choice through marginal acceptance at one or more steps, the procedure is started again at Step (3), and the next best performer is considered.

3.8 PROPULSION SYSTEM COMMONALITY

The areas of analysis of propulsion system commonality were limited to engine thrust level. The approach taken was to assume that each mission could be accomplished with engines at different thrust levels either singly or in clustered configurations. Thrust levels of 2,000, 8,000, and 30,000 lb were assumed with either one, two, or three of these thrust levels available to support the first nine missions shown in Table 2.

Table 19 presents the missions investigated, their payloads, nominal ΔV requirements, and nominal thrust used for the baseline vehicle analysis. In addition, the engine thrust and number of engines required for the single-thrust-level, two-thrust-level, and three-thrust-level cases are shown. The single-level case has an 8,000-lb-thrust engine, the two-level case has 2,000- and 8,000-lb-thrust engines, and the three-level case has 2,000-, 8,000-, and 30,000-lb-thrust engines. The basic 8,000-lb-thrust single engine was assumed to be usable derated to 5,000-lb thrust to assess the penalty when used to replace a 2,000- or 4,000-lb-thrust nominal requirement.

Table 19

MISSION AND ENGINE MATRIX FOR THRUST COMMONALITY ANALYSIS

Mission and Stage	Payload (lb)	ΔV (ft/sec)	Nominal Thrust (lbf)	Thrust (lb × 10^{-5}) Number of Thrust Levels Available: One	Two	Three
Saturn Unmanned Orbiter	2,000	6,000	2,000	8*	2 or 8*	2 or 8*
Jupiter Unmanned Orbiter	2,000	7,600	2,000	8*	2 or 8*	2 or 8*
Venus Manned Flyby – Orbiter Probe	1,500	13,000	4,000	8*	2 (2) or 8*	2 (2) or 8*
Mars Manned Flyby – Orbiter Probe (Stage 1)	1,000	10,500	8,000	8	8	8
Mars Manned Flyby – Orbiter Probe (Stage 2)	1,000	10,500	4,000	8*	2 (2) or 8*	2 (2) or 8*
Mars Manned Lander – MEM Ascent Stage	5,260	16,000	30,000	8 (4)	8 (4)	30
Venus Unmanned Orbiter	7,000	13,500	8,000	8	8	8
Mars Unmanned Orbiter	8,143	6,950	8,000	8	8	8
Lunar Manned Surface Station – Return Stage	19,340	9,186	15,000	8 (2)	8 (2)	8 (2) or 30
Earth Manned Synchronous Orbiter – Descent Stage	13,000	9,750	20,000	8 (2)	8 (2)	8 (2) or 30

* Used derated to 5,000-lb thrust.
( ) Number of engines used in a cluster.

The number of cases was limited by considering only four representative propellant combinations for each mission. The engine weights and specific impulse assumed are shown in Table 20. The values assumed for the 8,000-lb- and 30,000-lb-thrust engines were those used in the baseline Mars Orbiter (bell engine) and Mars Excursion Module (Aerospike engine), respectively. A 10-percent engine weight penalty was assumed for clustered engine configurations.

In the analysis, no alternative engine configurations were considered for the single-thrust-level case where the nominal thrust for the mission was 8,000 lb. The significant result was the insensitivity to variations in thrust level that was revealed. When only an 8,000-lb-thrust engine was assumed available, the penalty in initial weight never exceeded 3 percent of the weight for a system with the nominal engine thrust for the cases investigated. Availability of two thrust levels of 2,000 and 8,000 lbf or three thrust levels of 2,000, 8,000, and 30,000 lbf reduced the penalty to the range of 1 to 2 percent. The analysis results showed no clear relationship between thrust level and choice of propellant.

It should be noted that there are some severe limitations placed upon the results because of the nature of the data. The data used were received from two manufacturers, and both bell and aerospike engines were used. Engine characteristics for most propellants with thrust levels below 8,000 lb were extrapolated. In addition, volume and dimensional limits for the various engine-stage combinations were not determined.

3.9 SIMPLICITY/COMPLEXITY AND OPERATIONAL FLEXIBILITY

Space-storable propulsion stages are more complex than a competitive earth-storable stage, but less complex than systems using hydrogen as the fuel. A quantitative evaluation of complexity would require more data and analyses than are available at the present time; however, a qualitative comparison is presented in Table 21.

Table 20

ENGINE SYSTEM CHARACTERISTICS FOR COMMONALITY ANALYSIS

Propellant	2,000-lb-Thrust Bell Engine		8,000-lb-Thrust Bell Engine			30,000-lb-Thrust Aerospike Engine	
	Specific Impulse (sec)	Engine Weight (lb)	I_{sp} (sec) (Derated to 5,000 lbf)	I_{sp} (sec) (8,000 lbf)	Engine Weight (lb)	Specific Impulse (sec)	Engine Weight (lb)
F_2/H_2	461	41	464	468	152	463	440
O_2/H_2	447	41	448	451	152	449	520
FLOX/CH_4	401.5	41	406	410	152	400	440
N_2O_4/A-50	331	41	334	335	158	331	440

Table 21

SIMPLICITY/COMPLEXITY COMPARISON

Item	Earth Storables	Space Storables	Cryogenics
Ground Conditioning	Simple – cap-off days before launch Temperatures easily controlled	Refrigeration or tank topping required Cap-off 0 to 10 hr before liftoff at small penalty Insulation purge with dry nitrogen or dry air required to prevent condensation	Tank topping required until just prior to liftoff Insulation purge with helium required to prevent condensation
Pressurization System	Pressurant stored external to propellant tanks	Pressurant stored in coldest propellant	Pressurant stored in hydrogen tank Hot hydrogen bleed from engine used during burn
Plumbing System	Shutoff valves required only at the engine and fill lines	Shutoff valves required at tanks, engine, and fill lines Insulated, low-heat-leak lines required	Shutoff valves required at tanks, engine, and fill lines Insulated, low-heat-leak lines required Zero-g vent valve may be required
Thermal Control In Space	Simple – minimum thickness insulation and exposure to sun and spacecraft equipment to prevent freezing Low tank pressures	Multilayer superinsulation, low-heat-leak tank support structures and penetrations, highly efficient surface coatings, or shadow shields required Optimum at high tank pressures	Thick multilayer superinsulation, low-heat-leak tank support structures and penetrations, highly efficient surface coatings required. Shadow shields or shielding by spacecraft orientation advisable. Optimum at high tank pressures Subcooled or slush hydrogen helpful Venting of hydrogen helpful
Ignition	N_2O_4/A-50 and ClF_5/MHF-5 are hypergolic	Combinations studied are hypergolic except ignition device required for OF_2/CH_4	F_2/H_2 is hypergolic Ignition device required for O_2/H_2
Thermal Stability	Stability of N_2O_4 and A-50 at elevated temperatures (well beyond normal storage temperatures) is poor	Stability of B_2H_6 at elevated temperatures (well beyond normal storage temperatures) is poor	Chemically stable

Operational flexibility is very much mission and vehicle dependent, and the propellant selected adds another dimension to the problem. In general, earth-storable propellants can be stored and handled on the ground with a minimum of specialized facilities. In space they must be oriented to the sun and/or insulated to prevent them from freezing. Space storables require some conditioning on the ground and require insulation and orientation for maximum performance. The cryogens are the most sensitive to ground handling and storage, and require insulation and orientation and, in some cases, venting in order to maximize their performance. It also appears that the more sensitive propellants with lesser operational flexibility exhibit the higher performance.

3.10 TECHNOLOGY DEVELOPMENT REQUIREMENTS

There is need for continuing the development of fluroine and fluorinated oxidizer technology in all aspects. The primary requirements revealed by the study are related to thermal storage. These requirements fall in the following categories:

(1) Requirements and problems associated with handling and storage of propellants throughout all operational phases. Development needs are identified for:

- Simple, efficient, condensation-free ground thermal conditioning systems
- Simple, lightweight, economical surface coatings, with α/ϵ comparable with Lockheed Optical Solar Reflector (OSR), for use on large surfaces
- Evacuated multilayer insulation for use in planetary atmospheres
- Further development of the structural requirements and evaluation of the efficiency of multilayer superinsulation in thicknesses from 2 to 5 in. (this item is required for hydrogen systems only)
- Demonstration of venting times for multilayer insulation under simulated launch conditions

(2) Requirements associated with development of lightweight tanks and structural components. A need is seen for continued development, although no major problems are evident in this area. Work should be continued on:

- Lightweight tanks
- Lightweight, low-heat-leak support struts
- Low-heat-leak tank penetrations
- Materials compatibility

Technology development requirements related to rocket engine systems were not subjects for detailed analyses but were determined from inputs supplied by the supporting engine companies. These include:

- Low-flow, high ΔV fuel pumps and small turbines
- System tests to confirm realistically attainable performance
- Optimum nozzle cooling technique development for each propellant
- Two-step throttling, restart, and control of small engines

Section 4
CONCLUSIONS

The following conclusions were reached at the completion of the Space Storable Propellant Selection Study:

(1) Space-storable and deep cryogenic propellants have attractive performance potential for a wide range of space missions requiring thrust levels between about 1,000 and 50,000 lbf. No serious propellant storage problems are foreseen.

- Space-storables outperform earth-storables by from 20 to 45 percent, and F_2/H_2 outperforms earth-storables by from 5 to 50 percent for the missions considered as attractive candidates.
- The highest performance for the Mars Orbiter and the MEM Ascent Stage is obtained with F_2/H_2 in pump-fed systems.
- The highest performance for a pressure-fed Mars Orbiter is obtained with OF_2/B_2H_6.

(2) Space-storables and deep cryogens are relatively insensitive to variations in spacecraft orientation, mission length, and degradation of insulation and surfact coatings. Hydrogen is more sensitive than space-storables to these variables.

- All propellants studied can be flown nonvented for the Mars Orbiter and MEM Ascent Stage missions. If hydrogen tanks must be exposed to the sun, then venting of hydrogen for the Mars Orbiter reduces insulation thickness required from 4-1/2 in. to under 2 in. and results in a stage weight saving of 4-1/2 percent for F_2/H_2 and 5-1/2 percent for O_2/H_2.
- A preferred spacecraft orientation is desirable but not mandatory. Shielding the nonvented propulsion system of the Mars Orbiter from the sun resulted in a reduction of insulation thickness from 4-1/2 in. to 2 in. or less, and in a stage weight saving of about 2 percent for the space-storables, 6-1/2 percent for F_2/H_2, and 7-1/2 percent for O_2/H_2.

(3) Pump-fed systems are mandatory for the volume- and dimension-limited MEM Ascent Stage. A toroidal engine is also required if a single engine is specified.

(4) Insulation for the space-storable and deep cryogen propellant tanks for the MEM Ascent Stage must remain evacuated while on the surface of Mars.

(5) An O_2/H_2 system will not meet the mission requirements for the specified MEM configuration. Propellant that can be packaged within the volume limits is insufficient to provide the required ΔV.

(6) A space-storable or F_2/H_2 engine system in the 5,000- to 10,000-lb-thrust range, used singly or clustered, could meet the performance requirements of most of the missions analyzed, while incurring a stage weight penalty never exceeding three percent of the nominal. Further analysis is required to identify those cases where engine dimensions would exceed limits.

(7) Mars Orbiter pump-fed systems tend to optimize with tradeoffs between insulation and vapor weights. Pressure-fed systems tend to optimize with tradeoffs between pressurant system weight and tank weight and, to a lesser extent, insulation and vapor. The MEM Ascent Stage optimized with tradeoffs between insulation and pressurant weights for the outer (droppable) tanks and with insulation and vapor weights for the inner (core) tanks.

(8) Operational considerations include:

- A helium purge of the insulation is required during ground hold for the H_2 propellant and dry nitrogen or dry air for the space storables.
- Closed-loop vent or refrigeration systems are required for the fluorinated oxidizers (F_2, FLOX, and OF_2) during ground operations.
- To compensate for ground-hold propellant temperature rise, subcooling requirements range up to a maximum of 5° F for the cryogens for each hour from topping termination to liftoff. Additional subcooling could also reduce the system weights for cryogens and space-storables.
- A typical flight-type multilayer insulation installation will vent adequately in a typical booster trajectory following proper ground-hold conditioning. Ascent heating of ullage gas will not cause unacceptable pressure rises.

Section 5
RECOMMENDATIONS

The following recommendations are made on the basis of the Space Storable Propellant Selection Study results:

(1) A more detailed analysis of prelaunch, on-pad, and launch operations concerning propellant loading, topping, venting, thermal conditioning, payload accessibility, recycle, etc., would aid in the comparison of propellants.

(2) An effort should be made to narrow the matrix of space-storable propellant combinations, blends, etc., and to establish reliability and feasibility of pump vs. pressure feed, competitive cooling methods, practical chamber pressure design points, and throttleability characteristics. This would require critical analysis of all space-storable (and related) study and technology work accomplished to date.

(3) Study of the application of a common propulsion stage for multiple applications would be helpful in assessing propellant sensitivity to mission and design variables.

(4) A more detailed tradeoff analysis of primary vs. secondary propulsion for small velocity corrections would aid in propellant sensitivity assessments.

(5) The thermal storage impact of a common bulkhead for propellants with non-overlapping liquidus ranges should be assessed.

(6) The fabrication and structural practicality and the thermal efficiency of superinsulation in thicknesses up to 5 in. should be demonstrated.

(7) A more detailed study of the efficiency of local shadow shielding of propellant tanks should be performed.

(8) The development of surface coatings with the efficiency of the Lockheed Optical Solar Reflector ($\alpha/\epsilon = 0.06$) with lighter weight, easier handling, and lower cost should be continued (now underway at Lockheed).

(9) The performance effects of auxiliary thrust systems and propellant settling systems as compared with idle-mode engine start should be investigated.

(10) Additional analysis of propulsion-system complexity for various propellant/engine system combinations should be performed to aid in reliability comparisons.

(11) The leak-rate sensitivity effects for the various propellants should be analyzed.

(12) The practical design problems of developing evacuated superinsulation for use in planetary atmospheres should be investigated more fully.

(13) Better propellant-property data should be developed, particularly vapor enthalpy, internal energy, and heat-of-vaporization as a function of pressure

(14) Efforts in the rocket-engine area should include:

- Development of low-flow, high ΔP fuel pumps and small turbines
- System tests to establish realistically attainable performance for each propellant
- Development of optimum nozzle cooling technique for each propellant
- Demonstration of two-step throttling, restart, and control of small engines
- Better propellant-property data should be developed, particularly vapor enthalpy, internal energy, and heat-of-vaporization as a function of pressure