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AN ANALYTICAL STUDY OF STORAGE OF LIQUID-HYDROGEN PROPELLANT FOR NUCLEAR INTERPLANETARY SPACECRAFT

FIRST QUARTERLY PROGRESS REPORT
(30 June 1967 through 30 September 1967)

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This report covers the work performed during the period 30 June 1967 through 30 September 1967 under contract NAS8-21080, "An Analytical Study of Storage of Liquid Hydrogen Propellant for Nuclear Interplanetary Spacecraft." The program is being conducted by the Fort Worth Division of General Dynamics Corporation for the Propulsion and Vehicle Engineering Laboratories of the NASA George C. Marshall Space Flight Center. The program is under the technical direction of Messrs. Robert Middleton and David Price of NASA/MSFC.

The basic objective is to assess the problem of extended Earth orbital storage of liquid hydrogen propellant for a conjunction-class manned Mars vehicle. Other study parameters include the insulation thermal conductivity-density product, Mars orbit altitude, and the type of thermal management system - vented, nonvented, partial recondensation, or combination systems. During the first quarter the study ground rules were established and the preparatory work for the preliminary parametric analysis was largely completed. This comprised mission analysis, structural analysis, definition of the nominal or baseline vehicle, thermal analysis, and preparation of computer programs to accomplish the parametric analysis.

1.0 STUDY GROUNDRULES

This study is based upon the groundrules enumerated below:

1. Conjunction-class manned Mars vehicle will be based on the Lockheed Modular Nuclear Vehicle Study (Phase II).
2. Mission will be similar to that studied under contract NAS8-11161 with the exception that the Earth orbit altitude will be 485 km instead of 185 km.
3. Propellant initial thermodynamic state will be triple point liquid.
4. Required NPSP is 5.0 psi which includes pressure rise due to nuclear heating.
5. Ranges of variables:
   a. Earth orbit storage time: 90, 180, 270 days
   b. Mars orbit altitude: 17090 km (synchronous), two other values determined during mission analyses.
   c. Insulation $k\rho$ product: 5×10^{-6}, 7.5×10^{-5}, 1.5×10^{-3} $\frac{\text{BTU lb}}{\text{hr ft}^4\ ^{\circ}\text{R}}$
6. Tanking operation takes place just prior to earth departure and requires zero time.
7. Material allowables will be evaluated at 530°R.
8. Pressurant gas inlet temperature will be 530°R.

9. Solar shield data will be based on the results of contract NAS8-11317 and current contract NAS8-21132.

10. The English system of units will be the prime unit system with metric equivalents in parentheses in the final report.

2.0 MISSION ANALYSIS

The mission analysis conducted to date has resulted in the mission definition given below, and the mass variation with altitude of the Mars Excursion Module (MEM).

Mission definition

The mission is a conjunction-class manned Mars stopover mission with a total duration of 1020-1200 days, depending upon the Earth orbital staytime (see study groundrules). A graphical presentation of the mission is presented in Figure 1 and the variation of solar distance is presented in Figure 2. Additional mission data is tabulated in Table 1. Mars orbit altitudes of 216 and 3238 nm were selected for study in addition to the 9203 nm synchronous altitude. The smaller value is the lowest altitude considered feasible for a 510-day duration based on estimated orbital decay rates (upper density model atmosphere, Reference 3). The 3238 nm value results from estimates of the variation with altitude of the total ΔV for the Mars Braking and Mars Escape Stages and the mass of the Mars Excursion Module. This value yields a near minimum ΔV and allows a high-inclination orbit which is desirable for more complete planet coverage.

The orientation of the orbit with respect to the terminator for the three selected capture orbits at Mars was determined by selecting the inclination and computing the nodal precession rate.

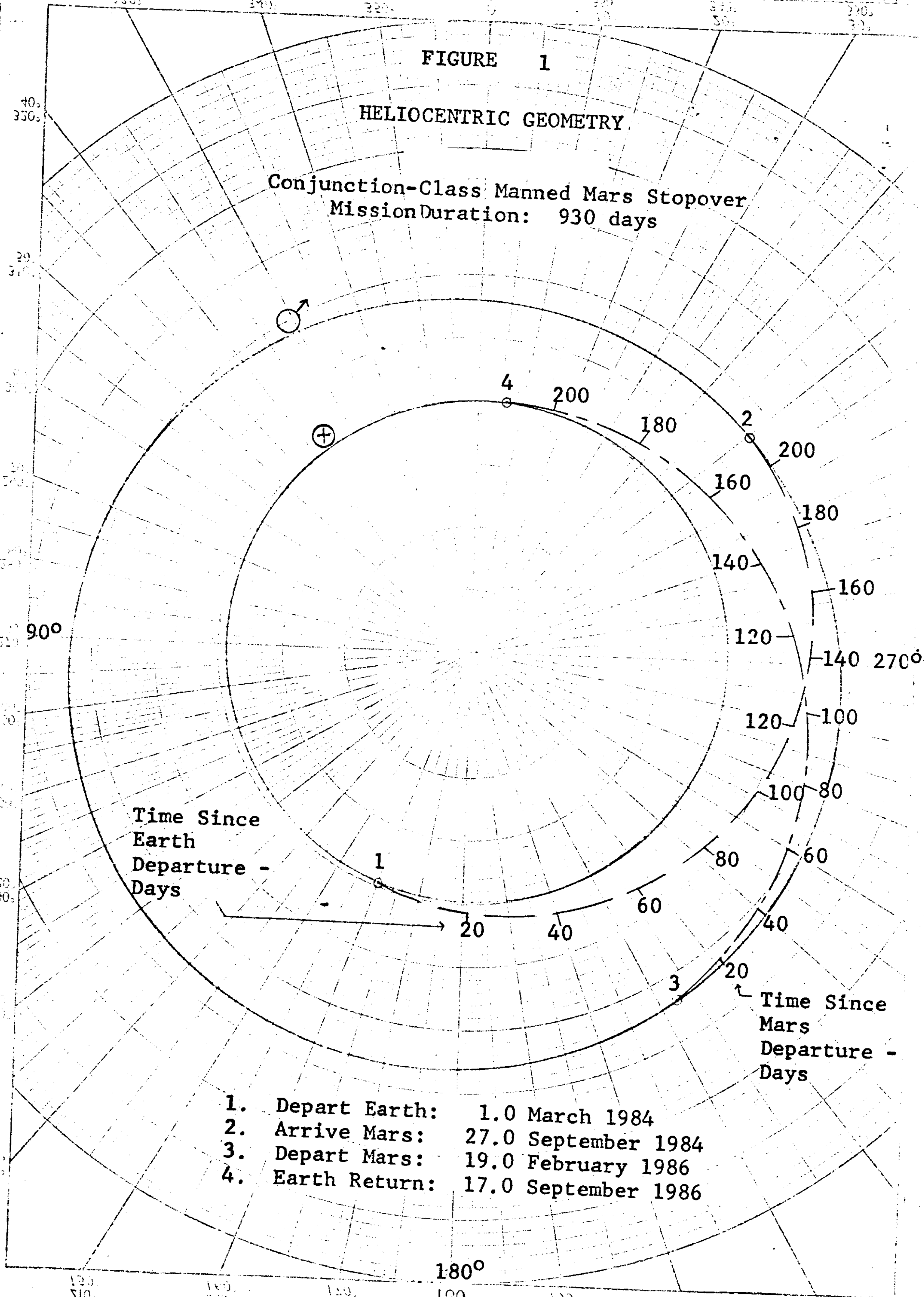


FIGURE 1

HELIOCENTRIC GEOMETRY

Conjunction-Class Manned Mars Stopover
Mission Duration: 930 days

1. Depart Earth: 1.0 March 1984
2. Arrive Mars: 27.0 September 1984
3. Depart Mars: 19.0 February 1986
4. Earth Return: 17.0 September 1986

FIGURE 2. VARIATION OF SOLAR DISTANCE WITH FLIGHT TIME

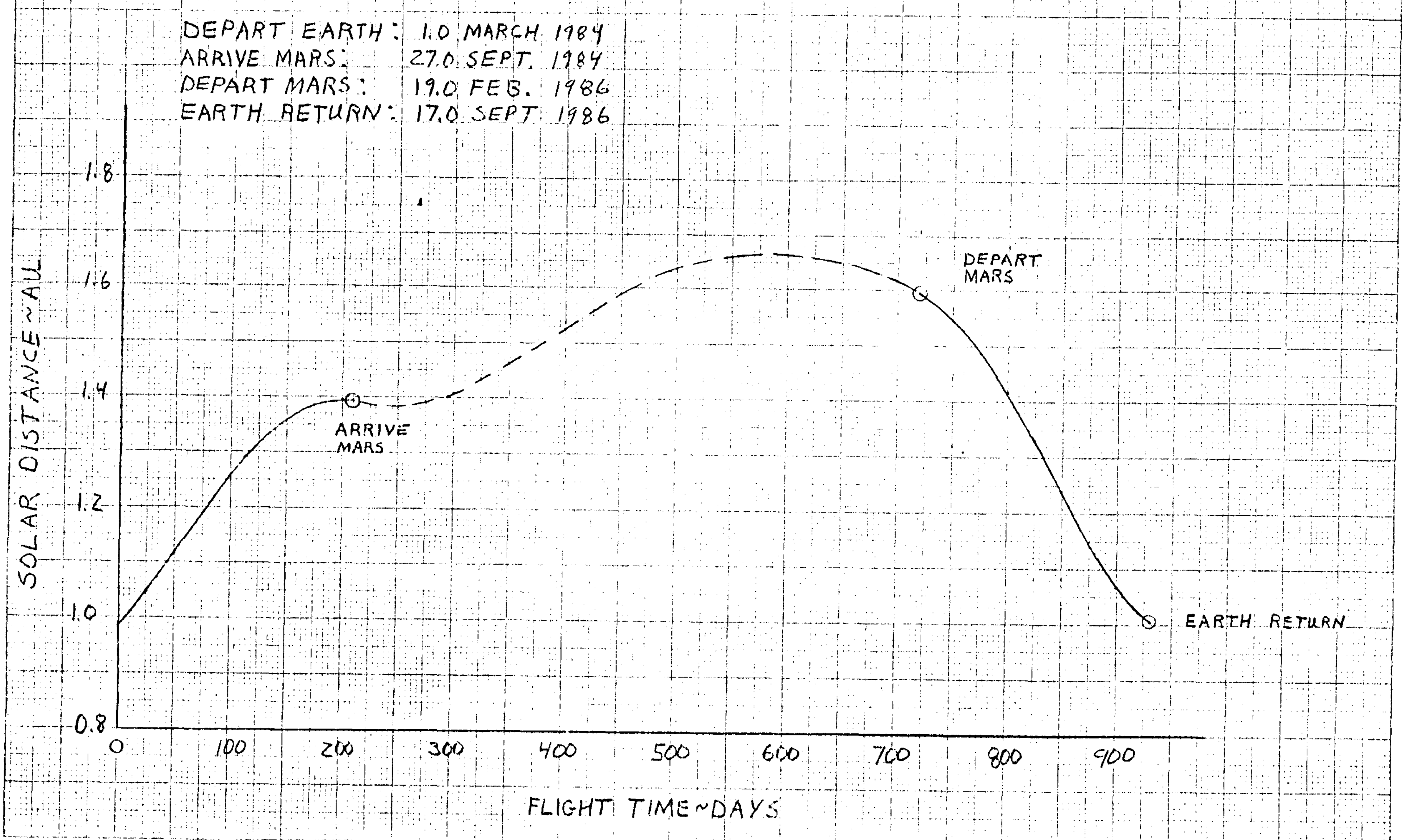


Table 1. Conjunction-Class Manned Mars Stopover Mission Summary

Depart Earth:	1.0 March 1984
Arrive Mars:	27.0 September 1984
Depart Mars:	19.0 February 1986
Return Earth:	17.0 September 1986

Outbound Flight Time:	210.0 Days
Stay Time:	510.0 Days
Inbound Flight Time:	210.0 Days

Planetocentric (Earth) Departure Phase

Parking Orbit Altitude: 262 N.Mi. = 485 KM (circular)
Hyperbolic Excess Speed: .1270 EMOS = 12,410 ft/sec
Declination of Departure Asymptote: -35.71 deg
Right Ascension of Departure Asymptote: 182.43 deg
Parking Orbit Inclination: 36.0 deg

Heliocentric Phase (outbound leg)

Heliocentric Transfer Angle:	148.89 deg
Inclination of Transfer Orbit:	3.53 deg
Eccentricity of Transfer Orbit:	.1835
Perihelion Distance:	.9621 AU (no parihelion transit)
Aphelion Distance:	1.3946 AU

Planetocentric (Mars) Arrival Phase

Parking Orbit Altitude: Selected*
Hyperbolic Excess Speed: .1272 EMOS = 12,430 ft/sec
Declination of Arrival Asymptote: 4.51 deg
Right Ascension of Arrival Asymptote: 316.53 deg
Parking Orbit Inclination: (at least) 9.62 deg*

* The selected circular orbit altitudes and inclinations are:

Altitude	Inclination
9203 N.Mi. = 17,053 KM	10.7 deg
3238 N.Mi. = 6,000 KM	63.0 deg
216 N.Mi. = 400 KM	75.2 deg

Table 1. (Con't)

Planetocentric (Mars) Departure Phase

Parking Orbit Altitude: Selected*
Hyperbolic Excess Speed: .0813 EMOS = 7945 ft/sec
Declination of Departure Asymptote: 9.62 deg
Right Ascension of Departure Asymptote: 212.72 deg
Parking Orbit Inclination: (at least) 9.62 deg*

Heliocentric Phase (Inbound Leg)

Heliocentric Transfer Angle: 141.77 deg
Inclination of Transfer Orbit: .894 deg
Eccentricity of Transfer Orbit: .2396 deg
Perihelion Distance: .9948 AU (no perihelion transit)
Aphelion Distance: 1.618 AU (No aphelion transit)

Planetocentric (Earth) Return Phase

Direct Reentry
Unbraked Entry Speed: 11.6801 km/sec = 38,321 ft/sec
Hyperbolic Excess Speed: .12350 EMOS = 12,068 ft/sec
Declination of Arrival Asymptote: 14.03 deg
Right Ascension of Arrival Asymptote: 110.25 deg

* The selected circular orbit altitudes and inclinations are:

Altitude	Inclination
9203 N.Mi. = 17,053 KM	10.7 deg
3238 N.Mi. = 6,000 KM	63.0 deg
216 N.Mi. = 400 KM	75.2 deg

The inclinations were selected so that the orbits were coplanar with both the arrival and departure hyperbolic trajectories at Mars. Only posigrade orbits were considered. The nodal precession rate was determined from the following equation:

$$\dot{\lambda}_p = \frac{-3\pi J_2}{T(R/R_o)^2} \sin \delta_p$$

where

$\dot{\lambda}_p$ = nodal precession rate

J_2 = coefficient of the second harmonic of the planet's gravitational potential

T = orbit period

R = orbit radius

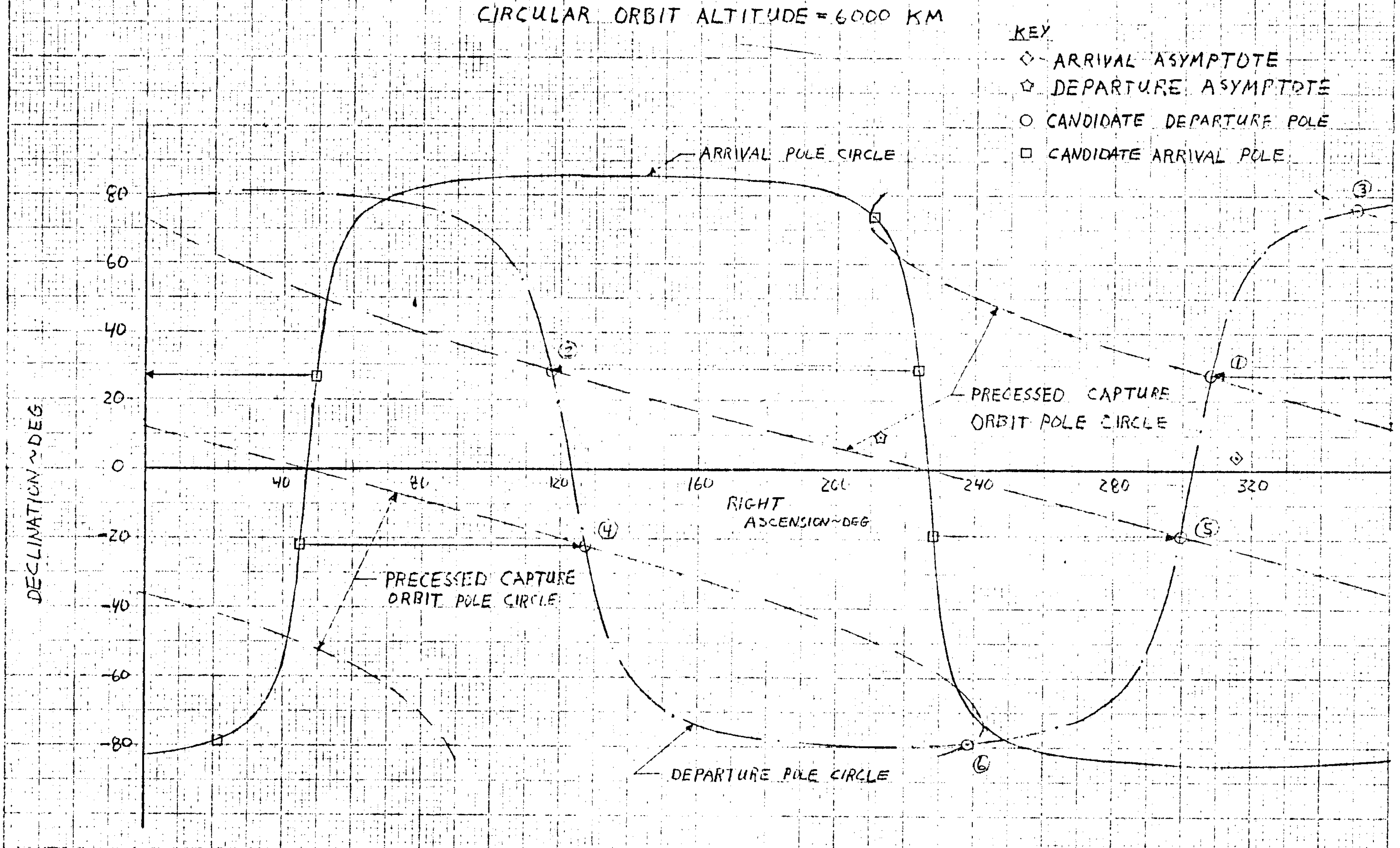
R_o = planet radius

δ_p = orbit pole declination.

Although this is only a first-order approximation of $\dot{\lambda}_p$, it is felt that the expression is sufficiently accurate for purposes of this study.

The orbit orientation analysis for the 3238 nm orbit is presented in Figure 3. The arrival and departure pole circles as well as the precessed capture orbit pole circle are shown. No plane change is required at departure at the points where the precessed capture orbit pole circle intersect the departure pole circle. The number of possible poles is six. Pole numbers 4,

FIGURE 3. MARS CAPTURE ORBIT ORIENTATION ANALYSIS



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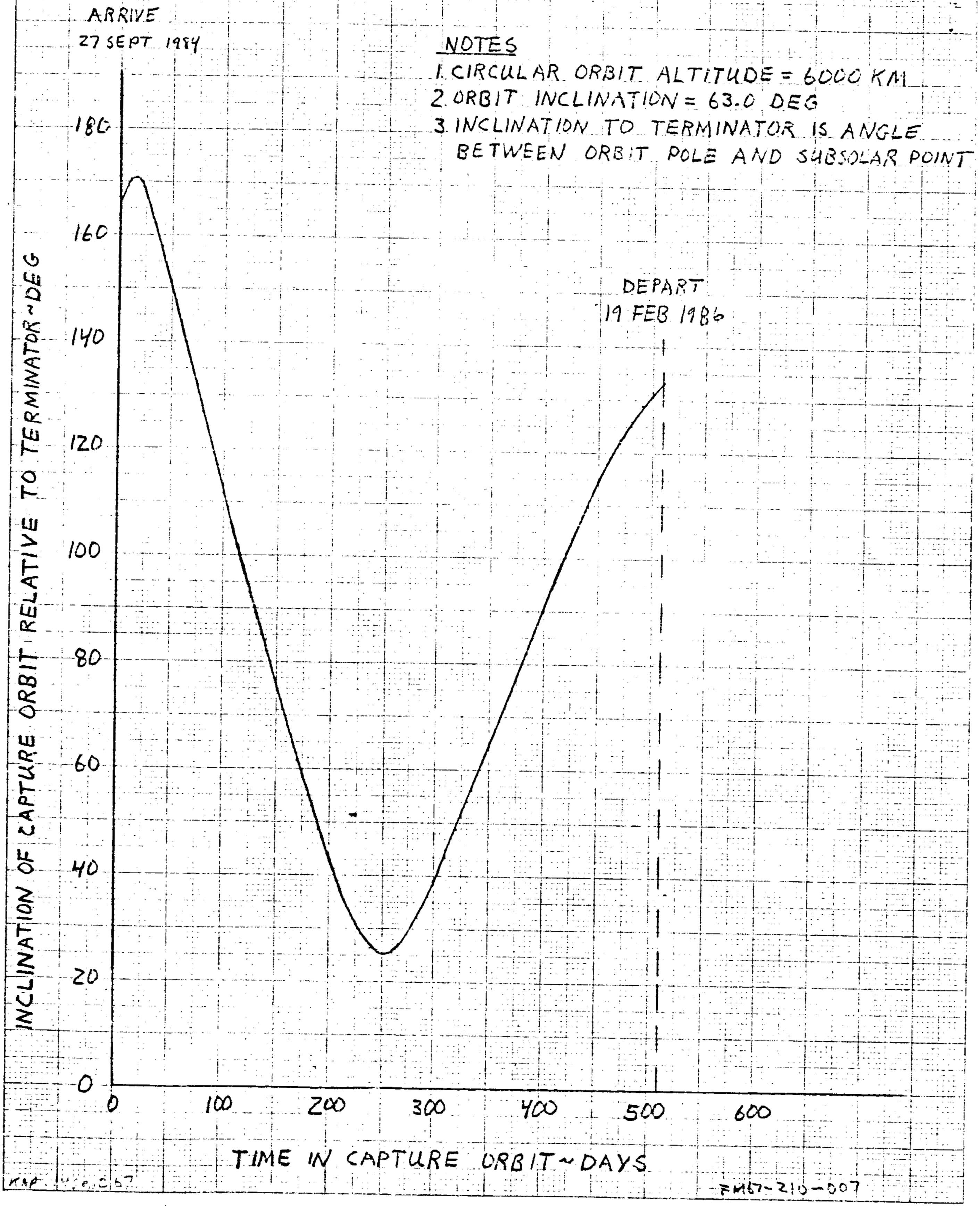
5, and 6 were eliminated because the orbits for these poles would be retrograde. Pole number 1 was selected because the corresponding capture orbit offers the most favorable orientation for planet reconnaissance, i.e., maximum inclination. A time history of the inclination of the parking orbit to the terminator for a stay time of 510 days is presented in Figure 4. Similar data were generated for the 216 nm orbit and the 9203 nm (synchronous) orbit.

The relatively high precession rate for the 216-nm capture orbit allows the selection of many inclinations. Although the maximum possible inclination is nearly polar, an orbit inclination of 75.2 degrees was selected. This orbit inclination provides good surface coverage and allows the selection of a landing site over a wide range of latitudes. This choice of inclination is not unique, but it is felt that this inclination is a reasonable choice for a low-altitude Mars orbit.

MEM Mass Analysis

Trajectory/vehicle analysis of the Mars Excursion Module (MEM) was aimed at providing the variation of the MEM mass with circular orbit altitude. The main effort was devoted to determining the mass variation with a consistent set of assumptions, and no attempt was made to optimize mass, the propulsion system, or the energy requirements. The analysis was based on the MEM

FIGURE 4. MARS CAPTURE ORBIT ORIENTATION HISTORY

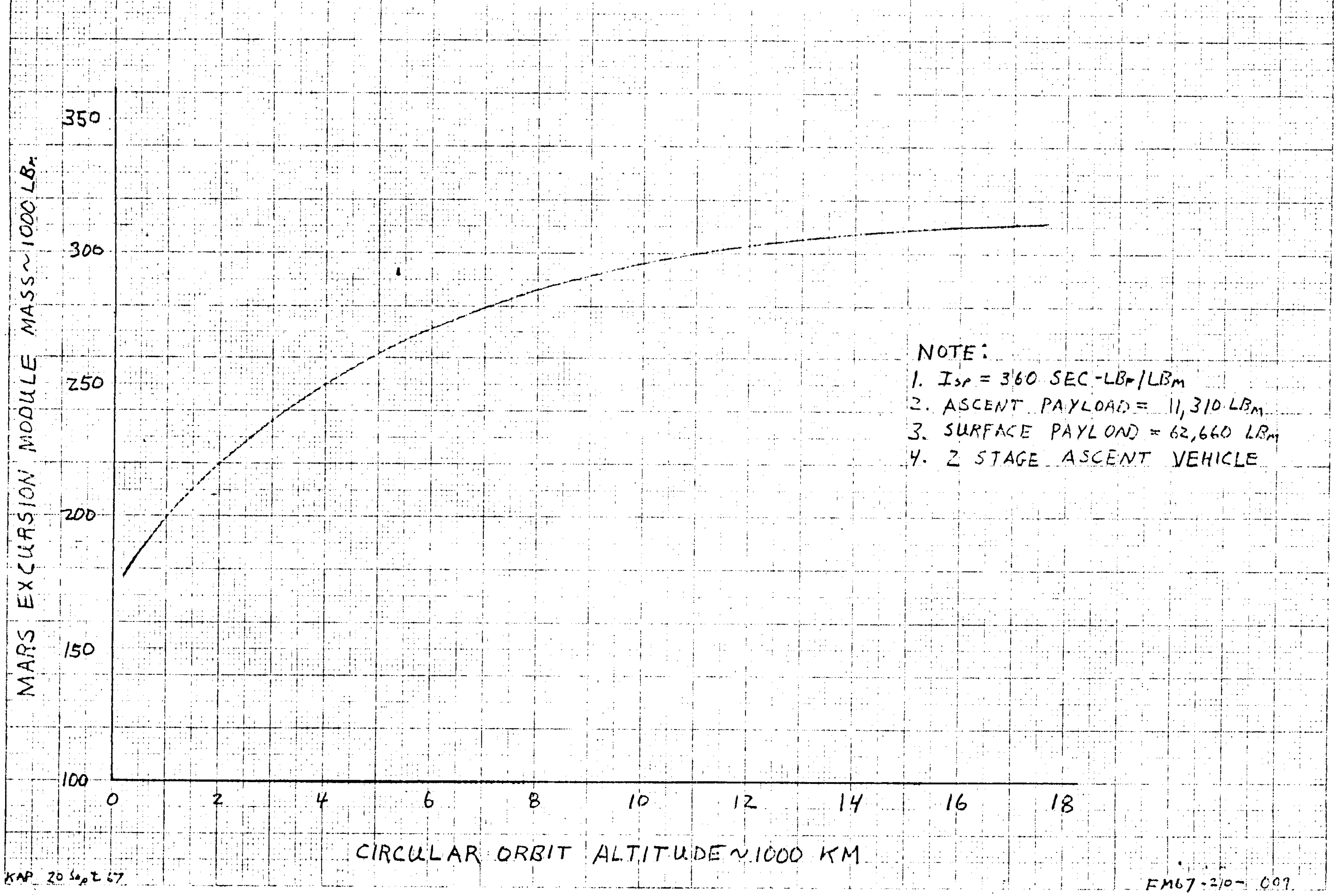


defined in Reference 1 with the most important deviation being the choice of propellant. Storable propellants were assumed rather than cryogenic propellants because of the long staytime on the planet. The specific impulse of both the ascent and descent vehicle propulsion system was assumed to be 360 sec-lb_f/lb_m. Figure 5 shows the variations of MEM mass with Mars orbit altitude.

A two-stage ascent vehicle was defined with direct ascent to 185 km followed by a Hohmann transfer to the selected parking orbit. The ΔV to ascend to 185 km was assumed to be 15,000 ft/sec. The ΔV for the Hohmann transfer was multiplied by a factor of 1.05 to account for losses, plane change requirements, and terminal rendezvous requirements. A payload of 11,310 lb_m and a propellant mass fraction of 0.90 was assumed.

The sequence of events for the descent vehicle consists of a transfer from the parking orbit to the entry altitude (100 km), lifting entry, propulsive braking, hover, translation, and landing. Parachute braking was not considered since the data of Reference 2 indicates that it may not be feasible. The de-orbit ΔV was assumed to be tangential (180° degrees to the velocity vector). The entry angles were selected to be near a skipout boundary which was defined as the minimum angle where the altitude is always decreasing for ballistic entry into the lower density

FIGURE 5. MEM MASS VS. CIRCULAR ORBIT ALTITUDE



model atmosphere of Reference 3. A fixed ΔV allowance for propulsive braking, translation, and hovering was assumed because it is felt that the primary effect of orbit altitude on ΔV requirements should be reflected in the ascent Hohmann transfer ΔV and the descent de-orbit ΔV. Propulsive braking, translation, and hovering requirements are primarily dependent upon entry and final touchdown techniques, e.g., L/D modulation, specified distance (translation), and time (hover), rather than on orbit altitude. The required propulsive braking ΔV was assumed to be 1640 ft/sec (based on data in Reference 2) and the ΔV allowance for hover and translation was assumed to be 1300 ft/sec (Reference 1). A propellant mass fraction of 0.85 was used.

The payload of the descent vehicle consists of the surface payload and the ascent vehicle. The mass of the surface payload was assumed to be 62,660 lb_m. In addition, a landing gear mass equal to one percent of the landed mass and a heat shield mass equal to 3.5% of the gross MEM mass were included. Part of the heat shield was assumed to be ablated and/or jettisoned prior to propulsive braking.

3.0 STRUCTURAL ANALYSIS

Structural analysis was directed toward determination of the tank pressure mass relationship and the meteoroid protection criteria. Following the Lockheed Modular Nuclear Vehicle Study, the basic structural concept is the load-carrying shell module in which the shell is jettisoned just prior to Earth departure.

Tank Mass

For the aluminum propellant tank with ellipsoidal heads, the pressure mass relationship is given by the following equation:

$$M_t = \frac{SF \cdot CF \cdot \rho_t a^2}{\sigma}\left[\pi \frac{a^2}{b} + \frac{\pi b}{2e} \ln\left(\frac{1+e}{1-e}\right)\right]\left[P + \rho_H n \left(\frac{L+2b}{2}\right)\right]$$

where

SF = safety factor (1.4)

CF = contingency factor

ρ_t = tank material density

a = semi-major axis

b = semi-minor axis

e = eccentricity

σ = tensile stress

P = ullage pressure

ρ_H = propellant density

n = acceleration, g's

L = cylindrical section length

The contingency factor accounts for mass that is proportional to tank pressure mass including the thrust structure, forward thrust cone, tank support cone, weld material and baffles. Individual analyses of the mass variations of these items is well beyond the scope of this study and thus necessitates the use of a contingency factor. Based on the Modular Nuclear Vehicle study, a contingency factor of 1.75 was selected for the parametric analysis.

Meteoroid Protection

Meteoroid protection criteria were founded upon the utilization of a jettisonable meteoroid bumper (Mars Braking and Mars Departure Stages only) with the tank wall. The ascent shell furnishes the additional protection required in Earth orbit. The Earth Departure Stage modules are protected by the ascent shroud and tank wall alone. The three-sheet, foam-filled meteoroid bumper is sized according to the requirements for the interplanetary portion of the mission and is jettisoned one hour prior to engine ignition on that particular stage. Similarly, the ascent shrouds are jettisoned from all modules one hour prior to Earth departure. For each tank then, the tank wall serves as the meteoroid protection for a one-hour period. The basic guideline for the analysis was a 0.995 probability of no penetrations.

The meteoroid fluxes, both cometary and asteroidal, were based on Reference 3. These fluxes were considered constant and evaluated for time-averaged values of solar distance. The flux equations are:

$$\log N = -1.34 \log m - 14.33 \qquad \text{(Cometary)}$$

$$\log N = - \log m - 13.45 \qquad \text{(Asteroidal)}$$

where N = (number of meteroids of mass m or greater)/m^2-sec

m = meteoroid mass, gm

Figure 6 presents the single wall thickness requirements for an aluminum wall. The curve labeled "Earth-shielded" incorporates the shielding effect of the Earth which is applicable to Earth Departure Stage modules. The curve labelled "unshielded" does not include the Earth's shielding effect and is applicable to the Mars Braking and Mars Departure Stages. To size the meteoroid bumper, the single wall thickness requirement is obtained from Figure 6 and appropriate effectiveness factors from Reference 4 are applied to determine the sheet thickness of the three-sheet, foam-filled bumper.

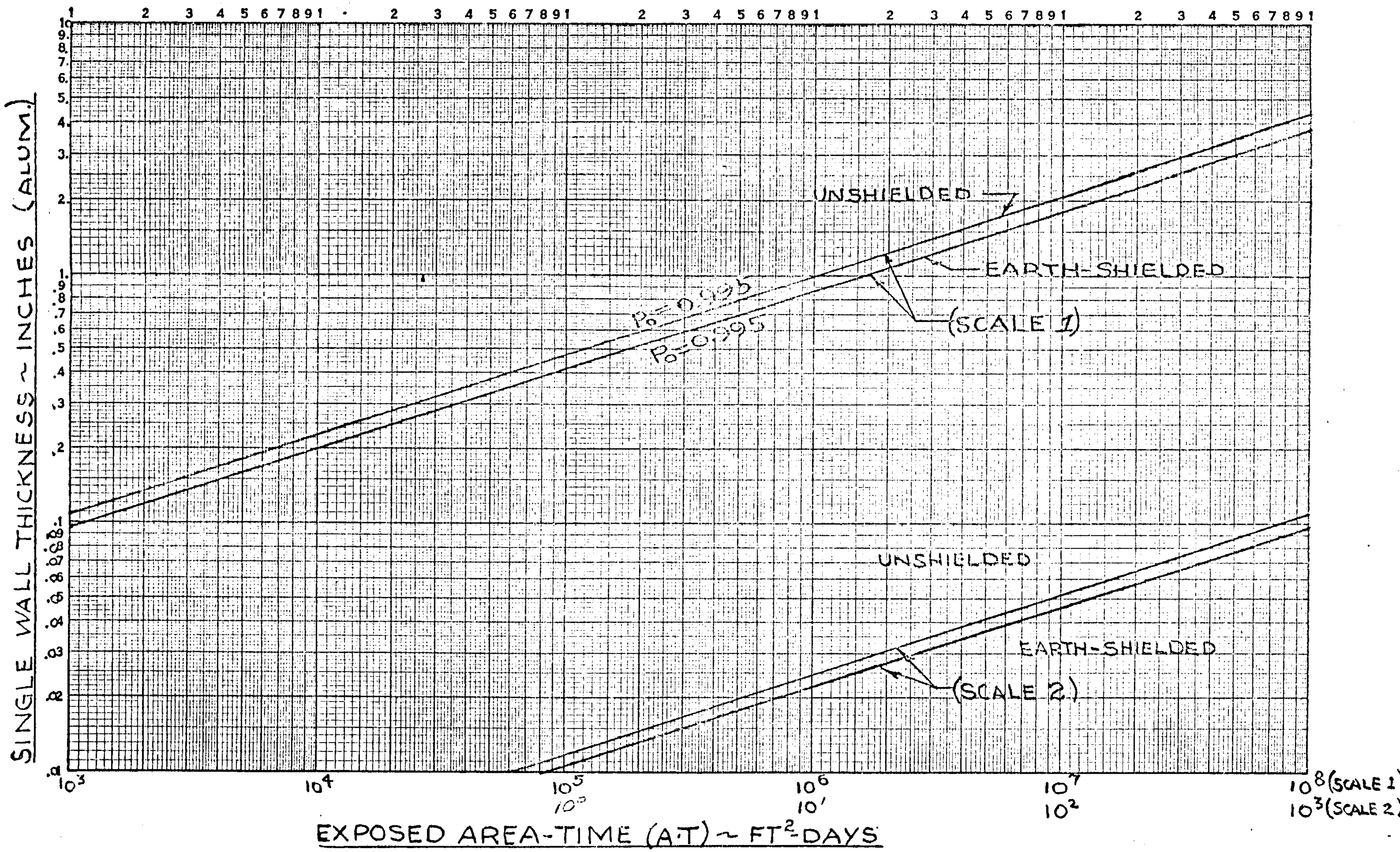


FIGURE 6. METEOROID PROTECTION REQUIREMENT

4.0 BASELINE VEHICLE

The approach used in this study to evaluate the various thermal management systems is related to the Initial Mass In Earth Orbit (IMIEO). Determination of IMIEO is accomplished by reference to a nominal or baseline vehicle whose total mass, stage masses, propellant requirements, etc., are computed with nominal values for the propellant storage system masses. This computation is done with the Rocket Vehicle Mass Buildup and Sensitivity Program, Fort Worth Division Computer Procedure V18. A discussion of the basic assumptions and a summary of the baseline vehicle data is presented below.

Basic Assumptions

Primary energy requirements for the baseline vehicle were based upon the maximum hyperbolic excess velocities for a 20-day launch opportunity at Earth and a 30-day departure opportunity at Mars. These values are 0.14, 0.15, and 0.09 EMOS at Earth departure, Mars braking, and Mars departure, respectively. The velocity changes for the primary propulsion phases were computed using these hyperbolic excess velocities with allowances for gravity losses, small plane changes, and performance reserves. Payload masses for the baseline vehicle were assumed as follows:

Mission module - 100,000 lb_m

Earth entry module - 15,000 lb_m

Solar flare shield - 16,000 lb_m

Mars scientific payload - 1500 lb_m

The Mars Excursion Module is an additional payload (see Figure 5).

Requirements for the mid-course correction system were based on Lockheed data. The mass fraction (mid-course system mass divided by vehicle mass) for both the outbound leg and the inbound leg was based on a ΔV of 500 ft/sec, an I_{sp} of 305 lb_f-sec/lb_m, and a propellant mass fraction of 0.80. Attitude-control system mass was set at one percent of the controlled-vehicle mass for both the outbound and inbound legs and at 0.2% during the stay period in Mars orbit. Engine mass and performance data are based on the Nerva engine with a mass of 35,000 lb_m and a specific impulse of 850 lb_f-sec/lb_m. Interstage masses were estimated from Lockheed data on the Modular Nuclear Vehicle.

Baseline Vehicle Data

Three baseline vehicles were defined corresponding to the three Mars orbit altitudes. The propellant loadings and mass data for each of the baseline vehicles are presented in Table 2. From the data of Table 2 and the payload capability of the Uprated Saturn V (approximately 330,000 lb_m in a 262 nm orbit), a vehicle configuration has been defined comprising a four-module Earth

Table 2. BASELINE VEHICLE DATA

Mars Orbit Altitude

	216 nm		3238 nm		9203 nm	
	Stage Mass	Useful Propellant	Stage Mass	Useful Propellant	Stage Mass	Useful Propellant
Earth Departure Stage	979958	711228	1108325	817622	1194890	889370
Mars Braking Stage	521551	252086	665052	300974	756168	347436
Mars Departure Stage	260357	59563	252081	52194	252319	52406
IMIEO	1761866 lb_m		2025458 lb_m		2203376 lb_m	

Departure Stage, a two-module Mars Braking Stage, and a single-module Mars Departure Stage. The Earth Departure Stage is made up of three propulsion modules arranged in an in-line configuration with a single propellant module stacked above the central propulsion module. The Mars Braking Stage is made up of a propellant module and a propulsion module stacked above the EDS propellant module.

5.0 THERMAL ANALYSIS

The primary effort in the thermal analysis has been to define the tank thermal model, evaluate the thermal environment for the various mission phases, and establish the vent pressure for the vented thermal management systems.

Thermal Model

For the preliminary parametric analysis, the tank thermal model is based on uniform insulation heat transfer around the tank and a constant thermal conductance associated with penetration of the insulation. The outer surface of the insulation is assumed to be at a uniform temperature equal to the adiabatic surface temperature. Insulation thermal conductivity is one of the basic study parameters and its value is determined from the $k\rho$ values established in the groundrules. The penetrations considered are tank support cone, aft skirt, engine support, engine feed line, fill and drain line, vent line, and pressurant line. Thermal conductances were calculated for the tank support cone and the aft skirt yielding values of 0.805 and 0.197 $\frac{BTU}{hr^{o}R}$ respectively. Conductance for the remainder of the penetrations were taken from Lockheed data since detailed information was not available. The combined thermal conductance for these penetrations is 0.762 $\frac{BTU}{hr^{o}R}$.

Thermal Environment

Evaluation of the tank thermal environment was accomplished in terms of the adiabatic surface temperature assumed by a cylindrical surface. These temperatures were based on a surface with a solar absorptance of 0.05 and an emittance of 0.80. The selected radiative properties correspond to the Lockheed Optical Solar Reflector surface coating. For Earth orbit, the vehicle orientation was parallel to the velocity vector in a terminator orbit. This is the most severe orientation from a thermal standpoint. However, orientation of the longitudinal axis toward the sun would not change the results appreciably because of the low orbit altitude.

For the Mars transfer mission phase, an average adiabatic surface temperature was computed using the solar distance data of Figure 2. A broadside orientation with respect to the sun was assumed. For cases where a solar shield is deployed during Mars transfer, the energy incident upon the vehicle is considered negligible except for periods of guidance correction when a broadside orientation is assumed for the total duration of the guidance corrections.

Temperatures in Mars orbit were determined by evaluating the average inclination to the terminator over the time in orbit (See Figure 4). Using this average inclination, adiabatic surface temperatures were computed for both a velocity vector orientation and orientation of the longitudinal axis toward the sun. The second case corresponds to the situation wherein a solar shield is deployed from the end of the vehicle. This orientation will be referred to as "solar orientation."

The adiabatic surface temperatures are summarized in Table 3.

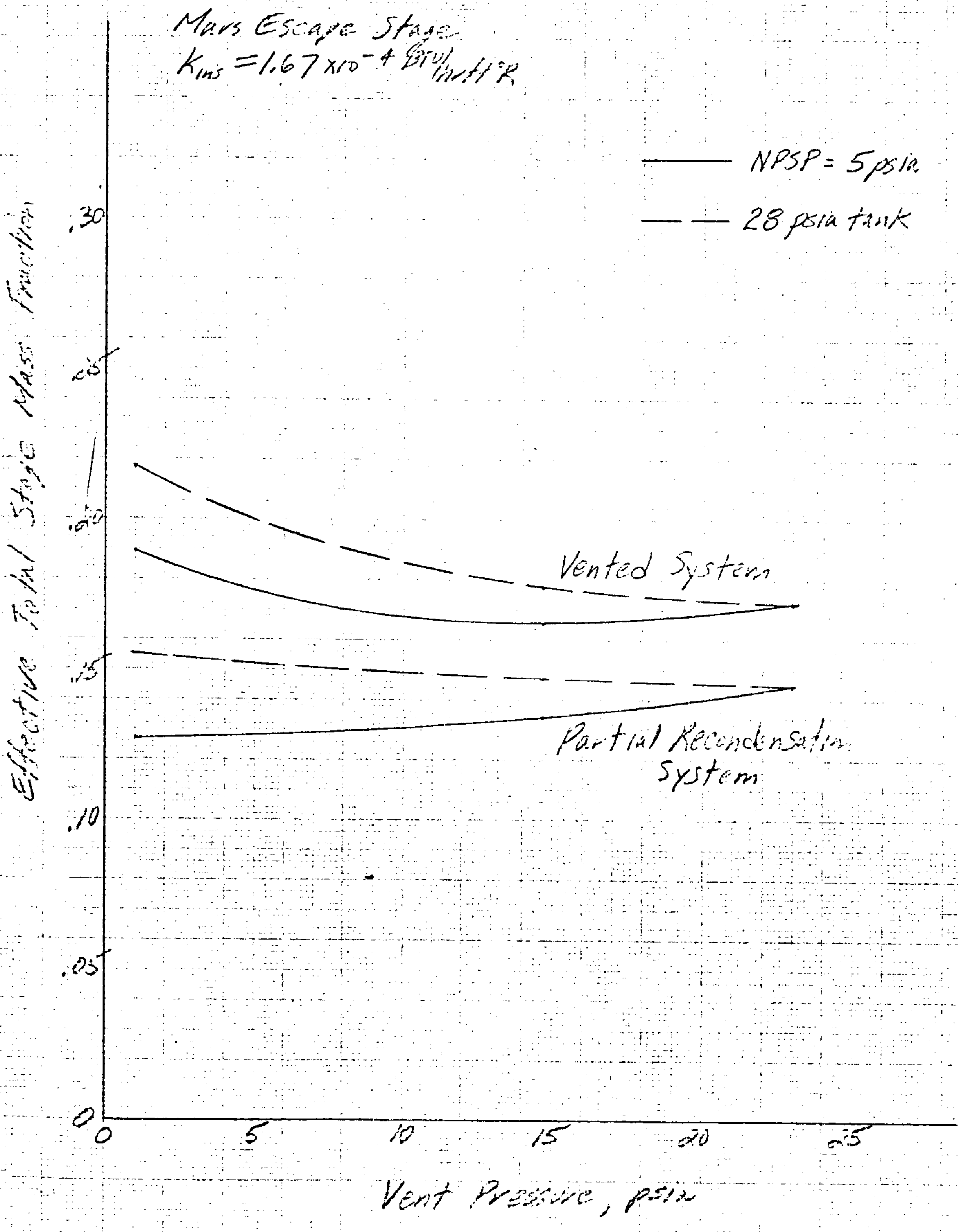
Table 3. ADIABATIC SURFACE TEMPERATURES

Earth Orbit	370°R	
Mars Transfer	242°R	
Mars Orbit	Velocity Vector Orientation	Solar Orientation
216 nm	304°R	285°R
3238 nm	232°R	170°R
9203 nm	208°R	111°R

Vent Pressure

To establish the vent pressure for vented and partial recondensation systems, a limited study of the Mars Escape Stage was undertaken with an insulation thermal conductivity of 1.67×10^{-4} $\frac{BTU}{hrft^{o}R}$. Two cases were examined. First, the tank design pressure was set at 5 psi above the vent pressure which was the independent variable. In the second case, the tank design pressure was 28 psia. From the results shown in Figure 7, a vent pressure of 14.7 psia was selected. From a design standpoint, this value seems reasonable since it is doubtful that a tank would be designed for a pressure much less than 14.7 psia.

Figure 7. Effective Total Stage Mass Fraction vs. Vent Pressure



6.0 COMPUTER PROGRAMS

Computer programs for each of the thermal management systems are in various stages of preparation. The nonvented systems program is operational and runs for the preliminary analysis have begun. The vented systems analysis and the vented system with tanking analysis is in checkout while the combination system analysis is being coded.

7.0 FUTURE WORK

Work during the next quarter will be directed toward completing the preliminary parametric analysis and interpretation of the data generated. Preparation for the final parametric analysis will also begin.

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