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AN ANALYSIS OF TRAJECTORY OPTIONS
FOR MANNED MARS MISSIONS UTILIZING
ELECTRIC PROPULSION

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ABSTRACT

A manned Mars mission whose purpose was to effect a brief exploratory/pro prospector class mission on the martian surface was investigated.

Historically, low thrust propulsion has been proposed as a means of greatly enhancing the transportation aspects of such a mission. However, past studies in this area have seldom gone beyond a top level "proof of concept" phase of analysis. Thus, although intriguing performance advantages have been cited in connection with electric propulsion, its actual feasibility from a detailed mission planning standpoint has yet to be proven.

To resolve this question, the current thesis work represents a detailed mission study which went beyond the standard "proof of concept " level. A systematic and consistent analysis of a manned mars mission that

relies on an electric propulsion vehicle was undertaken. Detailed mission models were incorporated. Differing operational strategies and various orbital nodes at both earth and mars were investigated.

The results of this effort indicate that operational aspects, previously neglected, do indeed have a dramatic impact. Both the initial on-orbit mass and round trip flight envelopes for electric propulsion vehicles were found to vary greatly, depending upon the particular option chosen at each phase. The study concluded that with this additional information one specific baseline vehicle/mission concept proved to be superior to all other low thrust concepts studied.

Finally, a pragmatic comparison of this baseline concept with other more conventional mars mission schemes was undertaken. Overall, the results of this study demonstrate that for both single and multiple mission environments, the electric propulsion concept remains a very competitive option.

To

TARK and DOC

- For your words of wisdom and insight

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LIST OF SYMBOLS AND TERMINOLOGY

<u>Symbol/Term</u>	<u>Definition</u>
AB	Aerobraking maneuver
All Up Option	Single vehicle mission option
ALPHA	Specific power plant mass for low thrust propulsion system (in lb/kW)
AREO	Aerosynchronous Mars Orbit, circular
At Planet Phase	All orbital operations occurring inside a planets sphere of influence
BOM	Beginning of mission
CT Trajectory	Constrained thrust trajectory
CHEM	Chemical propulsion (see high thrust propulsion)
Direct Heliocentric Transfer	Transfer angle about sun is less than or equal to 180 degrees. Usually referred to as a Type I trajectory
Electric Propulsion	Use of electromagnetic energy to provide low thrust levels for propulsion. Low acceleration, very long burn times
Enabling Technology	Systems without which a mission could not take place
Endogenous	Acceleration of matter by using the inherent internal energy of chemical combination (see high thrust propulsion)

<u>Symbol/Term</u>	<u>Definition</u>
Enhancing Technology	Systems which improve mission performance over and above enabling status
EOM	End of mission
ERV	Earth return vehicle
Escape Velocity	Minimum velocity necessary to recede an infinite distance from a gravitational body
Exogenous	Acceleration of matter with externally provided energy (see electric propulsion)
Flyby Maneuver	Close approach to a planetary body without a stop over being performed
48 hour ellipse	Highly eccentric ellipse about Mars with a 48 hour period; (270 NM (alt) perigee by 30,778 NM (alt) apogee)
GEO	Geosynchronous Earth Orbit, circular
Gravity Well	Fictitious "pit" or "well" representing energy required to escape from a celestial body's gravitational field
Heliocentric Flight Time	Time to travel from one planetary sphere of influence to another
High thrust propulsion	Thrust produced by the ejection of large amounts of material via a thermodynamics process resulting generally in high acceleration and short burn times

<u>Symbol/Term</u>	<u>Definition</u>
Impulsive maneuver	Use of high thrust to affect orbital parameters (see high thrust propulsion)
Indirect Heliocentric Transfer	Transfer angle about Sun is greater than 180 degrees. Usually referred to as a Type II trajectory
Inbound Trajectory	Transfer originating at Mars and whose destination is earth
Ion Drive	Use of electrostatic energy to accelerate ions to produce thrust (see electric propulsion)
ISP	Specific impulse (in seconds). A measure of propulsion system efficiency
Lagrange Point (L1)	One of the five gravitational equilibrium positions in a three body system
LEO	Low Earth Orbit, circular
LMO	Low Mars Orbit, circular
Low Thrust Propulsion	See (Electric propulsion)
LOX/H2	Liquid oxygen and hydrogen. Used as chemical propulsion fuel
MEM	Mars Excursion Module
$M_{P/L}$	Payload mass
MR	Mass ratio. Ratio of initial mass to final mass
M_{STR}	Structural mass

<u>Symbol/Term</u>	<u>Definition</u>
NEP	Nuclear electric propulsion
OTV	Orbital transfer vehicle
Outbound Trajectory	Transfer originating at Earth with Mars as destination
Planetocentric	Reference frame centered about a planetary body
Powered Flight Trajectory	No coast phases, produced by variable thrust trajectory profile
Rendezvous	A close approach of a celestial body such that a stop over is performed
Rocket Equation	Fundamental vehicle sizing equation for rockets. Relates propulsion system directly to mission energy requirements
SEP	Solar electric propulsion
Shaped Ellipse	A series of powered flight and coasting trajectory segments. Produced by constrained thrust profile
SOI	Sphere of influence, "fictitious" sphere about celestial body at or outside of which body's gravitation influence is assumed to be zero
Spiral Trajectory	Typical transfer trajectory profile low thrust systems perform while deep inside a celestial bodies gravity well

<u>Symbol/Term</u>	<u>Definition</u>
Split option	A two vehicle mission option
Swing by maneuver	Similar to flyby; however, through the intentional use of the flyby planets gravitational field the original spacecraft trajectory is altered
TOF	Time of flight
T/W	Thrust to weight ratio
V-Infinity (V_{∞})	Relative velocity of vehicle at entry into or departure from the sphere of influence of a celestial body
VT Trajectory	Variable thrust trajectory, ISP level optimized throughout flight

CHAPTER 1.0 INTRODUCTION

1.1 Comments

The current climate of interest within NASA concerning "Bold new initiatives" as outlined in the Ride Report (Reference 1), "Leadership and America's Future in Space" and in the "National Commission on Space Report" (Reference 2), "Pioneering the Space Frontier" has renewed interest in the study of various space mission scenarios including manned exploration of the planet Mars. Many possible scenarios have been put forth concerning Mars exploration using a variety of assumptions and technologies. Of these, one which needs serious re-examination, is that of electric propulsion.

Past studies indicate that an interplanetary vehicle's initial on-orbit mass can be greatly reduced when an electric acceleration device is used as the vehicle's primary propulsion system. Because initial vehicle mass is an indicator of an interplanetary missions overall cost and logistical complexity, such a reduction would prove to be very beneficial.

However, in spite of these results, electric propulsion studies have seldom gone beyond a top level "proof of concept" phase. The majority of the literature tends to consider only very idealized mission scenarios. In addition, many operational aspects of mission planning have been neglected. Thus, although intriguing performance advantages have been cited in connection with electric propulsion, its actual feasibility from a detailed mission planning stand point has yet to be proven.

The current thesis work represents a detailed mission study undertaken to resolve this feasibility issue. The primary contribution of this study was in the quantification of both the logistical and technological parameters affecting such a mission. In addition to the heliocentric transfer issues, detailed trades were conducted in regard to the on-orbit operations at both Earth and Mars. In many past studies, these on-orbit effects were often neglected.

In this regard, the effect that augmenting the low thrust propulsion vehicle with chemical "strap-on" stages for the at-planet departure or arrival maneuvers was investigated. Transportation requirements for the crew while at Earth and Mars was also incorporated into the study. The impact that aerobraking technology had on vehicle design was addressed. Finally, another key parameter studied was the effect that total trip time had on vehicle performance and design.

One conclusion of the study is that a propulsion system which requires a multi-megawatt power source and a thruster assembly that can generate specific impulse levels on the order of 3,000 seconds is necessary, if viable manned low thrust vehicles are to be used for Mars exploration. Research indicates that the current status of these propulsion related technologies meet or verge on the above cited performance thresholds. Thus, the thesis work demonstrates that the use of electric propulsion to conduct manned Mars exploration could be realized in a relatively near-term environment.

More specifically, it was found that a hybrid vehicle primarily relying on an electric propulsion system in combination with a chemical

earth departure stage could achieve round trip flight times as low as 1 year with an initial on orbit mass of approximately 2,000,000 pounds. On-orbit operations were found to play a significant role with overall mass and time of flight requirements varying greatly depending on the option chosen. The study concluded that one specific baseline vehicle/mission concept proved to be superior to all other low thrust concepts studied [see Table 3.3 and Figures 3.3 (A-C)].

Finally, a pragmatic comparison was made of this baseline concept with other more conventional Mars mission schemes. The results of this study demonstrate that for both single and multiple mission environments the electric propulsion concept remains a very competitive option [see Figures 3.4.2(A-B) and 3.4.3 (A-C)].

1.2 Thesis Organization

The paper which follows will discuss the above cited results in detail. The text is divided into two major sections.

The first, Chapter 2, will discuss the analysis process itself. Major mission phases that were studied will be highlighted as well as relevant caveats that were included. In addition, a discussion of the computer program used to generate a vast amount of the data required to assess various mission options has been included.

In the second half of the paper, Chapters 3 and 4, the results of the study are presented. The results are discussed in three major subsections. The first is related to early parametric findings. The second discusses data derived from a detailed mission analysis under

taken after the parametric results had been analyzed. Finally, the third section presents mass and time of flight data related to low thrust vehicle performance as compared to other mission scenarios which rely on vastly different primary propulsion systems.

The appendix section contains four appendices. The first three appendices are related to secondary calculations required to conduct "mini" trade studies within the thesis and are appropriately referenced in the text. Appendix 4 is a general discussion on electric propulsion. A reader, unfamiliar with the concept of electric propulsion and/or its potential usefulness to interplanetary vehicle propulsion, may find it beneficial to read Appendix 4 before proceeding to the body of the text.

CHAPTER 2.0 ANALYSIS DESCRIPTION

2.1 General Comments

The approach used in the conduct of this study was similar to the iterative approach suggested by Brewer (Reference 5). After initial feasibility studies were completed, more and more detailed analyses were undertaken until final concepts were obtained.

Thus, the study began with a parametric analysis of an array of low thrust manned mars vehicle/mission concepts so that a general range of power levels, spacecraft weights, time of flight envelopes, and other vital mission parameters could be found. Once a particular mission was identified as being feasible, a more detailed analysis was undertaken. This detailed analysis encompassed more sophisticated simulation of interplanetary flight performance as well as investigation of "at planet" transportation requirements. A matrix of various low thrust vehicle/mission concepts resulted. This data base provided the detailed mission data necessary to evaluate low thrust propulsion's true feasibility in relation to a manned mars mission.

2.2 Basic Assumptions and Constraints

The following parameters were fixed early in the study in order to focus on the effect that an electric propulsion system would have on a manned mars initiative.

2.2.1 Mission/Vehicle Constraints

The mission and vehicle constraints consisted of:

- o The missions studied were to be exploratory in nature (i.e., similar to an "Apollo type mission" in that crews would land

on Martian surface, conduct brief exploration, etc., and return to the Earth)

- o A single vehicle with a Mars Excursion Module (MEM) was chosen for study. Common payloads were manifested for each vehicle in order to standardize nonpropulsive related equipment. This included common habitation modules, science probes, medical equipment, etc. Table 2.2.1A lists these elements as well as their associated mass.
- o A stay time of 60 days on the Martian surface was chosen for all missions studied.
- o An artificial gravity device on board the main ship capable of providing 1 g was assumed for the entire transfer.
- o A crew of six was chosen to serve as the mission team
- o A piloted round trip flight time envelope of between 1 and 2 years was considered.

Table 2.2.1 summarizes the above mission/vehicle constraints.

2.2.2 Infrastructure Assumptions

In order to define the "environment" in which the vehicles were to operate, three infrastructure assumptions were made. These assumptions were incorporated in order to ensure commonality of space facilities available to each option.

The first infrastructure element assumed, was an assembly capability for the Mars vehicles in low earth orbit. Whether this facility is built specifically for a manned Mars effort or is the result of an evolutionary space station concept was left open. The second assumption was that of a launch vehicle with a payload delivery capability in the range of 200,000 lb LEO. This launcher would be used to rendezvous Mars vehicle components with the assembly facility. The third and last infrastructure assumption was that of an orbital

transfer vehicle(s) that could transfer crews, material, and equipment between orbits inside of the Earth's sphere of influence (i.e., from LEO to the Earth's sphere of influence or anywhere in between). No specific propulsion system for the OTV network was considered, although Reference 2 indicates that a combination of OTV's utilizing chemical, low thrust and or aerobraking techniques would probably provide the most flexible system. Table 2.2.2 summarizes the infrastructure assumptions.

2.3 Propulsion System Issues

With the establishment of the overall mission goals, common payload elements, total mission time range and infrastructure elements assumed in earth orbit, a propulsion system analysis was conducted.

The propulsion system is responsible for the transportation of the payload (crew and mission equipment) from earth departure or beginning of mission (BOM) to earth return or end of mission (EOM).

Low thrust propulsion was studied as the primary propulsion system for heliocentric mission phases. Trade studies were performed on the following parameters in order to determine the associated performance requirements for such a system.

The first trade concerned an assessment of the electric thruster/acceleration system. In this regard, systems that employ electromagnetic forces to accelerate matter (i.e., electric thrusters) are divided into four major categories (Reference 23). These categories are:

- o Ion Thruster Systems
- o Mass Driver Systems

- o Magneto Plasma Dynamic Thrusters
- o The Arc Jet Thrusting Systems.

Ion thrusters rely primarily on electrostatic forces to accelerate positive ions of gas which are then ejected as thrust. Mass drivers use electromagnetic forces to accelerate "pellets" of material along a track or rail. The MPD thruster again uses electromagnetic forces but instead of pellets it accelerates plasmas to high ejection velocities. Finally, the arc jet uses electric energy to disassociate gases (e.g. hydrogen) which absorbs energy during this process. The disassociated gas is allowed to recombine; converting the energy of disassociation into kinetic energy and producing a super heated gas which expands out the nozzle. In relation to these four thrusting systems, minimum operational life time and specific impulse levels were investigated.

The second trade concerned the electric power generation system. Important performance criteria in this area are related to the minimum total power level required and the associated power system mass.

The remaining propulsion issues dealt with the "at planet phases" of the mars mission (Note: this area is addressed in more detail in the following section). For the planetocentric phases, both the low thrust system sized for the heliocentric phase and impulsive braking schemes (either active or passive) were investigated. For chemical systems, a specific impulse value of 480 seconds and a structural mass fraction of 0.1 was assumed. Aerobraking was assessed using a standard empirical relationship discussed in Appendix 2.

2.4 Operational Trajectory Issues

For planetocentric operations, a matrix was established of orbits and propulsion system combinations to be used by potential mars vehicles. In this way, the impact that these orbits and propulsion systems had upon overall vehicle mass and performance could be accessed.

Specifically, this effort was divided into the following areas:

- o During earth departure: low thrust, chemical, and hybrid departure options (combinations of low and high thrust departure stages) were investigated.
- o At Mars five staging orbits were studied. These included; a low circular orbit (LMO), an orbit at Mars similar to geosynchronous orbit at Earth called areosynchronous orbit (AREO), a 48-hour high eccentricity ellipse with a 500-km in altitude perigee (48HrELL), an orbit established at the martian sphere of influence (see Appendix 1 for details) known as the "SOI maneuver" and the establishment of a staging point at the first Lagrange point of mars (Llmars). Again, for each orbital node low thrust, chemical, and aerobrake technology was investigated for all arrival and departure operations connected with the main interplanetary vehicle. In addition, a unique Mars excursion module was sized for each mars orbit studied. In this way each interplanetary vehicle carried a properly sized surface landing/return vehicle to the operational orbit assigned. Thus, the total "at mars" transportation penalty connected to a particular orbit could be addressed. Appendix 2 outlines the calculational procedure used for MEM mass sizing.
- o During earth return chemical, low thrust, aerobraking, and hybrid options were studied. These hybrid options called for the interplanetary vehicle to either fly-by the earth or return to either LEO or GEO via low thrust while the crew returned in a separate Earth Return Vehicle (ERV). Appendix 2 outlines the sizing procedure used to calculate the ERV mass penalty.

2.5 Computer Code Description

The following section represents a description of computer algorithm used to simulate the matrix of Mars mission operations

studied in this report. The program was found to be useful in the quantification of detailed mission performance parameters such as those outlined in Sections 2.1 through 2.4.

The computer code description section is divided into three parts. In section 2.5.1 a general qualitative discussion of the computer program is presented along with a description of how the code was used in regard to the present study. Section 2.5.2 presents the mathematical algorithm upon which the computer code is based. The algorithm is derived from first principles with full mathematical rigor. Finally, in section 2.5.3, data relating to the codes accuracy is presented. Validation of the computer algorithm as a reliable mission analysis tool is cited. Also presented is a brief history of the codes' past validation as an accurate predictive analysis tool.

2.5.1 Comments

The low thrust mars mission study was made practical through the use of the "QUICKTOP III" low thrust trajectory and mass optimization simulation code (Reference 14). "QUICKTOP" stands for Quick Trajectory and Mass Optimization Computer Program. Originally written for NASA by Boeing under contract with the AMES Research Center in 1973, the code was designed to determine system and performance requirements of electrically propelled spacecraft. In addition, to optimizing low thrust vehicle performance in heliocentric space the code can also simulate high or low thrust vehicle departure or capture during "at planet" operations. Thus, the code relies on a method whose organizational philosophy is similar in spirit to the standard Patched Conic Approach (see Reference 22).

QUICKTOP represents a master program. The core of QUICKTOP (i.e., the actual algorithm used to calculate heliocentric trajectories) is a program called CHEBYTOP III (CHEBYCHEV Trajectory Optimization Program Reference 15). Also created by Boeing under a NASA-AMES contract, CHEBYTOP III is a numerical algorithm that uses the CHEBYCHEV optimization method to calculate three dimensional low thrust heliocentric trajectories (method described with more mathematical rigor Section 2.5.2, see below). This method avoids the numerical integration and usual variational calculus solution, which suffers from very long execution times, by employing a polynomial representation scheme of the cost function (i.e., a payoff parameter representing both trajectory and propulsion parameters). When optimized, this function determines the trajectory and propulsion requirements for the transfer. It is specifically formulated to solve for interplanetary trajectories from sphere of influence (departure) to sphere of influence (arrival). By approximating the trajectory and associated payoff parameter with a series of CHEBYCHEV polynomials, the method reduces the standard variational problem to one of ordinary calculus. The code itself solves interplanetary trajectories with the planets assumed massless. Planetary positions and velocities are computed from orbital elements stored within the code. QUICKTOP calculates the high or low thrust trajectory inside the planetary spheres of influences using straight forward analytical expressions (see Reference 14 for listing). These are then matched to the heliocentric trajectory.

CHEBYTOP will compute both constrained thrust and variable thrust trajectories (see Appendix 4 for definitions). The variable thrust solution represents an exact solution to the mathematical problem posed while the constrained thrust solution is an approximate answer. The fundamental assumption in the constrained thrust derivation is that the constrained thrust trajectory may be approximated by the variable thrust solution. The constrained thrust solution is thus modeled as a combination of the variable thrust solution plus a residual term defined as the difference between the two solutions (see References 12 and 15 for rigorous mathematical treatment). In order to effectively calculate this residual term and hence the constrained thrust solution, two caveats are necessary: (1) the difference in acceleration levels between the variable and the constrained thrust solutions must be "small" and, (2) the power profiles of the two must be "close". Reference 15 states that comparisons to data obtained from other CT trajectory prediction schemes has shown that this approximation is sufficiently accurate to allow reliable mission analysis. The constrained thrust option was the mission mode actually used in this thesis. It represents a thrusting profile that can be more realistically maintained by low thrust engines without serious loss in thruster operating life as well as one that avoids operational complexity by having the thruster complex either in an all on or off mode.

The CHEBYTOP procedure is not directly accessed by the user. As stated before, QUICKTOP represents the master program and uses CHEBYTOP as an internal calculation algorithm. User interface with QUICKTOP is

through a namelist input file. In this way desired mission parameters are either fixed or allowed to be optimized. This namelist is then assigned to QUICKTOP which performs the trajectory and mission sizing automatically. A multitude of parameters or possible mission modes may be simulated and or optimized. The thesis work drew heavily on this aspect of the code in order to obtain the detailed mission data necessary for a rigorous evaluation of a low thrust manned mars mission.

Finally, it is important to understand one feature of the codes operation that did complicate its usefulness. Essentially, QUICKTOP is strictly a "one way" transfer algorithm. Thus the code, as written, does not simulate roundtrip missions automatically. However, by manually breaking up the round trip trajectory problem into two one way trajectories; the first going from Earth to Mars and the second going back while keeping track of payloads, arrival and departure dates, the proper trajectory sequence itself, and other bookkeeping aspects, the code could be manipulated into calculating round trip mission modes. This procedure, in essence, only increased the actual time needed to conduct the study.

2.5.2 CHEBYCHEV Optimization Method (Mathematical Development)

The following section outlines the mathematical foundation upon which the CHEBYTOP III computer algorithm is based. Additionally, Reference 11 presents this same derivation in a more rigorous fashion and it is from this source that the following development is modeled.

The basic idea used in the method is the approximation of the state time history by a polynomial. Differentiation and integration

are performed in closed form, and the optimization is reduced to one of ordinary calculus wherein a function of a finite number of parameters is to be minimized. The independent parameters chosen are not simply related to the coefficients of the polynomial as in usual applications (i.e., classical Rayleigh-Ritz method), rather they turn out to be interpolated values of the state vector itself. Newton's method is employed to achieve zero gradient, the necessary condition for an optimum.

DEVELOPMENT:

A large portion of any interplanetary trajectory can be characterized by the following differential equation:

$$\|\ddot{x}(t) + [ux(t)/\|x(t)\|^{**3}]\|^{**2} - \|a(t)\|^{**2} \quad \text{eqn. (1)}$$

where:

$x(t)$ = position vector time history

$a(t)$ = the applied acceleration vector time history

u = the gravitational constant of the sun.

For most problems involving constant power the problem is that of minimizing:

$$J = \int_0^T \|a(t)\|^{**2} dt \quad \text{eqn. (2)}$$

subject to various conditions on $a(t)$, T , $x(0)$, $\dot{x}(0)$, $x(T)$, $\dot{x}(T)$.

A derivation and discussion of eqn.(2) may be found in Reference 40.

Substituting (1) into (2) we have:

$$J = \int_0^T \|\ddot{x}(t) + ux(t)/\|x(t)\|^{**3}\|^{**2} dt. \quad \text{eqn. (3)}$$

Normalizing the independent variable t by setting $s = t / T$.

Then,

$$J = T \int_0^1 \left[\ddot{x}(s)^2 + u^2 x(s)^2 / \|x(s)\|^3 \right] ds \quad \text{eqn. (4)}$$

where the derivative dots refer to differentiation with respect to s .

Note that $x(0)$, $\dot{x}(0)$, $x(1)$, $\dot{x}(1)$ still satisfy the same conditions as before, modified by the normalization of t .

Finally, for convenience, write $J = 2P/T^3$. We can now express the cost function as:

$$P = 1/2 \int_0^1 \left[\ddot{x}(s)^2 + u^2 x(s)^2 / \|x(s)\|^3 \right] ds \quad \text{eqn. (5)}$$

and note that minimizing P implies the minimum for J .

The quadratic form $\left[\ddot{x}(s) + [u^2 x(s) / \|x(s)\|^3] \right]^2$ is sufficiently simple to make feasible the approximation of $x(s)$ by a polynomial.

Let $x(m,s)$ be the m th cartesian coordinate of $x(s)$, and set:

$$r(s) = \left[\sum_{m=1}^{nd} x(m,s)^2 \right]^{1/2}$$

$$y(m,s) = u^2 x(m,s) / r(s)^3$$

$$w(s) = u^2 / r(s)^2$$

Here nd is the number of dimensions of the position vector. Then from (5):

$$P = 1/2 \sum_{m=1}^{nd} \int_0^1 \ddot{x}(m,s)^2 ds + \sum_{m=1}^{nd} \int_0^1 \ddot{x}(m,s) y(m,s) ds + 1/2 \int_0^1 w(s)^2 ds \quad \text{eqn. (6)}$$

Let $X(m)$, $Y(m)$, and W be n vectors with $X(m)v = x(m,sv)$, $Y(m)v = y(m,sv)$

and $Wv = w(sv)$. Note that $X(m)$ and $Y(m)$ are sets of n vectors depending on the index m . The sv are Chebychev points defined on $[0,1]$ where:

$$sv = 1/2(1 - \cos(\pi(v-1)/(n-1))), \quad v = 1, 2, \dots, n. \quad \text{eqn. (7)}$$

Let $\hat{x}(m,s)$, $\hat{y}(m,s)$ and $\hat{w}(s)$ be the polynomials which interpolate $x(m,s)$, $y(m,s)$ and $w(s)$ at sv . Then by drawing on the properties of Chebychev polynomial it can be shown that:

$$\begin{aligned} \hat{x}(m,s) &= T(s)^T BAX(m) \\ \hat{y}(m,s) &= T(s)^T AY(m) \\ \hat{w}(s) &= T(s)^T AW \end{aligned} \quad \text{eqn. (8)}$$

where:

$T(s)$ is a n vector with the first n Chebychev polynomials as elements.

A is some n by n constant matrix satisfying the equation $\phi = AF$ where ϕ is a column matrix of ϕ^n elements and F is a column matrix of F^n elements whose individual terms form n equations in n unknowns for any generalized function $f(sv)$ which we wish to represent using Chebychev polynomials. When one knows A , one may express any function on an interval from 0 to 1 by $\hat{f}(s) = T(s)^T AF$.

B is an n by n upper triangular matrix which satisfies the property that: $\hat{\hat{T}}(s)^T = T(s)^T B$ which we can see implies that the second derivative of $\hat{f}(s)$ is simply given by; $\hat{\hat{f}}(s) = T(s)^T BAF$.

Note: When using Chebychev polynomials it is in general easy to calculate A and B and update them recursively. This is an aid in computational solution.

Note: The superscript τ denotes transposition.

Returning to the mathematical development, if we now substitute (8) into (6) we find:

$$P = 1/2 \sum_{m=1}^{nd} X(m)^{\tau} DX(m) + \sum_{m=1}^{nd} X(m)^{\tau} EY(m) + 1/2 W^{\tau} F W \quad \text{eqn. (9)}$$

where:

$$D = A^{\tau} B^{\tau} C B A$$

$$E = A^{\tau} B^{\tau} C A$$

$$F = A^{\tau} C A$$

with C being defined as an $n \times n$ matrix:

$$C = \int_0^1 T(s) T(s)^{\tau} ds$$

Note that the matrices D , E , F may be precomputed and stored.

This expression for P involves only the interpolated values of x , y and w . The differentiation, integration and computation of coefficients of \hat{x} , \hat{y} and \hat{w} have all been accounted for in D , E and F .

Also note that the vectors $Y(m)$, $m = 1, \dots, nd$, and W are explicit functions of the vectors $X(i)$, $i = 1, \dots, nd$. Thus, we may consider P to be a function of the parameters $X(i)v$, $i = 1, \dots, nd$; $v = 1, 2, \dots, n$. In order to minimize P it is necessary to know the partials derivatives [which can now be calculated from (9)] of P with respect to these parameters.

The iterative algorithm used to optimize P is based on Newton's method and can be stated as follows: "Let P be a function of any unconstrained vector Z . Let Z_0 be some point at which $P(Z)$ has a local

non-singular minimum. Let Z_i , $i = 1, 2, \dots$ be the i th iterative determination of Z_0 . If Z_1 is sufficiently close to Z_0 and Z_{i+1} satisfies

$$[\delta^2 P(Z_i) / \delta Z^2](Z_{i+1} - Z_i) = - \delta P(Z_i) / \delta Z \quad \text{eqn. (10)}$$

then Z_i will converge to Z_0 ." At each i , equation (10) may be solved as a linear set of equations in the unknown vector $Z_{i+1} - Z_i$, yielding Z_{i+1} .

Thus, using this method to minimize P , results in an optimum trajectory determined for the constant power problem posed.

2.5.3 Code Validation Section

At the present time, the QUICKTOP III computer algorithm has attained a high level of refinement and represents a mature analysis code. Over the 15 year history of the code, the basic mathematics behind its original formulation has remained virtually unchanged. The code has been validated for predictive accuracy by many authors (see References 11, 12, 14, 15).

In general, the code has been found to model low thrust missions to a high degree of accuracy. Figures 2.5.3A, 2.5.3B, and 2.5.3C taken from Reference 11 have been provided as a demonstration of the codes predictive power. In Figure 2.5.3A a comparison of optimum values of J (the integral of the thrust acceleration squared), as predicted by QUICKTOP, with data obtained from Reference 41, is presented. In Figure 2.5.3B similar data for three dimensional optimum trajectories is displayed while Figure 2.5.3C illustrates a parametric treatment of hyperbolic excess velocities. Comparison data for Figures 2.5.3B and 2.5.3C were taken from Reference 42.

CHAPTER 3.0 RESULTS AND DISCUSSIONS

3.1 Parametric Heliocentric Performance Study

Based on the mission requirements, payload demands and roundtrip time of flight envelopes imposed in Chapter 2, the following general performance levels for the low thrust vehicle were deemed necessary. These results were obtained from heliocentric data generated by the QUICKTOP computer code and through a parametric examination of the electric propulsion data base available from the literature. The results contained in section 3.1 are also displayed in Table 3.1 for quick reference.

3.1.1 Thruster Type and Specific Impulse

Upon investigating the current literature, it became apparent that of the four currently available or proposed thruster systems (see section 2.3 for details), only the ion thruster using argon as a working gas represents a technologically mature acceleration system. This ion thruster offers potential specific impulse ranges from 2,000 to 10,000 seconds (max of 5,000 seconds demonstrated (see References 7 and 23)). Currently, no other system offers specific impulses beyond 2,000 seconds and many have inherently low operating efficiencies (References 7 and 23). In addition, the MPD, Arc Jet, and Mass Driver are still in the prototype stage while the ion engine has been flight tested (e.g., the SERT I and SERT II flight demonstration projects conducted by NASA in the 1960s and early 1970s). The ion thruster and has undergone thousands of hours of laboratory testing and hardware

development (Reference 23). Ion thrusters also offer reliable lifetime and restart capability (Reference 25). Thus, the ion thruster system was chosen as a basis for these studies.

From a performance standpoint, Reference 13 and sample runs of QUICKTOP demonstrated that interplanetary vehicle operation within an ISP range of 2,000 to 4,000 seconds provided higher thrust levels (i.e., acceleration) than when operating at ISP's in the range of 5,000 to 10,000 seconds. Selecting the higher acceleration level reduces reduced payload efficiency, but is required to maintain the time of flight envelope within that specified for the interplanetary transfer phase. Based on this analysis, a specific impulse level of 3,000 seconds was chosen for the ion thruster system.

3.1.2 Electric Power Generation System

For the investigation at hand, it became apparent from the literature and from sample runs of the QUICKTOP code that a multi-megawatt power supply would be necessary. At the present time, only two power generation systems exist that could potentially fulfill this requirement. The first, utilizes solar cell technology. The second, nuclear reactor systems.

Solar cells are of course the more conventional technology, References 5 and 25 point out that for power levels of 1 megawatt or greater even a highly efficient solar power system would be very unwieldy. For example, based on projected (I.O.C.) U.S. Space Station Technology, one would need a 389,020 sq ft (9 acre) array system to produce 1 megawatt of power in low earth orbit. Furthermore, in the case of a Mars expedition, increased distance from the Sun while at

Mars would increase the size of such a system yet again by more than a factor of two. Thus, solar power was not viewed as a viable option.

With this, nuclear power generation systems were then considered. After a careful review of such systems, it was seen that they could provide a viable way to achieve the necessary high power levels at an associated mass that was not prohibitive. Since both the power level (P) (measured in kW) and specific weight (α = alpha) of the power subsystem (measured in lb/kW) are of vital importance a parametric review of these parameters was performed. By taking into account time of flight and payload trends as well as projected capabilities in nuclear reactor systems to be used in space (see Reference 16), both the power level and reactor mass ranges were determined for the present study.

In relation to total power output, a maximum power level of 10 MW with a normal operating range of 5 MW was found to meet the minimum requirements for the electric propulsion mission. In regard to reactor mass, three power generation system specific mass (alpha) ranges were seen as producing viable propulsion systems.

These alpha ranges were 22 lb/kW (projected late 1960s technology), 11 lb/kW (projected current technology), and 2.2 lb/kW (projected future technology). The technology time frames were suggested by Reference 16. Quantifying the variation that this parameter had upon vehicle performance was seen as important. This is because the parameter alpha provides an indication of how "lightweight" such power generation systems have to be in order to either become effective or provide substantial gains over older technology systems.

Thus, all three ranges were chosen and incorporated into the present study.

3.2 Detailed Mission Simulation Study

Once general viable performance values for the low thrust interplanetary vehicle's propulsion system were established, a detailed mission study was undertaken (as outlined in Chapter 2). What follows are the results of this study. Entire missions were simulated (using the QUICKTOP computer program) from earth departure (BOM) to earth return (EOM). Since all mission phases were dependent and simultaneously considered by QUICKTOP, the order in which the results are discussed is somewhat arbitrary. However, in order to access major areas of interest, the results are subdivided by major mission phases (i.e., earth return, earth departure, the "at Mars" segment and the heliocentric flight phase).

3.2.1 Earth Return

The return problem quickly converged toward one particular mode of operation whose benefits were clearly superior to all others studied.

Classically, three options have been suggested as a means of returning manned low thrust vehicles to the vicinity of the Earth.

They are:

- o A low thrust spiral descent to a low earth orbit
- o An impulsive (chemical or aerobrake) maneuver to LEO
- o Flyby maneuver at the Earth is performed by the interplanetary vehicle while the crew descends in a separate smaller, crew return vehicle which is impulsively braked at LEO via a chemical or aerobrake stage.

It was found that Option 1 yielded very low propellant requirements as opposed to chemical stages (reductions on the order of a factor of five were typically found). This is because the more efficient low thrust system is used. However, the low thrust system demanded very long transfer times from the Earth's sphere of influence to low earth orbit (on the order of hundreds of days). Such a return mode was used in past studies (e.g., Reference 8) that minimize vehicle and propellant mass at all costs. Such a focus places a low priority upon the minimization of the time that crews must stay aboard the vehicle. Option 2 which used impulsive braking schemes (either active or passive) returned the crew from the SOI very quickly (hours to a few days). However, as alluded to earlier, it was found that if chemical braking (active systems) were used, propellant mass increased dramatically. (Passive) Aerobrake systems were found to perform the fast transfer but at a substantially reduced mass penalty since they use the Earth's atmosphere and not propellant to slow the ship. However, because of the nuclear reactor on board the returning vehicle, aerobraking is not only risky but presents dangerous failure modes.

Option 3 essentially "throws away" the entire electric propulsion vehicle. Such a mode of operation allows for both quick crew return and the elimination of the braking problem for the low thrust vehicle. However, this approach is really one of solving a problem by avoiding it. The option gives up any possible benefit that low thrust vehicles may offer by way of its inherent high reusability of equipment and vehicle components.

When conducting the first cursory sizing cycles of the study, the QUICKTOP program indicated that the propulsion system outlined in Section 3.1 was capable of performing round trip missions in the 1 to 2 year envelope with a great deal of flexibility and with optimized Earth return velocities that were very low (less than 1,623 ft/s at SOI). Thus, Option 3 seems extreme since the loss of the low thrust vehicle would not be caused by trajectory inflexibility (i.e., an option forced on the scenario by fast hyperbolic entry velocities in order to meet short time of flight demands). Option 3 was, therefore, viewed as an option to be avoided if some type of reasonable compromise between the other options could be found. Likewise, because of safety concerns, the aerobraking of the low thrust ship was seen as not viable and placed at an even a lower priority than the above mentioned flyby option. This left Options 1 and 2 (active propulsion aspect only) available for study. As discussed earlier, each contain tempting capabilities as well as serious drawbacks. Although neither option seemed acceptable, a hybrid of the two was found to be acceptable. By borrowing the concept of a crew return vehicle from Option 3, the best aspects of Options 1 and 2 were utilized while avoiding their inherent negative points. This derivative approach will be referred to as the hybrid descent option.

The hybrid descent option, utilizes a low thrust spiral capture for the electric vehicle while the crew return to LEO impulsively in a small earth return vehicle. Thus, the more massive interplanetary vehicle is returned to a specified earth orbit efficiently and with no

time of flight constraint (since the crew is no longer on board). At the same time, the crew returns quickly to low earth orbit. The details of this option are discussed below. However, Figure 3.2.1A provides a bottom line numerical comparison of the options mentioned above against this hybrid option.

As mentioned above, the hybrid descent option is really two separate maneuvers. The exact orbits and requirements imposed on the crew return vehicle as well as the low thrust vehicle require further discussion. As mentioned above, at the end of the mission the crew would return to low earth orbit. An analysis of the final braking maneuver upon reaching LEO was undertaken. Both solid and chemical stages were considered. Figure 3.2.1B outlines the results of this work. After reviewing the data, it was found that an aerobraking system could potentially save mass and provide a viable return mode. However, the all-propulsive option, although more massive, was not too large and was well within the overall payload limitations of the round trip mission profile. Because the exact mass of an aerobrake, in general, is a subject of great debate, and because aerobraking in itself represents an advanced aerodynamic concept yet to be demonstrated or refined, it was decided that the all chemical propulsive option for the crews descent would be utilized.

The second maneuver alluded to was that of the electric vehicle spiraling to a yet to be discussed earth orbit. Using the QUICKTOP program, a procedure was utilized where by at the Earth's sphere of influence, the mass of the ERV was dropped from the total electric

propulsion vehicle mass. This simulated the ERV separating from the low thrust ship before returning to low earth orbit. At this point, the low thrust vehicle spiraled down into the Earth's gravity well. Two operational orbits were investigated for this descent (geostationary orbit and low earth orbit). Reviewing Figure 3.2.1A shows the time of flight and propellant requirements to spiral to each of these orbital nodes. The results demonstrate that spiraling the ship to LEO requires a great deal of fuel and time as compared to a similar flight to GEO. In a single mission environment, the time of descent is no longer a critical issue since the crew is not on board. However, the rather large propellant savings from a GEO return as opposed to a LEO return is an important point. A GEO return not only yields a favorable mass savings, but also represents a reasonable storage orbit for the low thrust vehicle until it is again desired for use. This "reuse issue" is discussed in more detail in Section 3.4 and in Appendix 3. However, the essence of the discussion in these two sections is that GEO provides a convenient and flexible node which could allow many possible options in a reuse environment. Thus, for the reasons given above, a low thrust return spiral to GEO was utilized within the context of the hybrid descent option.

3.2.2 Earth Departure

Departure from Earth orbit, like arrival, represented a phase which quickly iterated to a solution in the analysis process.

Because of the infrastructure assumptions two major operational orbits were seen as potentially beneficial; low earth orbit and

geosynchronous orbit. Using QUICKTOP trends detailing departure requirements were found. These results are similar to those found for the earth arrival problem at the end of mission phase (EOM). Recall that the earth arrival problem was solved by relying on a hybrid arrival option (i.e., separating the crew from the interplanetary ship at the SOI with their quick return to LEO while a low thrust spiral was used to return the return the interplanetary ship to GEO). Similarly, after examining various departure options (all high thrust, all low thrust and two hybrid departure options using differing combinations of chemical and electrical propulsion: see Figure 3.2.2), it was apparent that a hybrid option utilizing both electric and chemical systems would provide the most viable departure option.

In this scenario, an interplanetary vehicle once assembled in LEO is efficiently raised to GEO (unmanned) by its low thrust propulsion system. Once at GEO the vehicle would receive a crew via a chemical OTV. The crew would then perform a final checkout of the vehicle before departure. From GEO, a chemical stage provides a fast escape departure at a greatly reduced propellant penalty (approximately 743,480 pounds) over LEO high thrust departure (approximately 2,540,000 pounds).

The departure hybrid option was seen to provide significant overall performance gains and operational flexibility for the departure phase. As alluded to above, Figure 3.2.2 summarizes the results of the departure trade and presents pertinent quantitative data.

3.2.3 At Mars Segment (Arrival/Departure)

The "at Mars" segment of the study represented the most involved area of research. Both high thrust and low thrust arrival and departure stages were investigated, for the representative orbits studied: (1) low circular Mars orbit 500 km (LMO cir); (2) areosynchronous orbit 20,450 km (AREO cir); (3) 48 hour period ellipse (48 hour ell.); (4) sphere of influence "flight maneuver;" and (5) rendezvous at the Mars libration point (L1).

For all of the above orbits in question, recall from Chapter 2 that separate transfer vehicles were sized that would taxi the landing party from the ship to the surface of Mars and back. Figure 3.2.3A presents the mass and TOF manifests for these vehicles.

Thus, with these transfer vehicles appropriately assigned, detailed propellant and time of flight maps for the descent/ascent requirements of the main interplanetary vehicle were calculated. Figures 3.2.3(B-D) outline the results of this work).

In relation to aerobraking technology, such systems were not found to be particularly attractive. In the case of the low thrust vehicle, the same failure modes exist for a Mars aerobrake as at the Earth. Also, for the two orbits which could possibly utilize this maneuver (i.e., LMO and the 48 hour ellipse) the added weight to allow aerobraking (retractable truss segments, and the aerobrake itself) were seen to cancel out any potential savings of fuel. In the case of the transfer vehicles, aerobraking was seen as being potentially beneficial. However, chemical propellant requirements, although more

massive, were seen as competitive. Thus, the all propulsive transfer vehicles were adopted. Table 3.2.3E represents the results of this study.

After a careful review of the data cited above and Figures 3.2.3(F-H), which show the net mass due to both braking of the primary vehicle and its associated transfer vehicle, several conclusions seem appropriate. First, from the results a reoccurring theme emerged. Figures 3.2.3 (F-H) show that for fixed alpha and heliocentric transfer time, high thrust entry to a particular operational orbit allows for short transfer time from the Mars SOI. However, for LMO or AREO the propellant requirements to do this are much higher than for all of the other options. Also, even though no propellant was necessary for the low thrust vehicle at L1 Mars, its associated taxi vehicle mass pushed this option out of consideration. A low thrust spiral to LMO looks attractive from a mass standpoint (it is consistently the third "lightest" option) but takes on the order of 64 days to spiral in and out of the gravity well. Of the other options remaining, all have total descent/ascent times on the order of 5 to 8 days and are thus considered equally competitive from this standpoint. From a mass standpoint, the elliptical high and low thrust orbits are only slightly more competitive than the SOI maneuver and the low thrust areosynchronous option.

From an operational standpoint, the aerosynchronous orbit represents a node from which one could observe and communicate with the crew at all times. Both the elliptical orbits and the SOI maneuver represent nodes which are more operationally demanding (both in terms

of rendezvous and abort strategies). Thus, of the remaining orbital nodes, the areosynchronous low thrust option was seen as the most attractive orbital node and propulsion option studied.

3.2.4 Heliocentric Flight Phase

As a result of the detailed mission models incorporated in this study, effects upon the vehicle (as related to overall mass) and its heliocentric performance (as related to round trip flight time envelopes) were observed.

The results are a series of detailed time of flight vs. mass requirement envelopes. Figures 3.2.4(A-C) present these results, and represent the final iterated output. They are organized by the assumed value of alpha (i.e., 22 lb/kW, 11 lb/kW, and 2.2 lb/kW). Within each figure, curves showing various arrival and departure orbits are specified. In this respect, the "at Mars" phase was the only varying orbit choice plotted since the hybrid Earth departure and arrival modes were settled upon early in the study. Thus, these figures demonstrate how significantly the "at Mars phase" impacts the mission scenarios studied.

After review of the data, several general trends emerge. The first was that, as expected, as trip time decreased propellant and hence total vehicle weight increased. In addition to the curves which represent vehicles run at 5 MW, the "*" indicate places where for the same propulsion system mass as used with 5 MW system--a vehicle was allowed to operate at 10 MW of power. This was done to determine the

effect of power level on mass and TOF performance. It was observed that time of flight envelopes could be slightly extended over the 5 MW case.

As expected, lower values of alpha produced lower overall vehicle weights. This mass reduction demonstrates how decreases in power plant mass for a constant power output can increase performance. For power generation system specific mass (alpha) values of 2.2 lb/kW and 11 lb/kW vehicles could achieve piloted roundtrip flight times on the order of a year if the power were increased to 10 MW. The other option, in which alpha equaled 22 lb/kW, achieved a minimum piloted trip time of around 1.25 years. These results are more pessimistic than those obtained from past "proof of concept studies". For example, References 6, 10, and 37 suggest that with a propulsion system consisting of a 22 lb/kW reactor and 5 MW of output power should be sufficient to produce a one to two year flight envelope for missions similar to the one studied in this report.

For future studies, if a 1-year piloted heliocentric time of flight is desired, two alternate design paths could be pursued. The first assumes that power levels on the order of 5 MW will remain the standard output power. With this constraint power level assumed, this work demonstrates that systems with specific masses of below 2.2 lb/kW would have to be developed to obtain 1 year piloted missions. On the other hand, if total power output capacities in the 10 MW range are pursued than specific weights even as high as 11 lb/kW will still permit 1 year round trip transfers. Materials that would allow

reactors to obtain specific weight levels as low as 2.2 lb/kW are not projected until an early 2000 time frame (i.e., 2010 to 2030) (Reference 16). Thus, it is the conclusion of this section that if time of flight reductions are desired, those reductions would most easily be obtained, in a near term environment, by boosting power output capabilities of reactor systems to levels on the order of 10 MW.

3.3 Baseline Low Thrust Mars Mission

After a careful review of the data presented thus far, several viable trends were seen to emerge. Of these, one combination of propulsion/on-orbit operations stood above the rest and shall be referred to throughout the rest of this work as the scenario of choice when low thrust propulsion is compared to other propulsion concepts in a generalized single or multi-use mission environment.

Specifically, based on the results from sections 3.1 and 3.2, the baseline mission incorporates the following:

The Mars vehicle is assembled in LEO. The low thrust propulsion system incorporates an ion thruster system with a specific impulse rating of 3,000 seconds, a nuclear reactor with a total power output capacity of 5 MW (nominal); 10 MW maximum, and an associated power plant specific mass (alpha) equal to 11 lb/kW.

After assembly in LEO the vehicle takes 270 days to spiral to GEO utilizing its primary propulsion system. This initial "spiral up" time is not counted in the total round trip time of flight envelopes because at this point, the vehicle is unmanned. With respect to total piloted travel time, this maneuver was deemed to fall under the category of a

"pre-flight readiness check". However, the LEO to GEO spiral fuel consumed is included in the overall mass estimates. This was done so that the total mass making up the vehicle in GEO and mass needed to get the vehicle to GEO were specified. In this way the net mass to orbit requirements for the mission are referenced to LEO (i.e., net mass estimates predict how much mass must be lifted to LEO for a given mission).

After the vehicle is at GEO, a crew of six rendezvous with the vehicle using an orbital transfer craft. Once the crew is aboard the Mars vehicle, they depart on an escape trajectory provided by a high thrust (480 seconds) chemical booster stage attached to the vehicle. Throughout the heliocentric flight phase (both outbound and inbound) only the low thrust system is used. At Mars arrival or departure, the low thrust system is again utilized spiraling to and from circular areosynchronous orbit. The descent and departure spiral time of flight phase takes 5 to 7 days for each maneuver. While parked at AREO, a landing party of four astronauts board a descent craft. This descent craft transfers the landing party to LMO and then descends to the surface. After 60 days of exploration, science experiments, etc., are performed by both the landing party and the on board crew, the landing party leaves the surface for rendezvous with the main ship. At this point, as stated above, a low thrust escape and transfer back to Earth is performed. Finally, at the Earth's SOI, the crew departs the main vehicle and returns to a low earth orbit in an Earth return vehicle

(which is chemically powered) while the main vehicle (using its low thrust propulsion system) spirals down to a geostationary storage orbit.

Table 3.3 outlines the specific design parameters chosen for this mission. In addition, Figure 3.3A also provide a graphic portrayal of the net mass history for this concept. Figures 3.3(B-C) show how a typical mission would unfold and an artists concept (Reference 44) of the heliocentric flight configuration of the vehicle.

3.4 Comparison of Baseline Concept with Alternative Propulsion

Schemes

The baseline vehicle and mission concept described in Section 3.3 provides a possible method of affecting a manned landing on the surface of Mars. However, no idea should or can be adequately judged in total isolation. Thus, the following section is intended to demonstrate how the single low thrust vehicle concept ranks against other vehicle designs which use different propulsion/mission schemes to achieve the same overall goal of placing a human expeditionary team on the surface of Mars.

3.4.1 Description of Alternate Approaches

The data used to do direct numerical comparisons with the baseline vehicle were taken directly from Reference 18. In regards to Reference 18, net vehicle mass (referenced to LEO) versus round trip flight data for missions requiring between 1 to 2 years of piloted flight time were presented. The mission objectives, payload assumptions, stay time ranges, infrastructure assumptions, etc., used by Reference 18 closely

matched those of the baseline vehicle. Because of these similarities, a fair comparison of these options and the baseline concept could be made.

The following represents a description of the alternative concepts that were used for comparative purposes. Specifically, they are:

- o A single all chemical (LOX/H₂) propulsion vehicle.
- o A dual vehicle scheme, which also uses LOX/H₂ as primary propulsion.
- o Another dual vehicle scheme, in which one vehicle uses LOX/H₂ while the other relies on a low thrust propulsion system.

In the single vehicle "all up(chem)" scenario, a single vehicle carries all of the fuel, equipment, habitation modules, etc., required for an entire mission. In the dual vehicle "split option (chem/chem)" scenario, one vehicle serves as a one way unmanned cargo vehicle carrying mission equipment, landers, etc., to Mars on a low fuel consumption trajectory. A second vehicle then serves as a manned "taxi" vehicle and carries the crew to Mars after the cargo vehicle has safely arrived at Mars. At Mars arrival, the taxi vehicle rendezvous with the cargo vehicle and use the equipment on board. Lastly, the crew would return home in the taxi vehicle by using a return stage which would also be initially carried to Mars by the cargo vehicle. Lastly, the second dual vehicle option known as the "split option (NEP/chem) replaces the chemical cargo vehicle with one propelled by an electric propulsion system. The low thrust cargo vehicle has a power level (2 to 4 MW) and specific weight ($\alpha = 22 \text{ lb/kW}$).

In addition, Reference 18 also provided net vehicle mass versus round trip flight time data for each of the above described options in which aerobraking maneuvers at both earth and mars were incorporated into the basic mission models described. Table 3.4.1 provides a summary of the alternative approaches discussed.

3.4.2. Comparison of Single Mission Data

Figures 3.4.2(A-B) highlight the major differences between the Baseline option and the others discussed in Section 3.4.1. Specifically, the figures show the low Earth orbit initial mass requirements for each concept in 200,000 pound increments vs. the piloted round trip time in years.

The most striking feature in both figures is the dramatic mass saving that the Baseline option has over the all up (CHEM) and split option (CHEM/CHEM) options. This mass differential is most dramatic in the 1.25 year time frame (between 50 percent to 75 percent mass savings). It should also be noted that when the power level for the Baseline option is allowed to vary up to a maximum of 10 MW in the 1 to 1.25 year time frame, this mass trend becomes even more pronounced (60 to 87 percent savings).

Another obvious trend is that as the total trip time increases to 2 years, all of the options converge. However, the mass savings of the Baseline option as opposed to the chemical vehicles is still between 40 and 50 percent. The dramatic affect that aerobraking has on the chemical options should also be noted. This aerodynamic technique lowers the mass requirements of the all up (CHEM) option by 50 percent in a 1 year to 1 1/2 year time frame. To a lesser extent, the split

option (CHEM/CHEM) is helped by aerobraking (by about 23 percent) while the split option (NEP/CHEM) with aerobraking is lower than the Baseline option (by a 20 percent difference).

Thus, based on the above comparison it appears that for a single exploratory mission the Baseline option offers a competitive to substantial mass savings. The only option that is more competitive is the logistically complex split option (NEP/CHEM) vehicle with aerobraking included.

3.4.3 Comparison of Multiple Mission Data

Unlike Section 3.4.2 which only considered the mass requirements for a single Mars mission, the following section assumes that multiple Mars expeditions will take place. For the purposes of simplicity, three mission cycles were chosen to represent this multiple use of the Mars scenarios discussed.

For the vehicles outlined in Reference 18 only the habitation modules (which the crew is assumed to return in) are considered refurbishable. For the Baseline vehicle all those elements returned to the assembly facility are considered refurbishable (see Table 3.4.3). The trade outlining the multiple mission servicing for the Baseline vehicle is discussed in Appendix 3. From this analysis, the LEO refit option was used.

Figures 3.4.3(A-C) highlight the results of the multiple mission study. For each indicated round trip time the total mass of the three mission cycles is presented for each propulsion scheme. The data represents the amount of mass required in low earth orbit to produce

three separate Mars vehicles under each option if components from preceding missions are allowed to be reused.

Several trends emerged from the study. As suspected from the "single use" data, the Baseline vehicle was among the minimum mass options in this flight environment. As in Section 3.4.2, net differences between the Baseline vehicle and other options is more pronounced around the 1 to 1.5 year time frame. However, even at the 2 year interval these differences remain more significant in the multi-use environment. In the case of an all up (CHEM) option, the net mass for three separate Mars missions with a 1.25 year round trip time of flight was calculated to be 24 million pounds. This is opposed to the Baseline option which has a calculated net mass of 4.8 million pounds.

The data shows several other interesting trends. As before the effect that aerobraking has on chemical missions (1 to 1.5 years) is very pronounced. In the 2 year time of flight range, only slight benefits from aerobraking are possible.

In comparison, the Baseline option is seen as a competitive mass option and one that is very constant in total mission mass from the 1 to 2 year range. This suggests that unlike the other options which must keep going to more complicated modes of operation to increase their performance, the Baseline vehicle concept offers a streamlined option which can perform efficiently in a 1 to 2 year range.

Thus, as in the "single use" environment the multi-use enhancements offered by the all Baseline vehicle option allow robust access to interplanetary space at reduced mass requirements and

streamlined logistical uncertainties. If multiple exploratory missions are deemed a capability that the space community will one day desire, the Baseline vehicle should be seen as the most efficient concept.

CHAPTER 4.0 FINAL ASSESSMENT OF RESULTS AND CONCLUDING REMARKS

In light of the results presented, several observations can be made. The first is that the application of low thrust systems for primary propulsion in manned Mars mission schemes beyond the traditional "proof of concept" level have been thoroughly reviewed.

A detailed study concerning the planetocentric phases of a manned Mars mission was undertaken. The results of this effort indicate that such operational aspects, previously neglected by many past studies, do indeed have a dramatic impact on overall mission design. Both the initial on orbit mass and round trip flight envelopes for electric propulsion vehicles varied greatly depending on the particular option chosen at each phase.

The study concluded that a multi-megawatt nuclear power source and a thruster assembly that can generate specific impulse levels on the order of 3,000 seconds was necessary.

Overall, one low thrust vehicle/mission concept proved to be superior to all other low thrust concepts studied. This Baseline vehicle (see Table 3.3 and Figures 3.3(A-C)) incorporated a high thrust departure stage at earth departure (BOM), while utilizing the areosynchronous orbital node for operations at Mars. The vehicle also carried a deployable crew return capsule once the vehicle re-entered the Earth's sphere of influence at the end of mission phase.

With minimum technology and infrastructure thresholds determined, the only barrier to the concept technologically lies in the development

of a multi-megawatt space worthy nuclear reactor system. In this regard, Project Pathfinder (Reference 30) represents the kind of development program necessary to bring this last remaining key element into being.

Additionally, a pragmatic comparison was under taken of the Baseline concept with manned mars mission schemes. The results of this study demonstrate, for both single and multiple mission environments, that low thrust systems can provide a very competitive propulsion alternative.

In conclusion, it should be remembered that major revolutions in transportation are always preceded by breakthroughs in propulsion technology. Electric propulsion, as discussed in this thesis work, represents just such a breakthrough. Its possible applications and impact on space transportation should prove to be similarly revolutionary. It will provide those who choose to utilize it with the ability to lead in the exploration and exploitation of the many resources available in the solar system.

CHAPTER 5.0 RECOMMENDATIONS FOR FUTURE WORK

Under the original goals set forth at the beginning of this study, all major areas of interest were investigated to a sufficient level for the trade study that this thesis represents. However, for further completeness additional detailed "iterations" can and should be done to further quantify various aspects of such a mission. Of these, three major areas stand out as requiring further investigation. The first, to ease computation, would be to identify a "QUICKTOP type" fully automated low thrust trajectory optimization code. Although the QUICKTOP III computer code required very little computer memory (VAX level of operation) and proved to be adequate for the task at hand, the manual splicing of out-bound and in-bound trajectory legs was found to be a tedious exercise. Currently, automatic round trip codes are either not available or rely on mathematical formulations that require very large computer memory allocations (on the order of a CRAY computing system). It is, therefore, the recommendation of this thesis work that the QUICKTOP III computer code be modified to handle automatic round trip trajectory and mass optimization. At present this task is being undertaken by a team composed of GWU-JIAFS graduate students and NASA engineers who currently work at the NASA-Langley Research Center's Vehicle Analysis Branch located in Hampton, Virginia.

The second recommendation is that a detailed structural analysis of the Baseline vehicle concept with an assessment as to its actual configuration be undertaken. Better subsystem mass estimates and closer examination of the structural components for the vehicle are

necessary to establish even more accurate mass estimates for the entire concept. Lastly, the entire assembly, servicing and OTV infrastructure assumptions require further detailed study. By quantifying these areas, the complete scope of the basic operational infrastructure facilities assumed available for the Baseline mission can be more realistically understood.

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TABLES

TABLE 2.2.1

MISSION/VEHICLE CONSTRAINTS

- EXPLORATION/PROSPECTOR CLASS MISSION
- SINGLE VEHICLE CONCEPT WITH MARS SURFACE LANDER
- STAY TIME AT MARS OF 60 DAYS
- ARTIFICIAL GRAVITY ENVIRONMENT (1g)
- CREW OF SIX
- PILOTED ROUND TRIP FLIGHT TIME (1 TO 2 YEARS CONSIDERED)

TABLE 2.2.1A

MANIFEST OF PAYLOAD SUBSYSTEMS COMMON TO ALL MISSIONS

COMMON VEHICLE COMPONENTS (1-2 YEAR MISSIONS)	MASS (LB)
● HABITATION MODULES (TWO) (REFERENCE 16)	
SUBSYSTEMS:	
PRIMARY STRUCTURE	11,400
SECONDARY STRUCTURE	8,510
ENVIRONMENTAL CONTROL	13,980
LIFE SUPPORT	29,810
CREW SYSTEMS	2,660
COMMUNICATIONS	2,310
DATA MANAGEMENT	530
GUIDANCE AND NAVIGATION	250
ATTITUDE CONTROL	7,280
DISPLAYS AND CONTROLS	510
ELECTRICAL POWER	10,850
EXPERIMENT EQUIPMENT	12,290
GROWTH ALLOWANCE	16,200
TOTAL MASS:	116,580
● SPACECRAFT STRUCTURE	
PRIMARY TRUSS (REFERENCE 16)	93,196
SECONDARY STRUCTURE (ANTENNA ASSEMBLY, MISC.)	11,023
ARTIFICIAL GRAVITY SYSTEM (25% CONTINGENCY)	26,055
TOTAL MASS:	130,274
● PROBES AND EXPERIMENTS (REFERENCE 19)	<u>24,462</u>
TOTAL PAYLOAD MASS COMMON TO ALL MISSIONS	271,316

TABLE 2.2.2

INFRASTRUCTURE ASSUMPTIONS

- ON-ORBIT ASSEMBLY CAPABILITY AT A LEO NODE
- AVAILABILITY OF A HEAVY LIFT LAUNCH VEHICLE (200,000 LB TO LEO)
- ORBITAL TRANSFER VEHICLE NETWORK (CAPABLE OF TRANSPORTING CREWS AND/OR CARGO TO AND FROM POINTS INSIDE THE EARTH'S SPHERE OF INFLUENCE)

TABLE 3.1

GENERAL LOW THRUST SPACECRAFT DETERMINATIONS

● THRUSTER TYPE	ELECTROSTATIC ION
-- DESIGN POINT ISP	3,000 SEC
-- LOW THRUST FUEL	ARGON
● POWER SOURCE	NUCLEAR REACTOR
-- SYSTEM MASS RANGE INVESTIGATED (MEASURED IN POUNDS/KILOWATT)	22, 11, 2.2
● POWER LEVEL REQUIRED	MULTI-MEGAWATT
-- NOMINAL OPERATING RANGE	5 MW
-- MAXIMUM OUTPUT LEVEL	10 MW

TABLE 3.3

BASELINE VEHICLE DESIGN PARAMETERS

<u>SPACECRAFT RELATED PARAMETERS</u>		
● PROPULSION		
-- THRUST TYPE		ELECTROSTATIC ION
- DESIGN POINT ISP		3,000 SEC
- LOW THRUST FUEL		ARGON
-- POWER SOURCE		NUCLEAR REACTOR
- POWER PLANT SPECIFIC MASS		11 LB/kW
-- POWER LEVEL		MULTI-MEGAWATT
- NOMINAL		5 MW
- MAXIMUM		10 MW
<u>TRAJECTORY RELATED PARAMETERS</u>		
● EARTH DEPARTURE		
-- ASSEMBLY IN LEO		
-- LOW THRUST TRANSFER LEO TO GEO (UNMANNED)		270 DAYS
-- HT DEPARTURE FROM GEO TO SOI EARTH (MANNED)		
- CREW TRANSPORTED TO GEO VIA AN ORBITAL TRANSFER VEHICLE		
● HELIOCENTRIC TRANSFERS		
-- LOW THRUST (C-T) TRAJECTORIES TO AND FROM MARS		
● MARS ARRIVAL AND DEPARTURE		
-- ORBIT OPERATIONS NODE		AREOSYNCHRONOUS ORBIT
-- STAY TIME AT MARS		60 DAYS
-- LT ASCENT/DESCENT		5-7 DAYS
-- ALL PROPULSIVE MEM UTILIZED		-----
-- SURFACE PARTY		4
-- COMMAND CREW (IN ORBIT)		2
● EARTH RETURN		
-- HYBRID RETURN OPTION UTILIZED		
- UPON REACHING SOI (EARTH) CREW RETURNS TO LEO IN ERV(CHEM)		~ 6 DAYS
- AT SOI (EARTH) NEP VEHICLE TRANSFERS TO GEO VIA LT DESCENT		~ 60 DAYS

TABLE 3.4.1

ALTERNATIVE MISSION CONCEPTS*

MISSION MODE	NUMBER OF VEHICLES	DESCRIPTION	FUEL	PERFORMANCE	FLIGHT MODE	MASS AND TOF DATA AVAILABLE ALL IMPULSIVE	UTILIZED AEROBRAKE
ALL UP(CHEM)	ONE	ALL CREW AND EQUIPMENT ABOARD ONE VEHICLE	LOX/H2	ISP - 450	ROUND TRIP	YES	YES AT EARTH AND MARS
SPLIT OPTION (CHEM/CHEM)	TWO	MANNED TRANSPORT VEHICLE CARGO VEHICLE	LOX/H2	ISP - 450	ROUND TRIP ONE WAY	YES	YES AT EARTH AND MARS
SPLIT OPTION (NEP/CHEM)	TWO	MANNED TRANSPORT VEHICLE CARGO VEHICLE	LOX/H2 NEP	ISP - 450 POWER (2-4 MW) ALPHA - 22 LB/kW	ROUND TRIP ONE WAY	YES	YES AT EARTH AND MARS

* (REFERENCE 18)

NOTE: (ALTERNATIVE MISSION ASSUMPTIONS)

- LEO VEHICLE ASSEMBLY, LAUNCH, AND RETURN
- ELLIPTICAL ORBIT UTILIZED AS OPERATIONAL NODE AT MARS

TABLE 3.4.3

BASELINE VEHICLE COMPONENTS
(MULTI-MISSION ENVIRONMENT)

VEHICLE COMPONENTS	REPLACE (EOM)	REFURBISH (EOM)
● PROPULSION SYSTEM		
-- POWER PLANT		Y
-- RADIATOR		Y
-- THRUSTER SUBSYSTEM		Y
● MAIN VEHICLE STRUCTURE		
-- BASIC TRUSS		Y
-- ARTIFICIAL GRAVITY STRUCTURE		Y
-- SECONDARY STRUCTURE		Y
● HABITATION MODULES		Y
● MARS EXCURSION MODULE	Y	
● EARTH RETURN VEHICLE		
-- STRUCTURE		Y
-- PROPELLANT	Y	
● HIGH THRUST GEO DEPARTURE STAGE	Y	
● INTERPLANETARY LT FUEL	Y	
● LEO TO GEO PRE-FLIGHT FUEL	Y (*)	

* MUST ADDITIONALLY INCLUDE GEO TO LEO POST-FLIGHT FUEL FOR TRANSFER BACK TO ASSEMBLY NODE.

NOTE: RETURNED HABITATION MODULES FOR ALTERNATIVE MISSION CONCEPTS (TABLE 3.4.1) ARE ALSO CONSIDERED REFURBISHABLE IN A MULTI-MISSION ENVIRONMENT.

SUPPLEMENTAL TABLE I

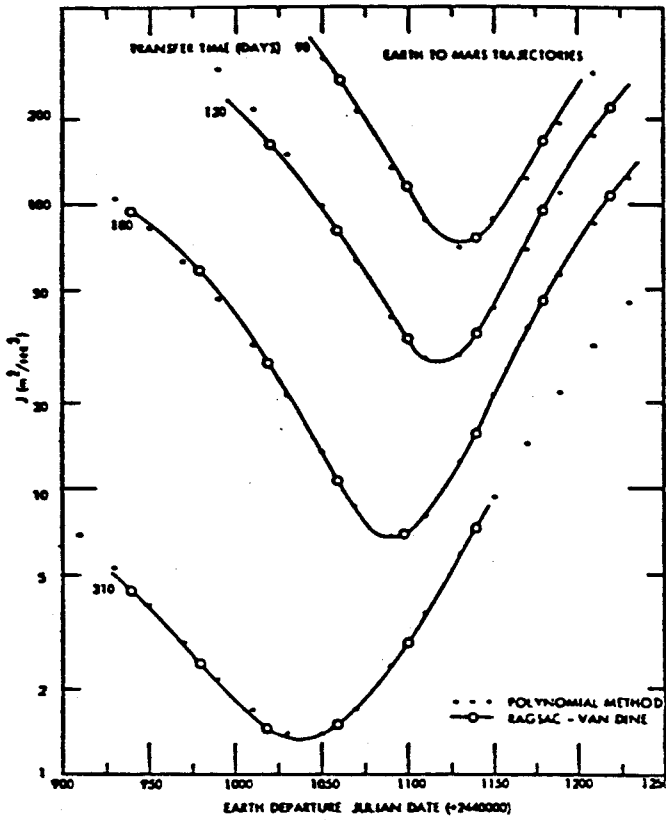
ORBITAL PARAMETERS AND ASTRODYNAMIC CONSTANTS*

● DISTANCE (MILES)	
RADIUS OF THE EARTH	3,964
EARTH - SUN DISTANCE	9.2915E+7
LOW EARTH ORBIT (LEO)	310 (ALT.)
GEOSYNCHRONOUS EARTH ORBIT (GEO)	2.6205E+4
RADIUS OF SPHERE OF INFLUENCE (EARTH)	5.7396E+5
RADIUS OF MARS	2,101
MARS - SUN DISTANCE	1.4158E+8
LOW MARS ORBIT (LMO)	2,411
48 HOURS ELLIPSE AT MARS (48 HR. ELL.)	
PERIGEE	2,411
APOGEE	37,527
ECCENTRICITY (DIMENSIONLESS)	.8792342
AREOSYNCHRONOUS MARS ORBIT (AREO)	12,710
RADIUS OF SPHERE OF INFLUENCE (MARS)	3.5905E+5
MARS LIBRATION POINT ONE (L1)	6.57180E+5
● MASS (LB)	
EARTH	1.3183E+25
MARS	1.3977E+25
SUN	4.3871E+30
● OTHER (MI ³ /SEC ²)	
MARS GRAVITATIONAL PARAMETER (U _{mars})	8.9189E+3
EARTH GRAVITATIONAL PARAMETER (U _{earth})	8.258E+4
SUN GRAVITATIONAL PARAMETER (U _{sun})	2.7492E+10
GRAVITATIONAL ACCELERATION AT SEA LEVEL-EARTH (G _e)	32.81 FT/SEC ²

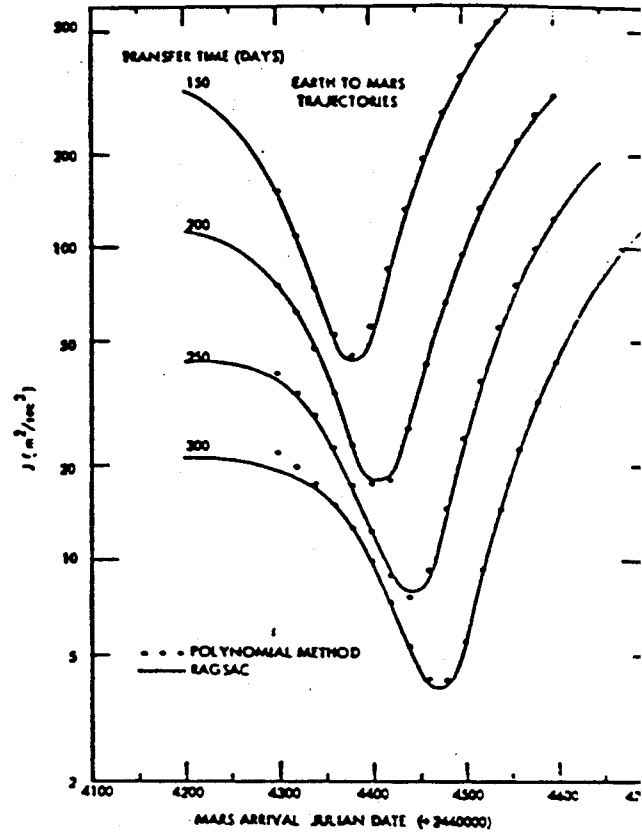
* VALUES USED IN THIS WORK (TAKEN FROM REFERENCE 22).

FIGURES

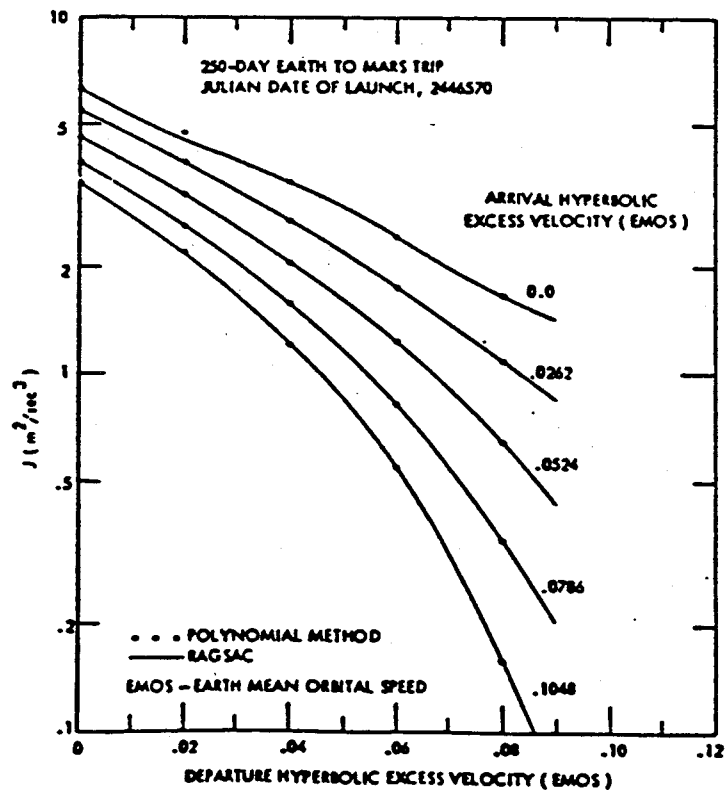
Figures 2.5.3(A-C) [Reference 11]
Code Validation Comparison Data



Two-dimensional rendezvous.
Figure 2.5.3A



Three-dimensional rendezvous.
Figure 2.5.3B



Effect of hyperbolic excess velocities.
Figure 2.5.3C

Figure 3.2.1A

Earth Return Mode Options

(Mass and Time of Flight Data)

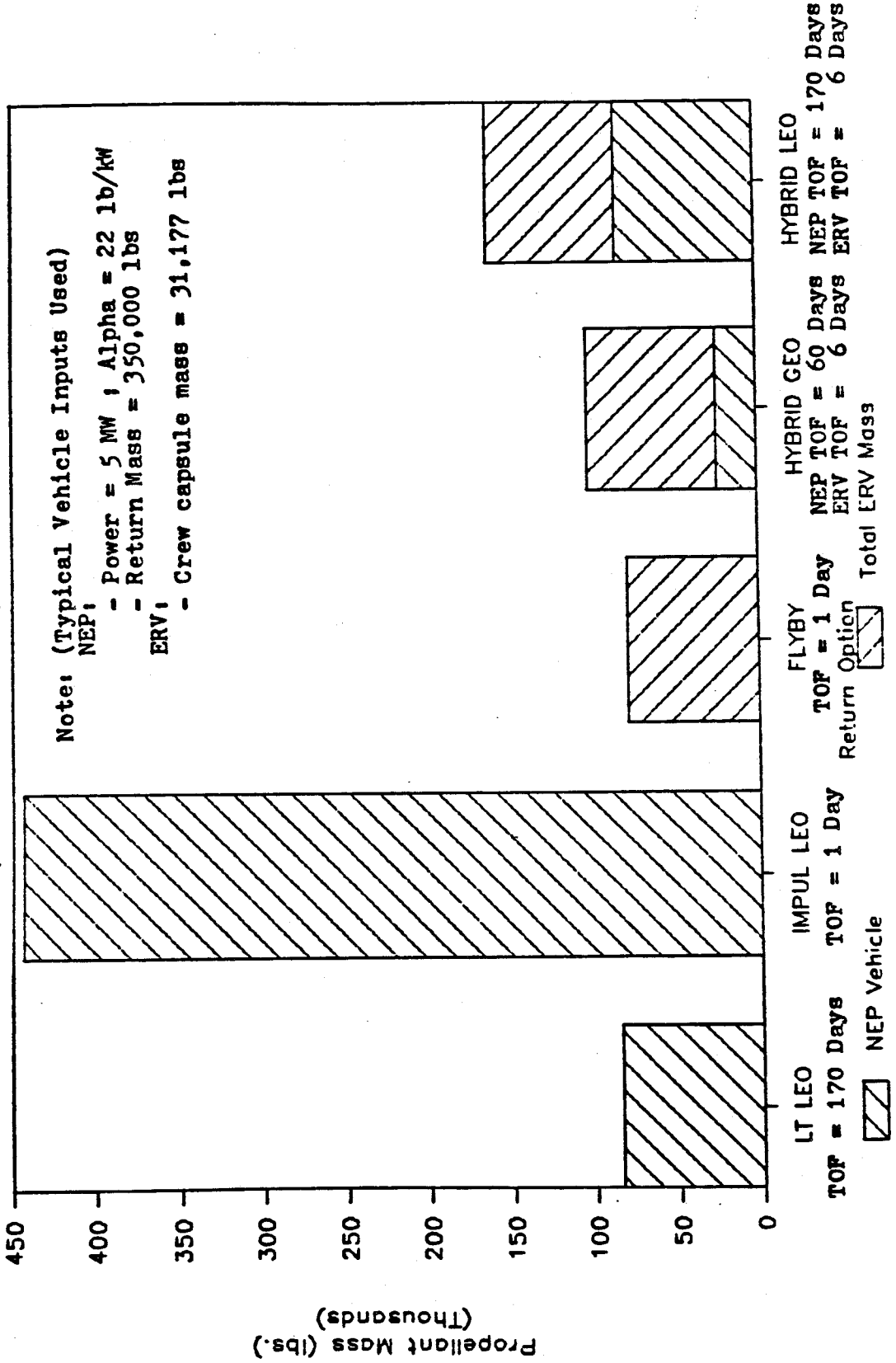


Figure 3.2.1B
Earth Return Vehicle Sizing Predictions
 (Aerobrake vs. Chemical)

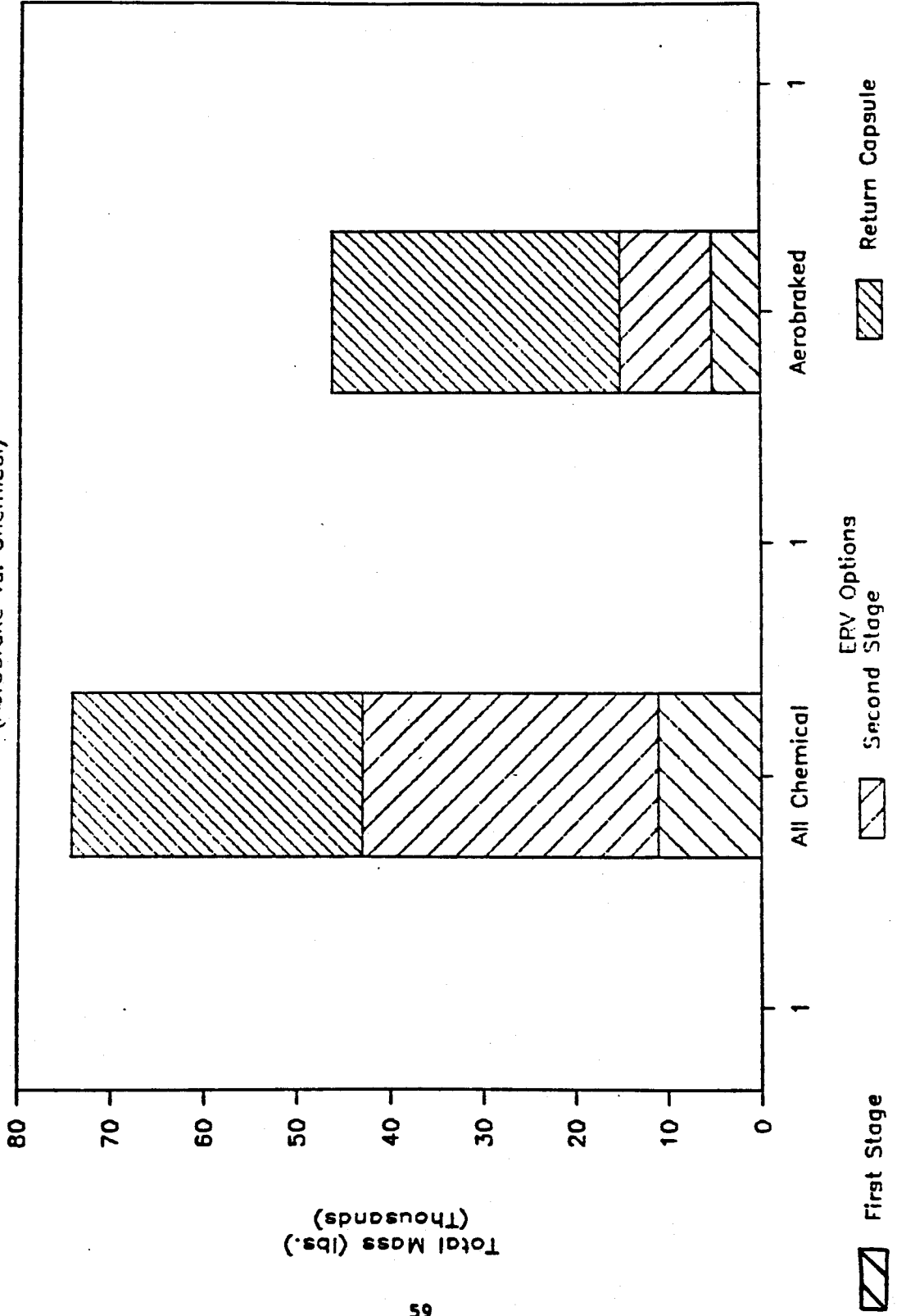


Figure 3.2.2

Earth Departure Options

(Mass and Time of Flight Data)

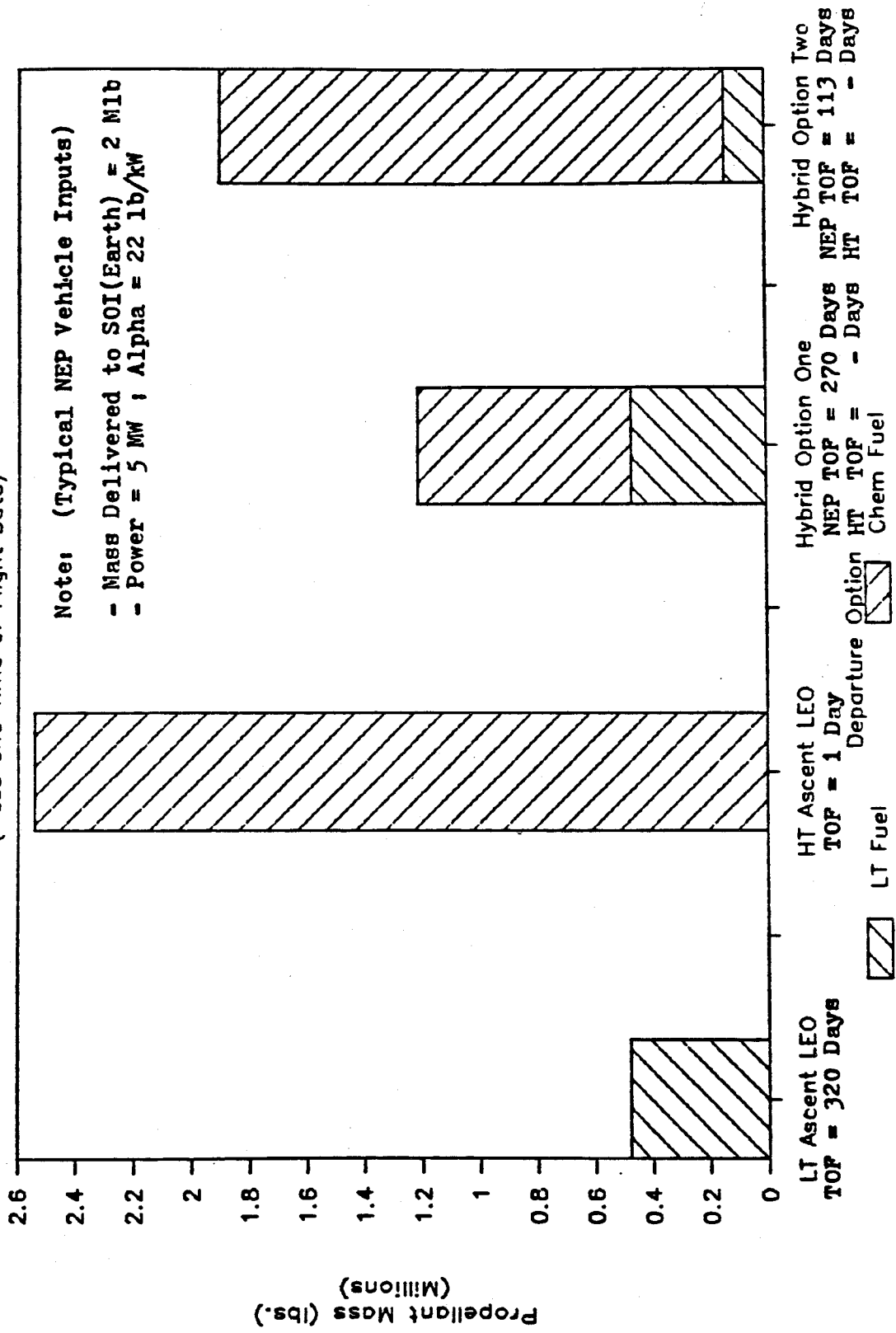


Figure 3.2.3A

Mars Excursion Module

(Mass and Time of Flight Data)

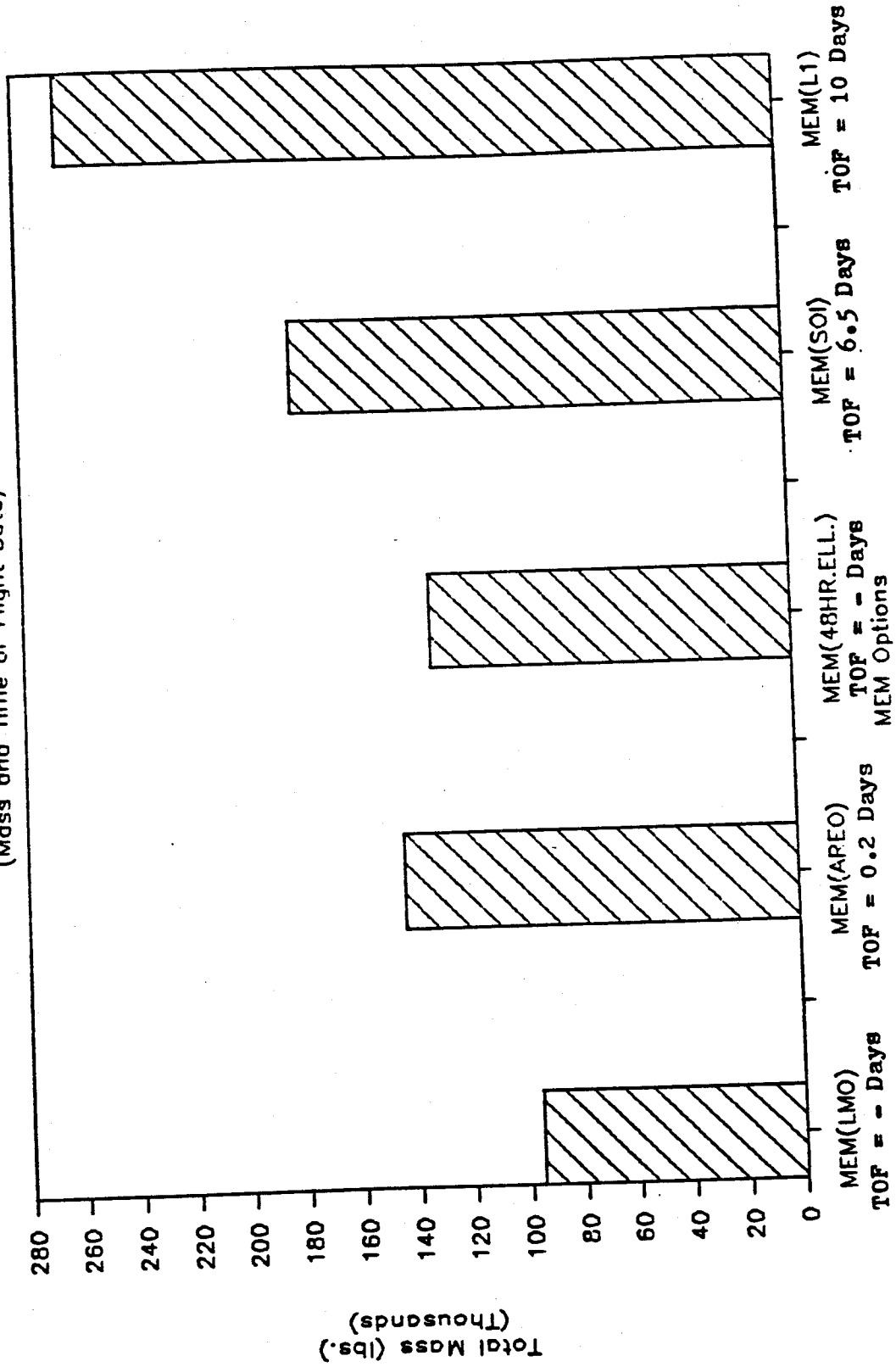
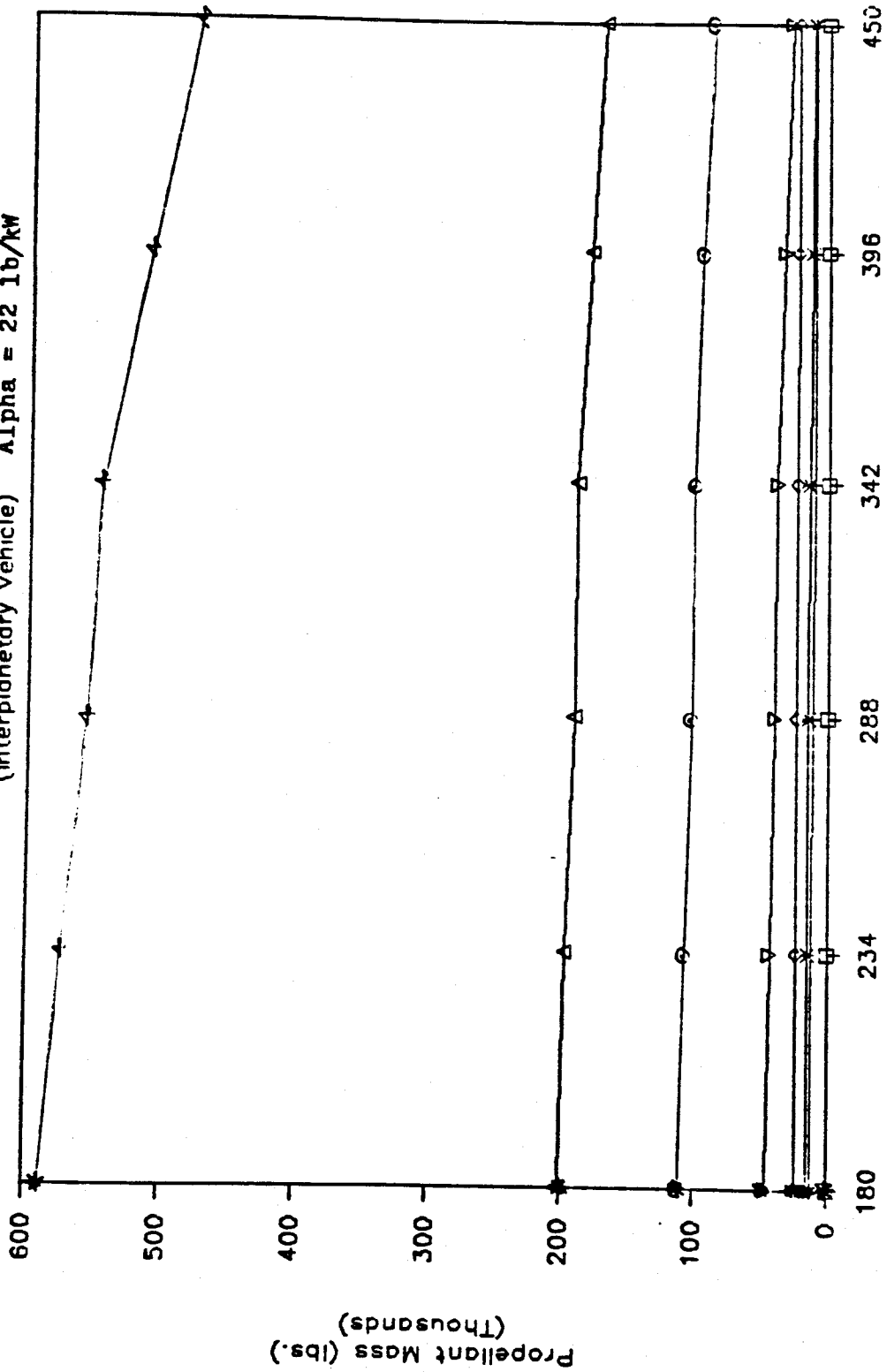


Figure 3.2.3B

Mars Ascent/Descent Propellant Req.

(Interplanetary Vehicle) Alpha = 22 lb/kW



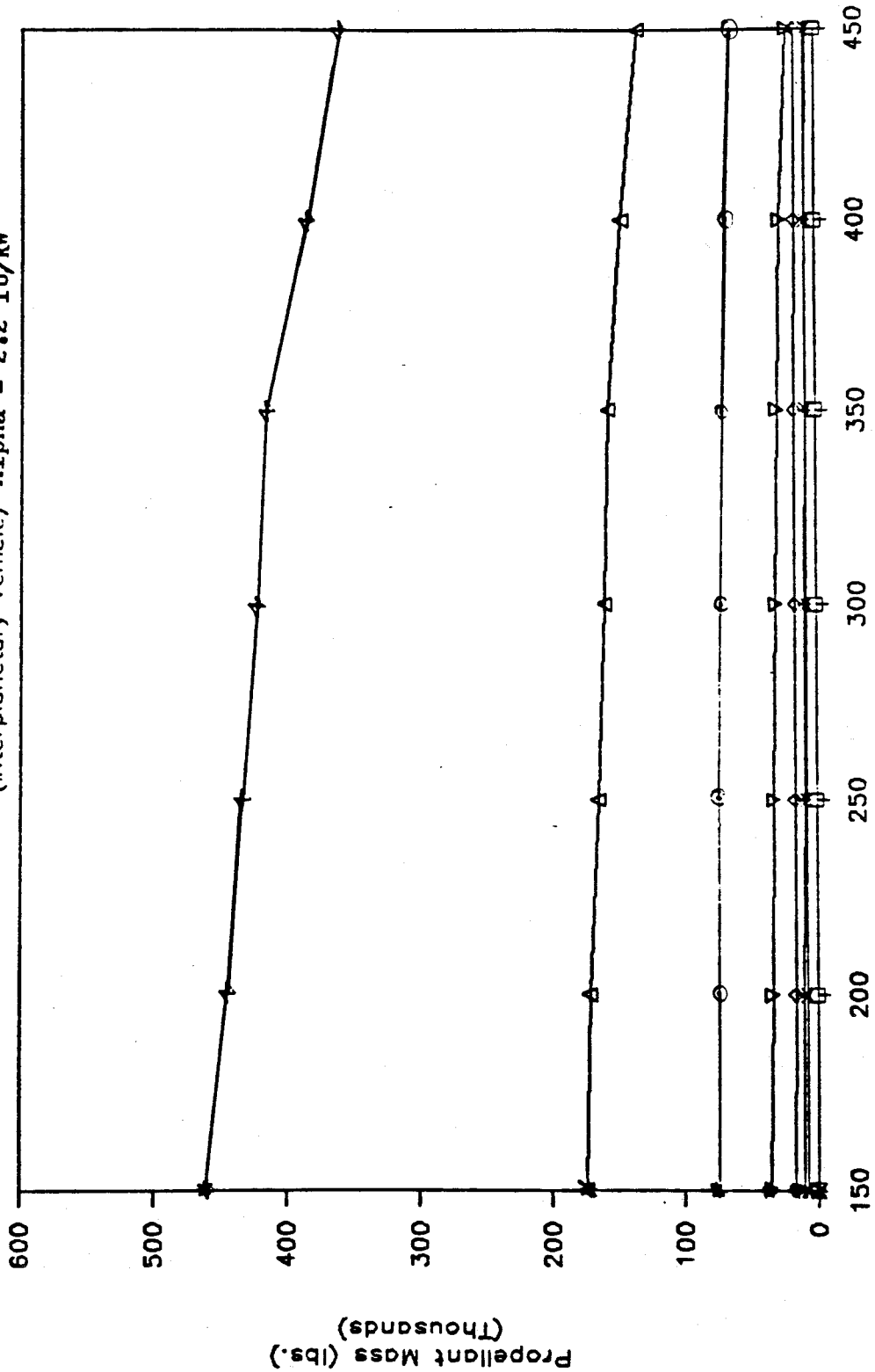
□ L1(Mars) + SOI Maneuver ○ AREO(LT) Δ AREO(HT) X 48 Hr.E11(LT) ▽ 48 Hr.E11(HT)
 ○ LMO(LT) ◄ LMO(HT)

* Power level raised from 5 MW to 10 MW

Figure 3.2.3D

Mars Ascent/Descent Propellant Req.

(Interplanetary Vehicle) Alpha = 2.2 lb/kW



□ L1(Mars) + SOI Maneuver ◊ ARSO(LT) Δ ARSO(HT) × 48 Hr.E11(LT) ▽ 48 Hr.E11(HT)
○ LMO(LT) † LMO(HT)

* Power level raised from 5 MW to 10 MW

Figure 3.2.3E

MEM Sizing Predictions (Aerobrake vs. Chemical)

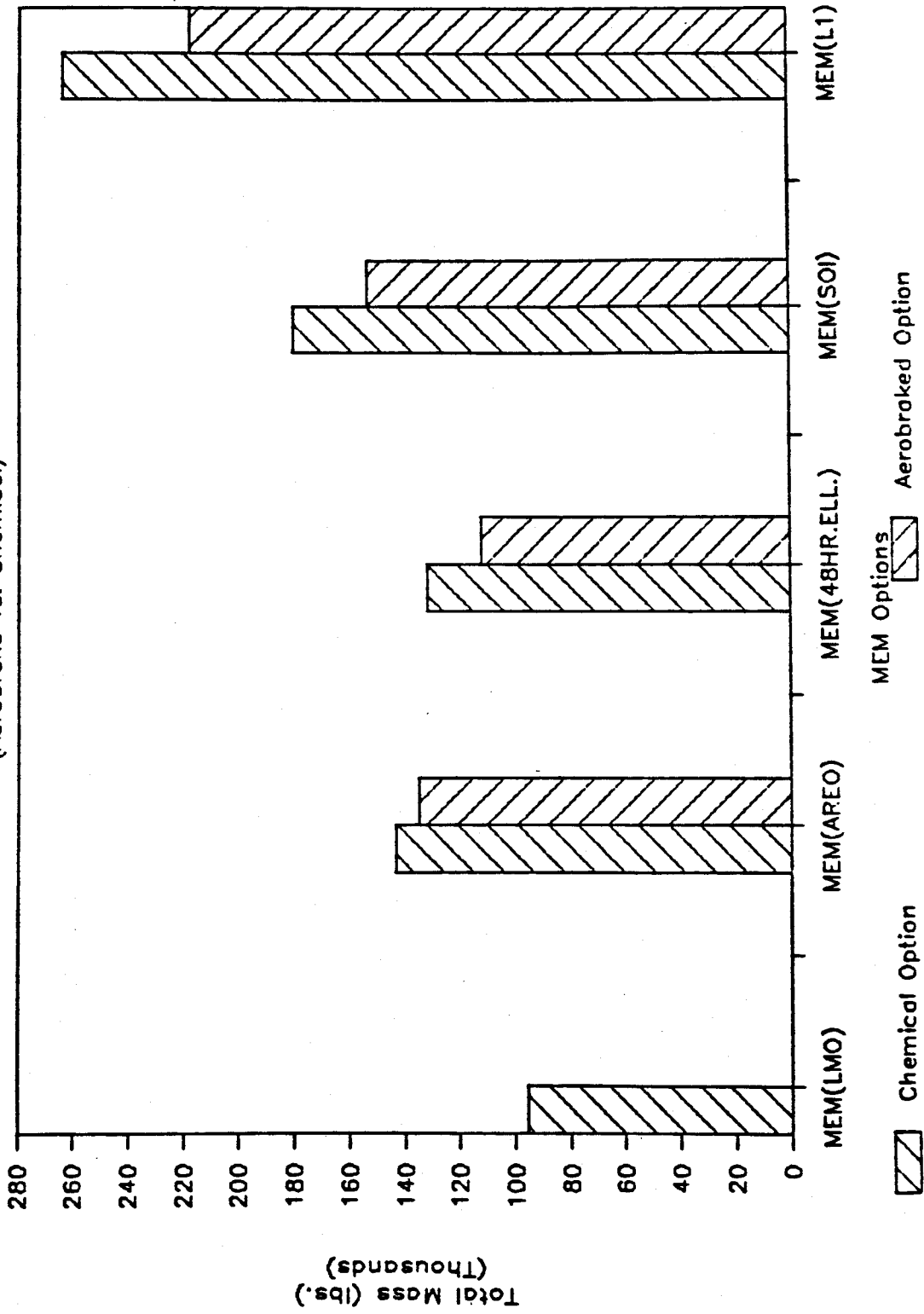
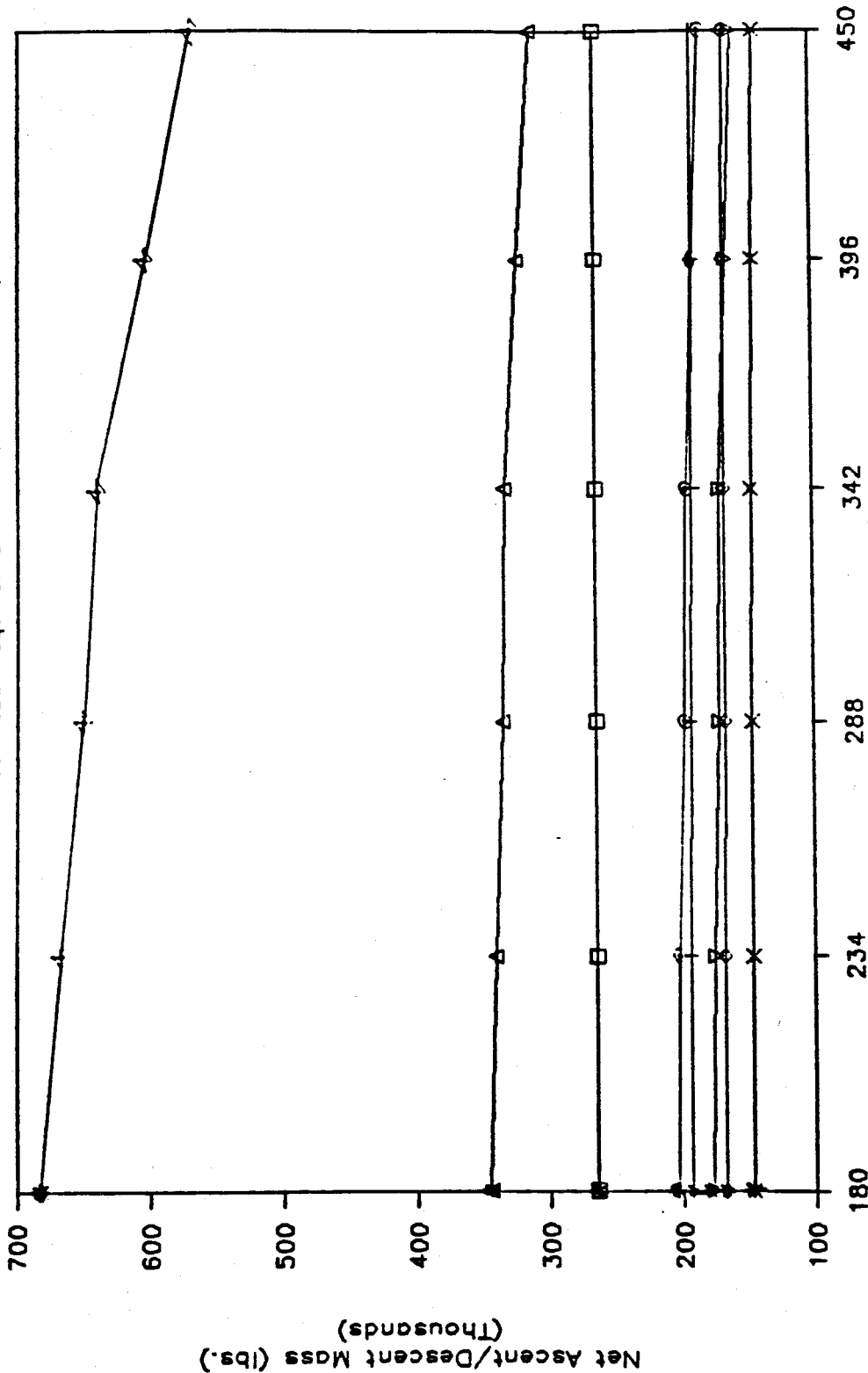


Figure 3.2.3F

Mars Ascent/Descent

Net Mass Requirements Alpha = 22 lb/kW



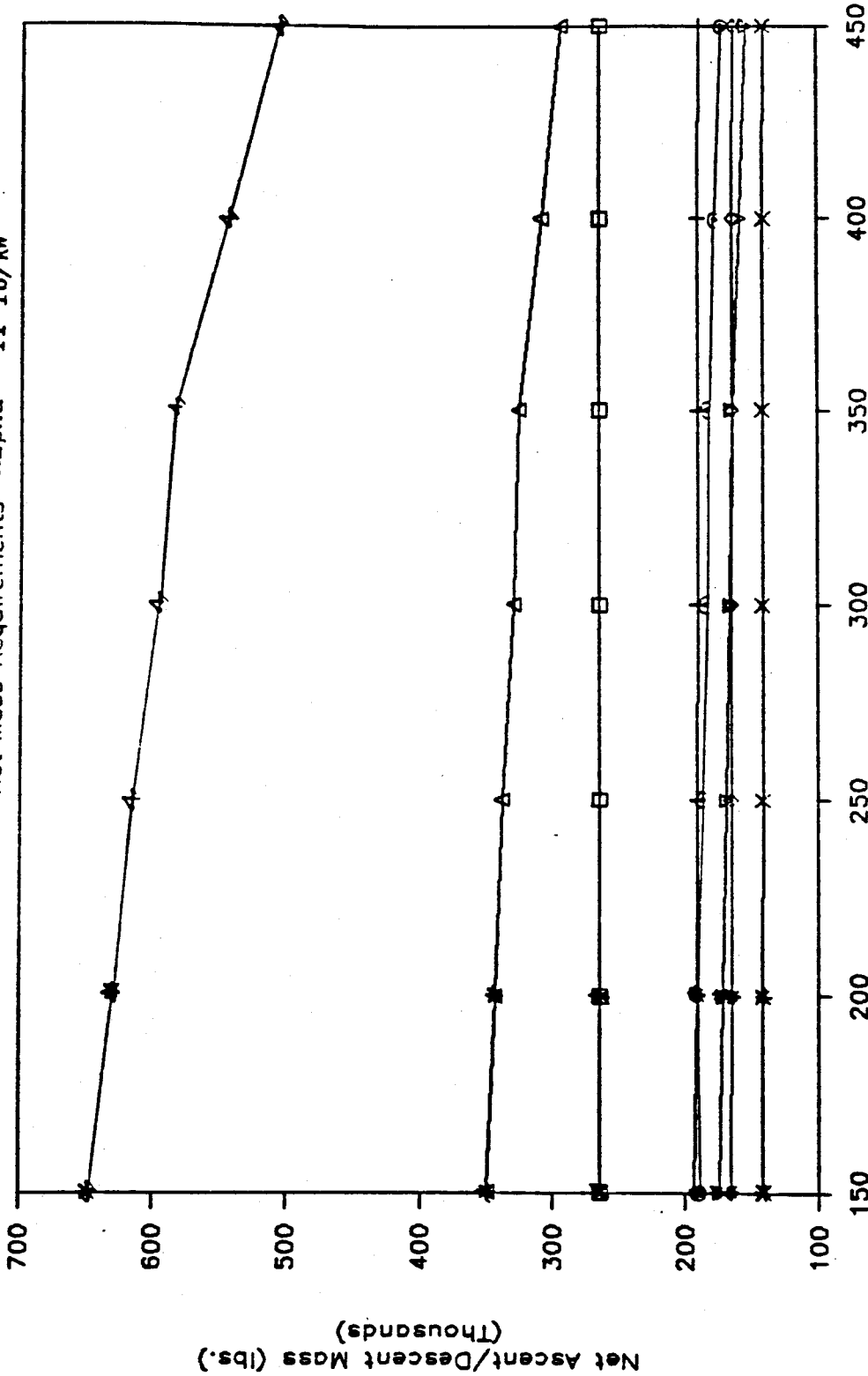
□ L1(Mars) + SOI Maneuver ○ AREO(LT) △ AREO(HT) × 48 Hr.E11(LT) ▽ 48 Hr.E11(HT)
 ○ LMO(LT) ◀ LMO(HT)

* Power level raised from 5 MW to 10 MW

Figure 3.2.3G

Mars Ascent/Descent

Net Mass Requirements Alpha = 11 lb/MW



L1(Mars)
 + SOI Maneuver
 ◊ AREO(LT)
 Δ AREO(HT)
 X 48 Hr.E11(LT)
 ▽ 48 Hr.E11(HT)
 M-E Return Transfer Time (Days)

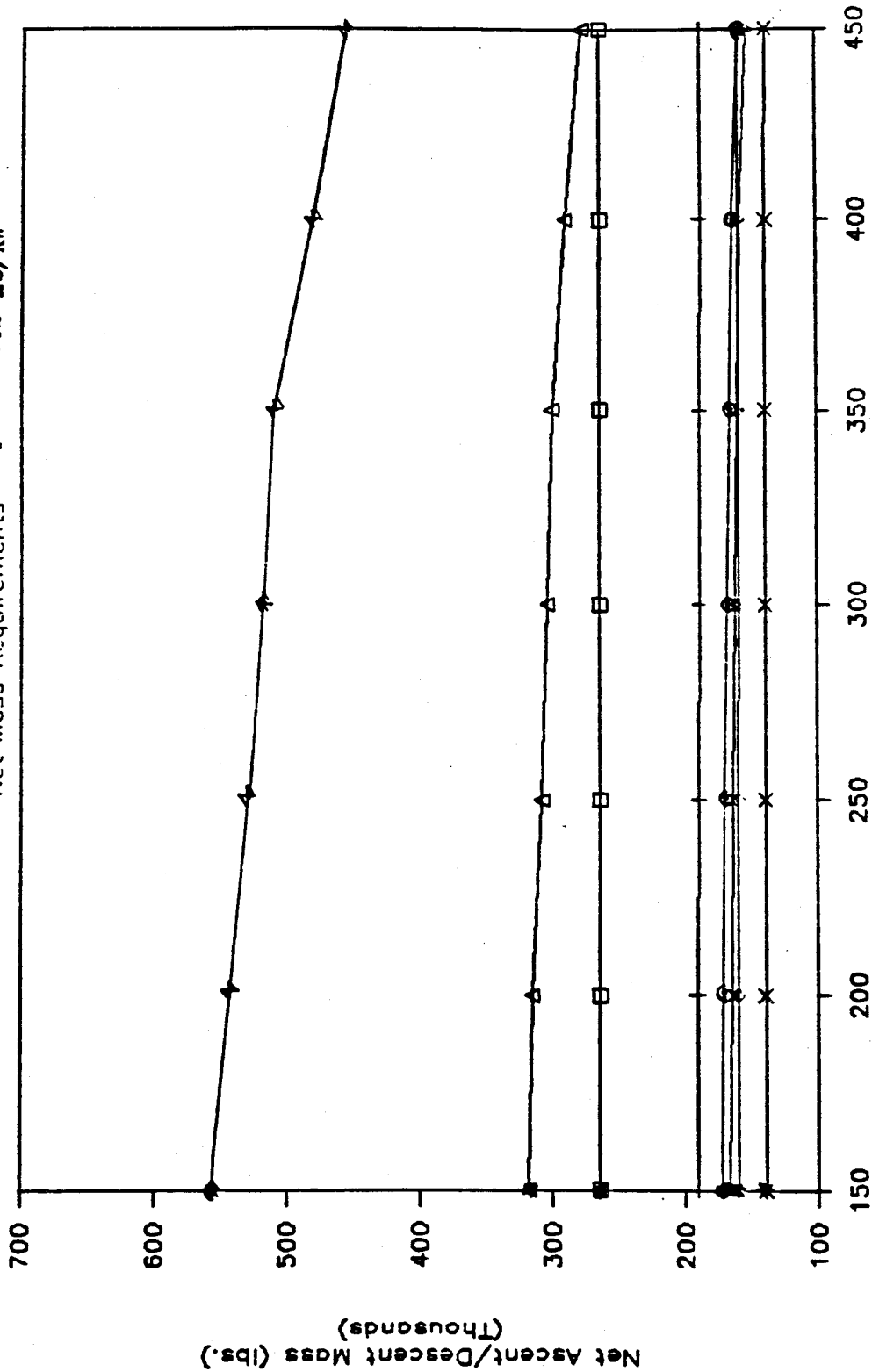
○ LMO(LT)
 ↗ LMO(HT)

* Power level raised from 5 MW to 10 MW

Figure 3.2.3H

Mars Ascent/Descent

Net Mass Requirements Alpha = 2.2 lb/kW



L1(Mars)
 + SOI Maneuver
 ◊ AREO(LT)
 Δ AREO(HT)
 X 48 Hr. E11(LT)
 ▽ 48 Hr. E11(HT)

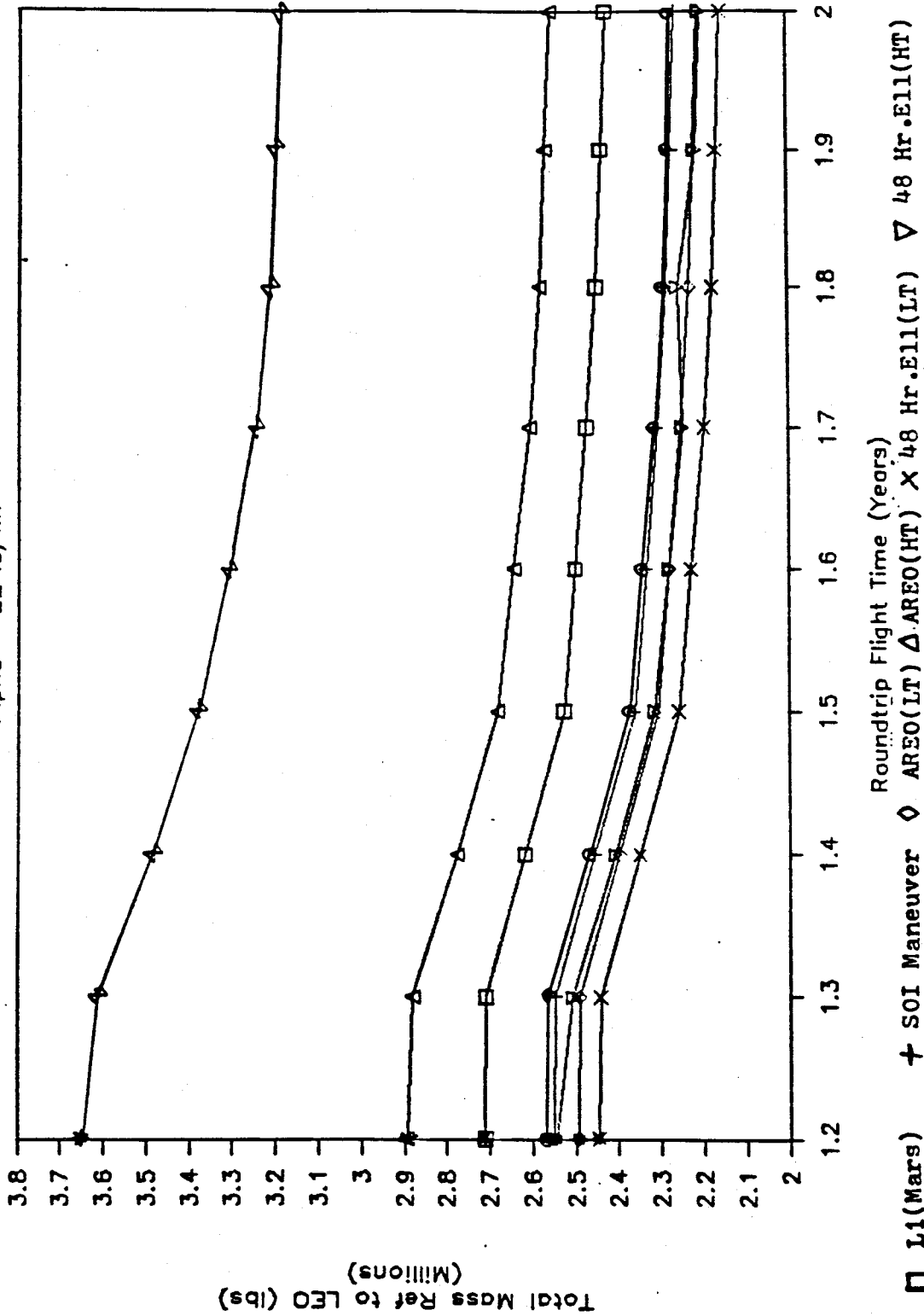
○ LMO(LT)
 † LMO(HT)

* Power level raised from 5 MW to 10 MW

Figure 3.2.4A

Mass vs. Roundtrip Flight Time

Alpha = 22 lb/kW

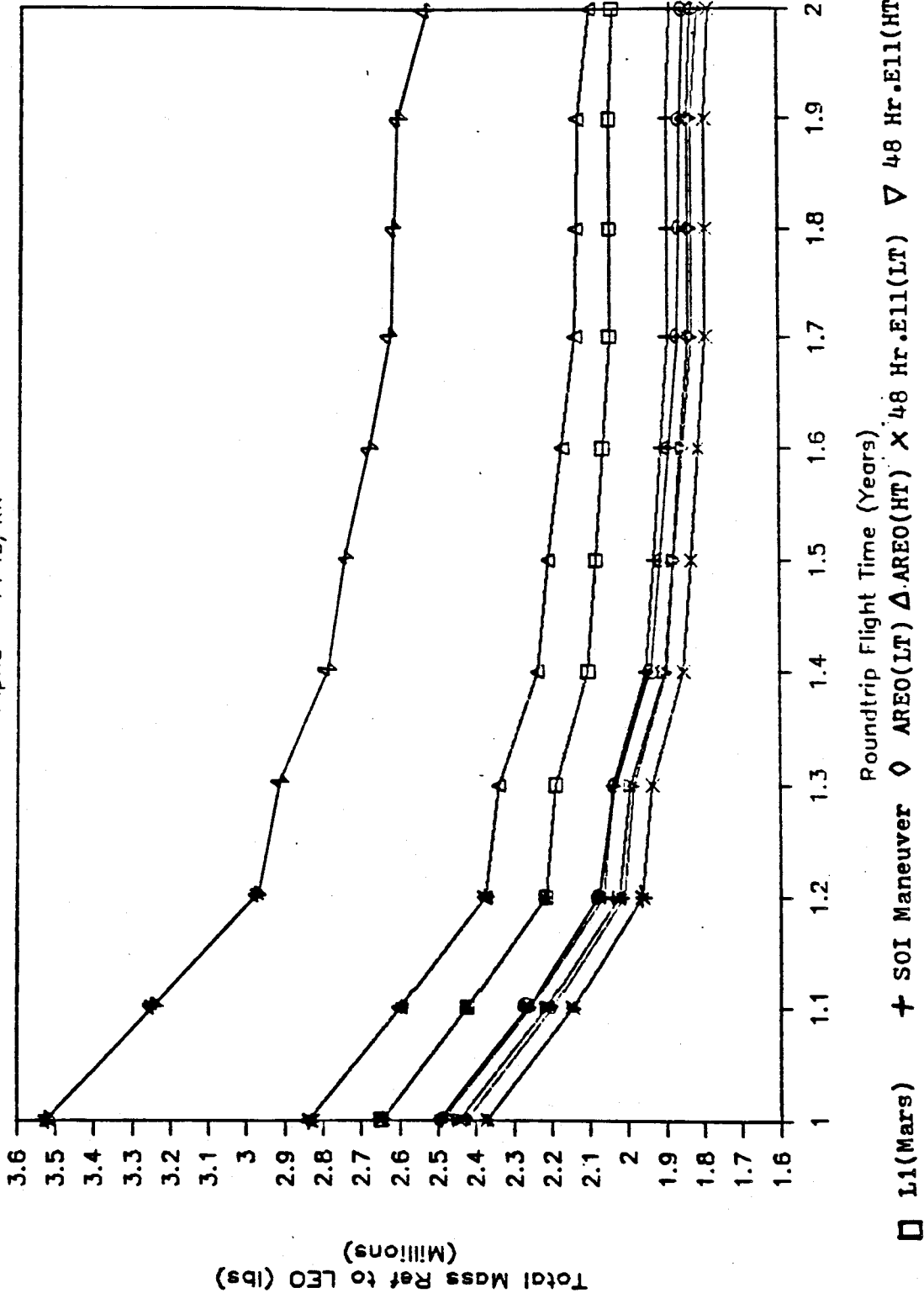


* Power level raised from 5 MW to 10 MW

Figure 3.2.4B

Mass vs. Roundtrip Flight Time

Alpha = 11 lb/kW

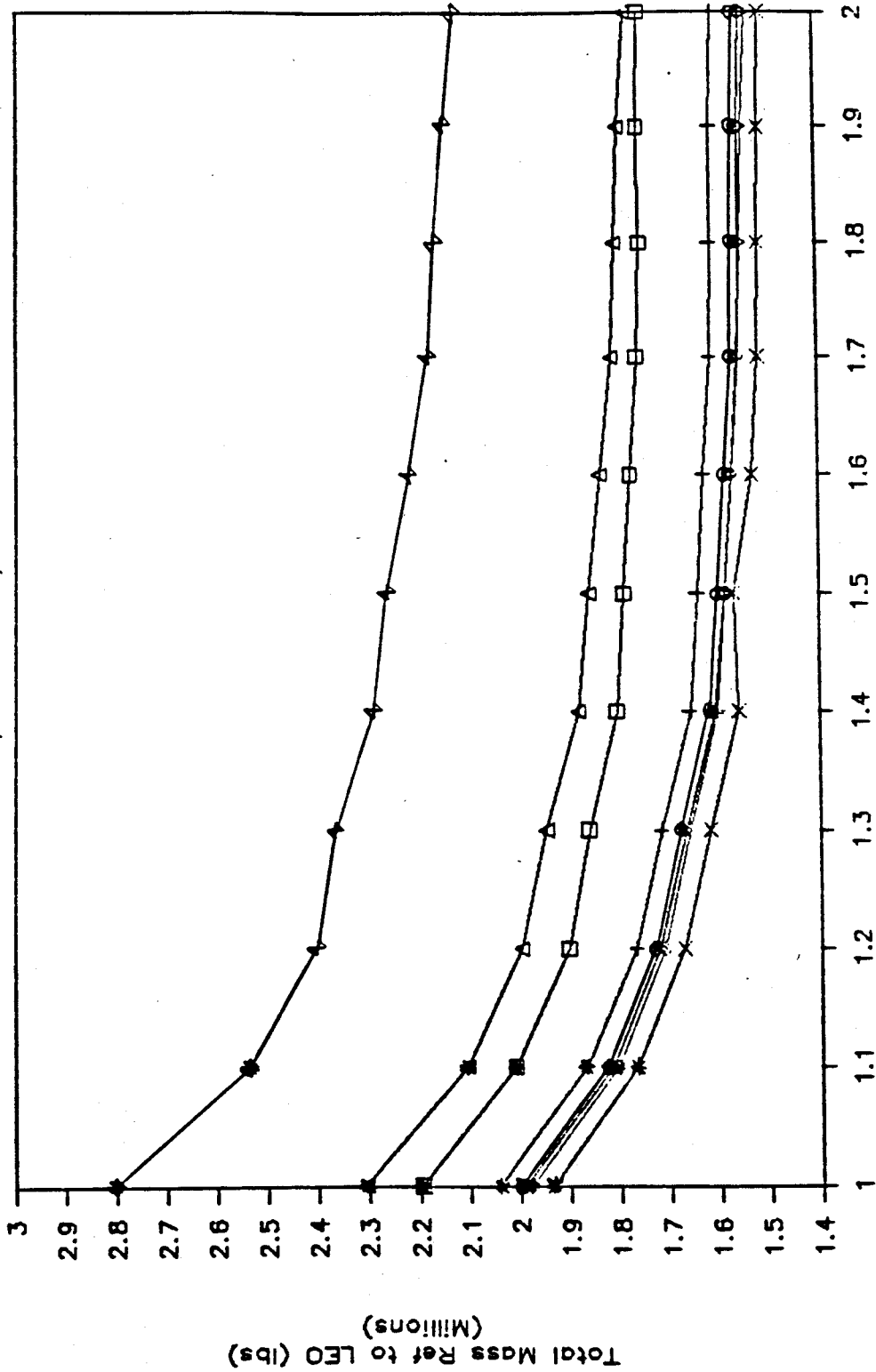


* Power level varied from 5 MW to 10 MW

Figure 3.2.4C

Mass vs. Roundtrip Flight Time

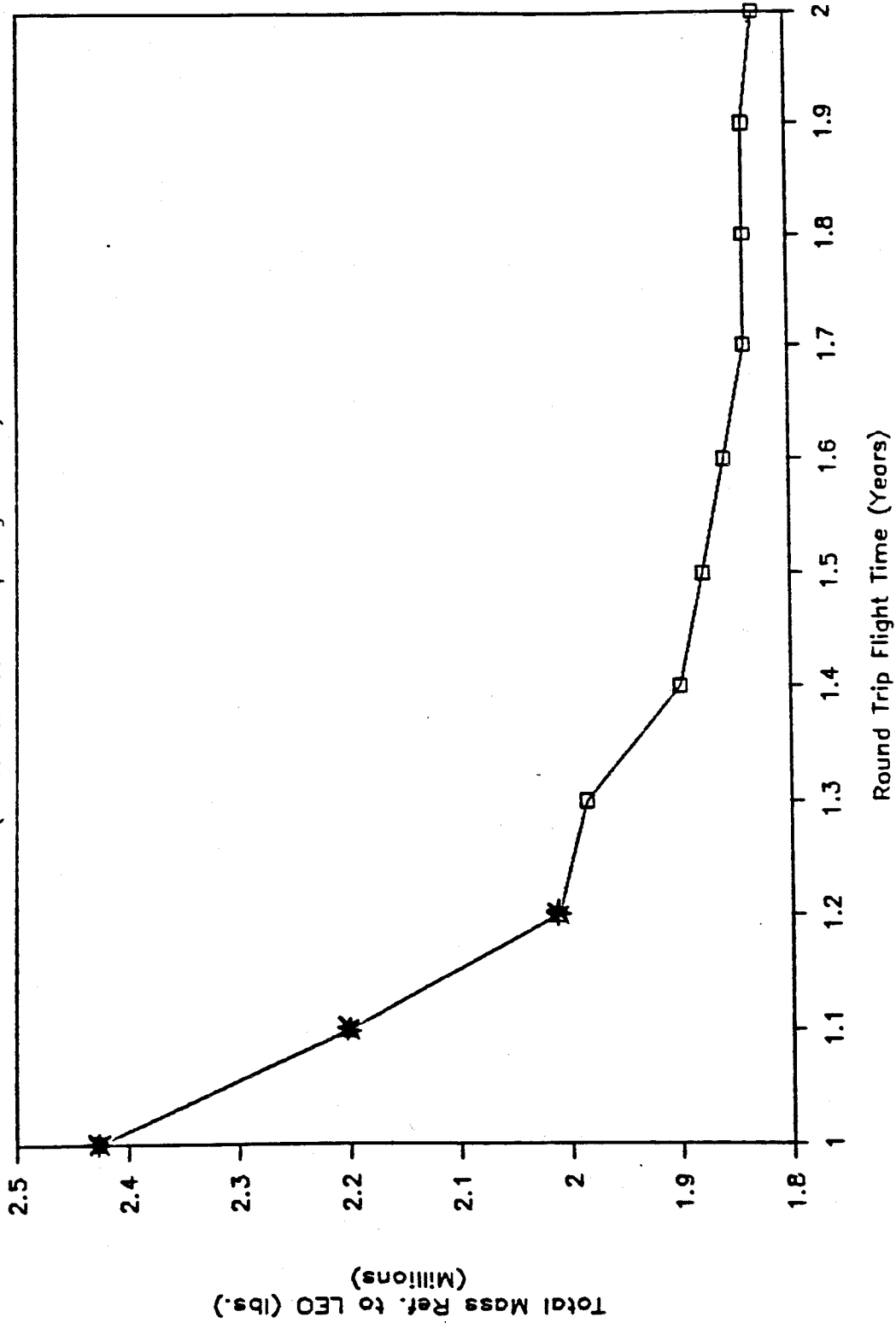
Alpha = 2.2 lb/kW



□ L1(Mars) + SOI Maneuver ◊ AREO(LT) Δ AREO(HT) X 48 Hr. E11(LT) ▽ 48 Hr. E11(HT)
 ○ LMO(LT) ◄ LMO(HT)

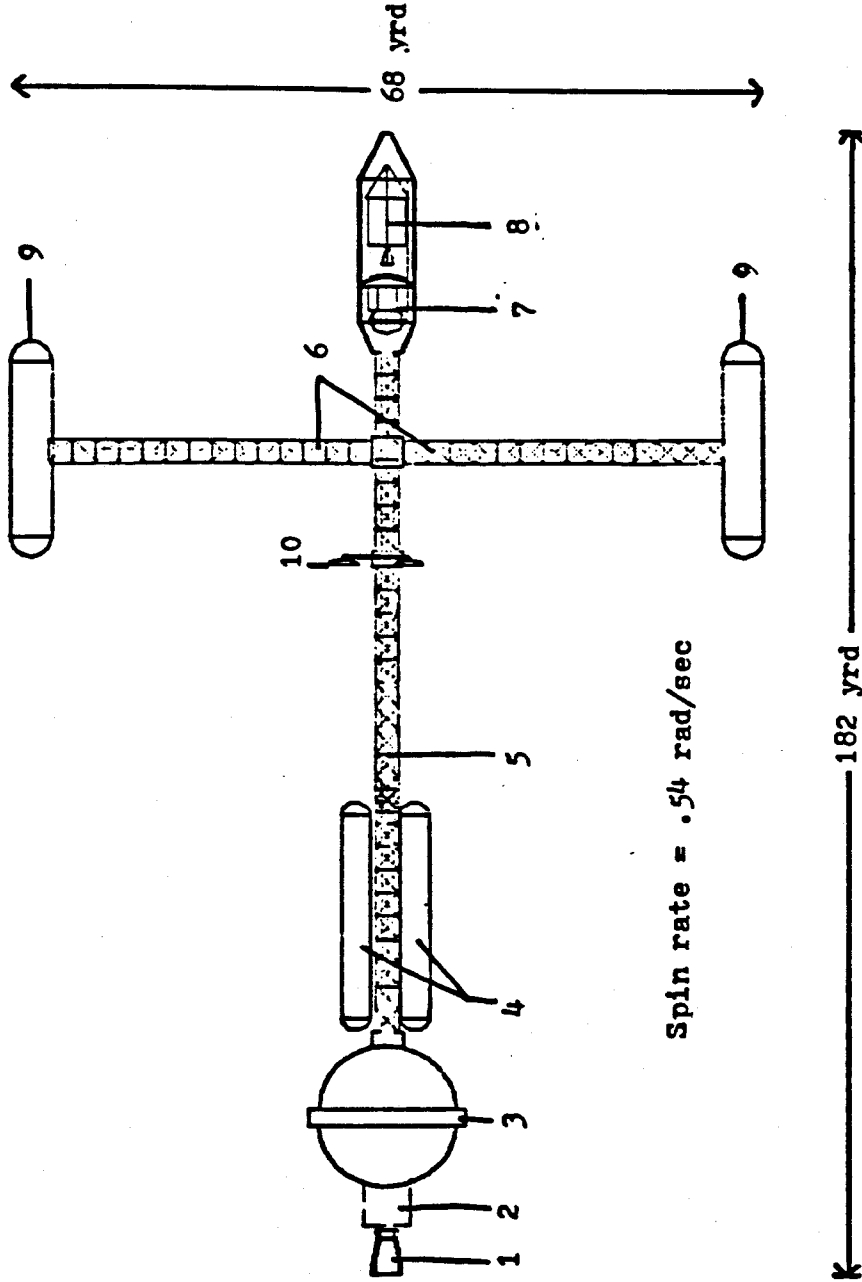
* Power level raised from 5 MW to 10 MW

Figure 3.3A
 Baseline Mission Concept
 (Mass vs. Round Trip Flight Time)



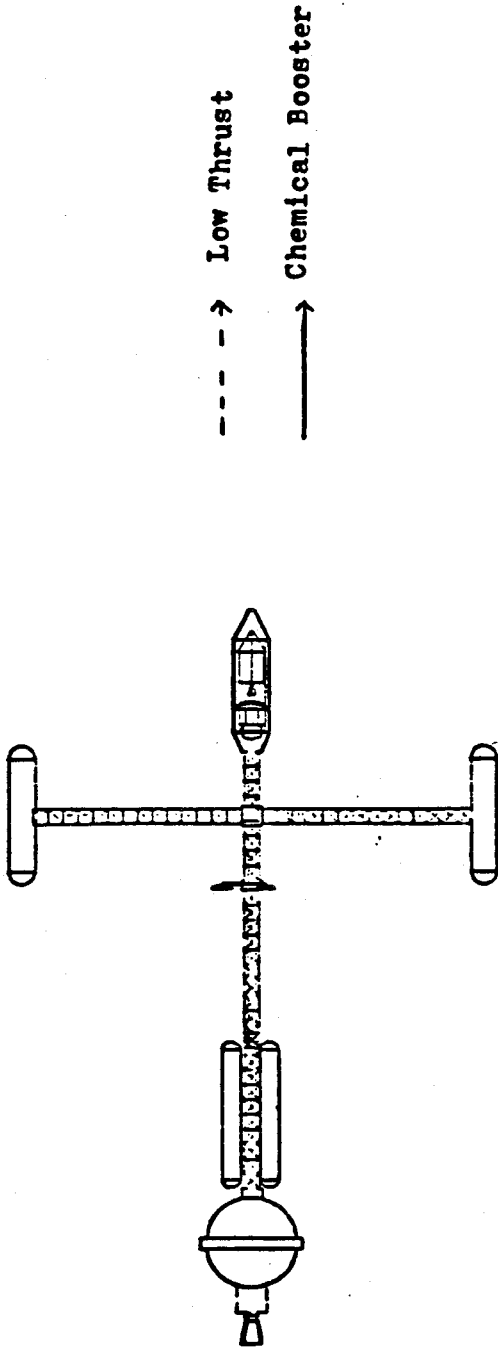
* Power level raised from 5 MW to 10 MW

Figure 3.3B
 Baseline Mission Concept
 (Heliocentric Flight Configuration)



#	Item
1	Thruster subsystem
2	Reactor/Shielding
3	Radiator
4	Propellant tanks
5	Primary truss work
6	Gravity structure
7	Earth Return Vehicle
8	Mars Excursion Module
9	Habitat Module
10	Antenna

Figure 3.30
Baseline Mission Concept
(Mission Phases)



Baseline Vehicle

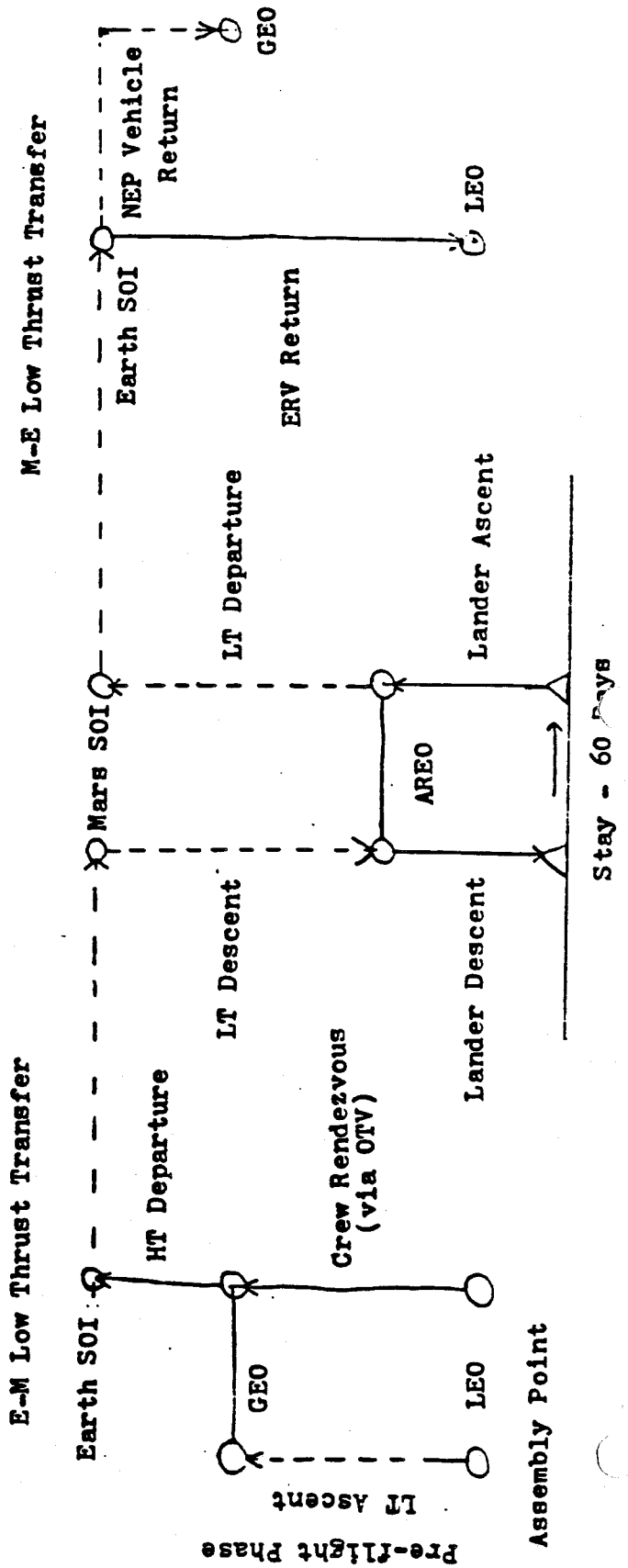
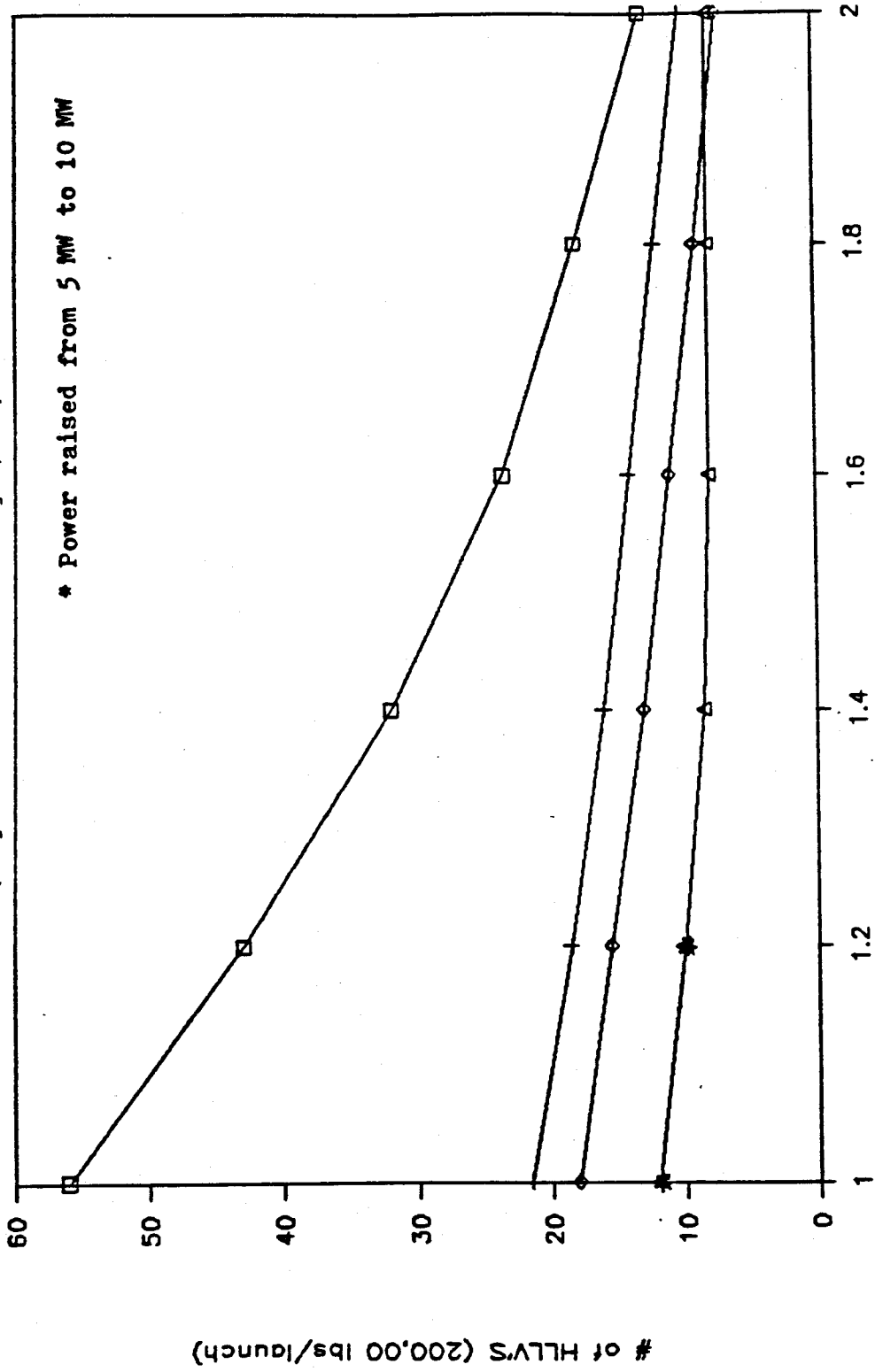


Figure 3.4.2A

Mars Mission Performance

(Single Mission: No Aerobraking Option)

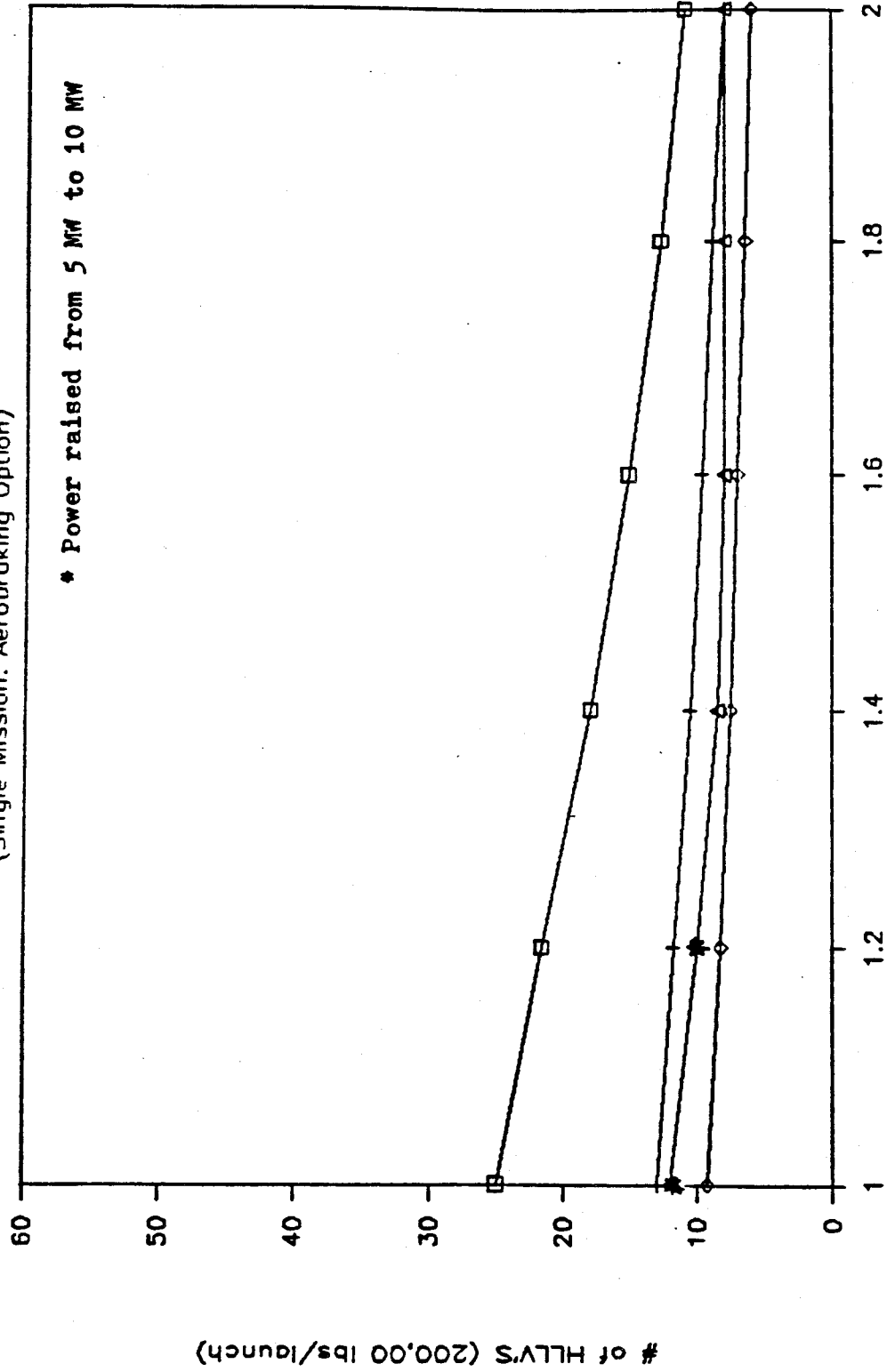


□ ALL UP(CHEM)
 + SPLIT OPTION(CHEM)
 ◇ SPLIT OPTION(NEP/Chem)
 △ Baseline Vehicle

● Reference 18

Figure 3.4.2B Mars Mission Performance

(Single Mission: Aerobraking Option)



□ ALL UP(CHEM)
 + SPLIT OPTION(CHEM/CHEM)
 ◇ SPLIT OPTION(NEP/CHEM)
 △ Baseline Vehicle

Figure 3.4.3A

Mars Mission Performance

(1.0 Year Mission: 3 Mission Cycles)

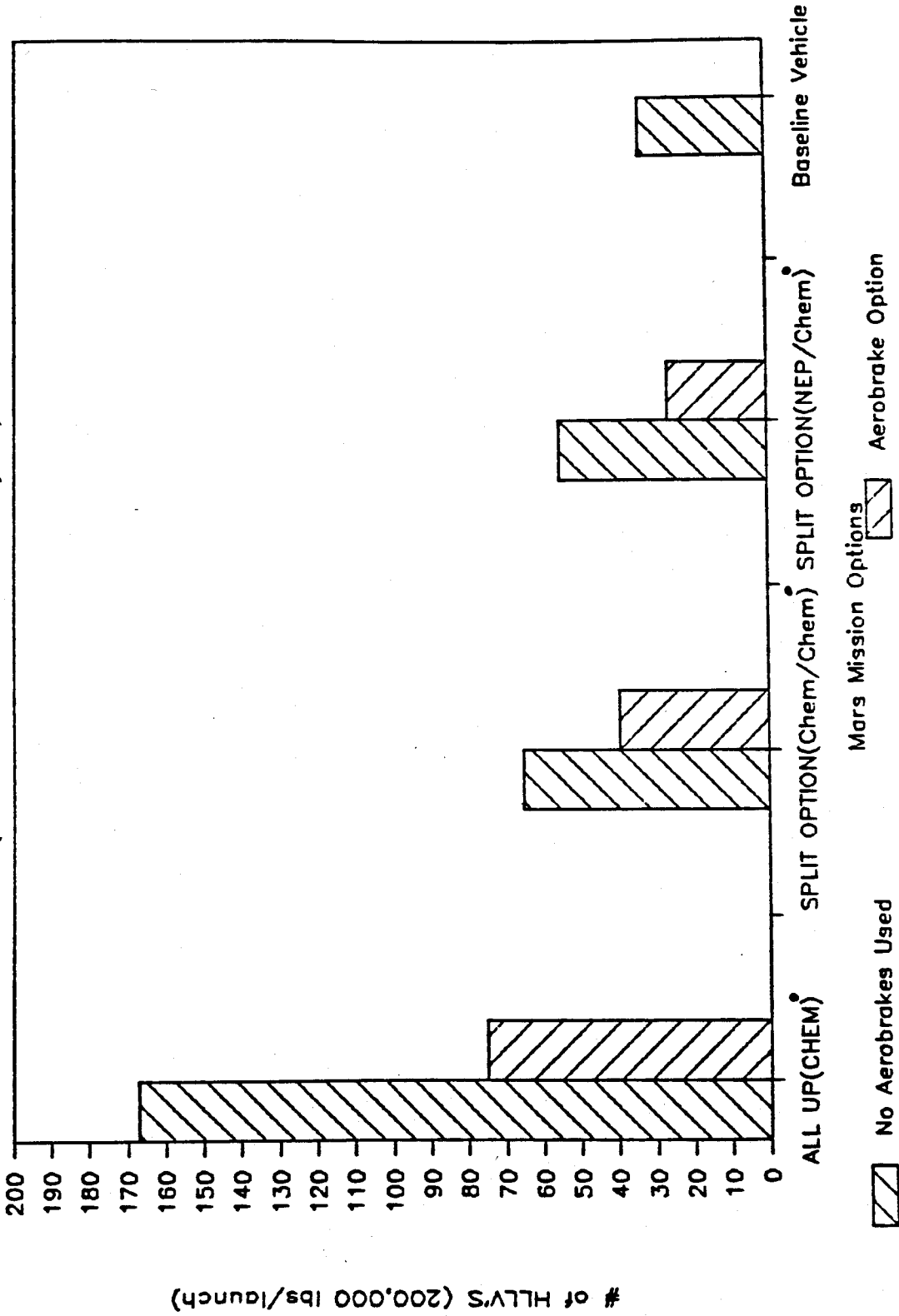
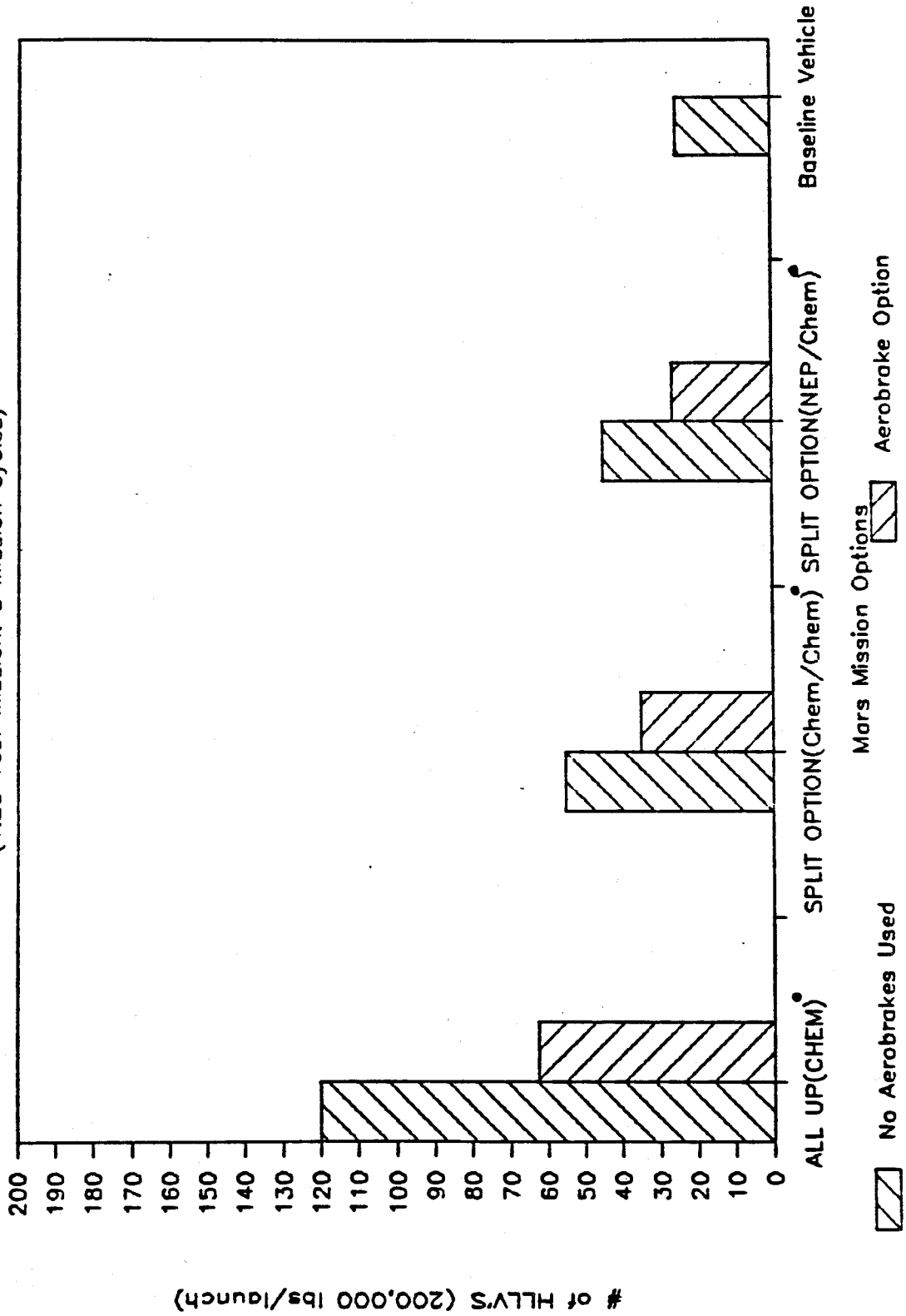


Figure 3.4.3B
Mars Mission Performance
 (1.25 Year Mission: 3 Mission Cycles)



• Reference 18

Figure 3.4.3C
 Mars Mission Performance
 (2.0 Year Mission: 3 Mission Cycles)

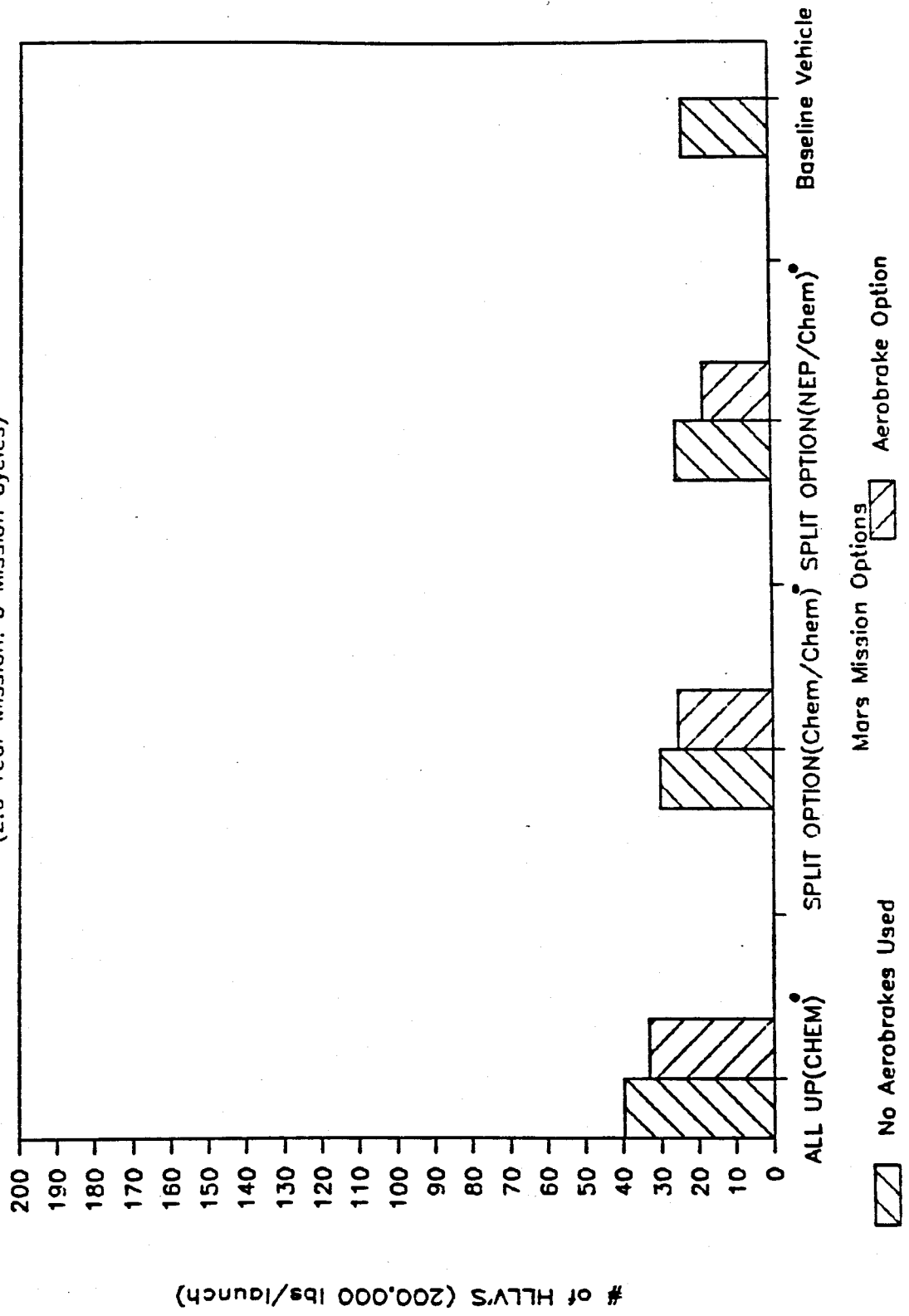
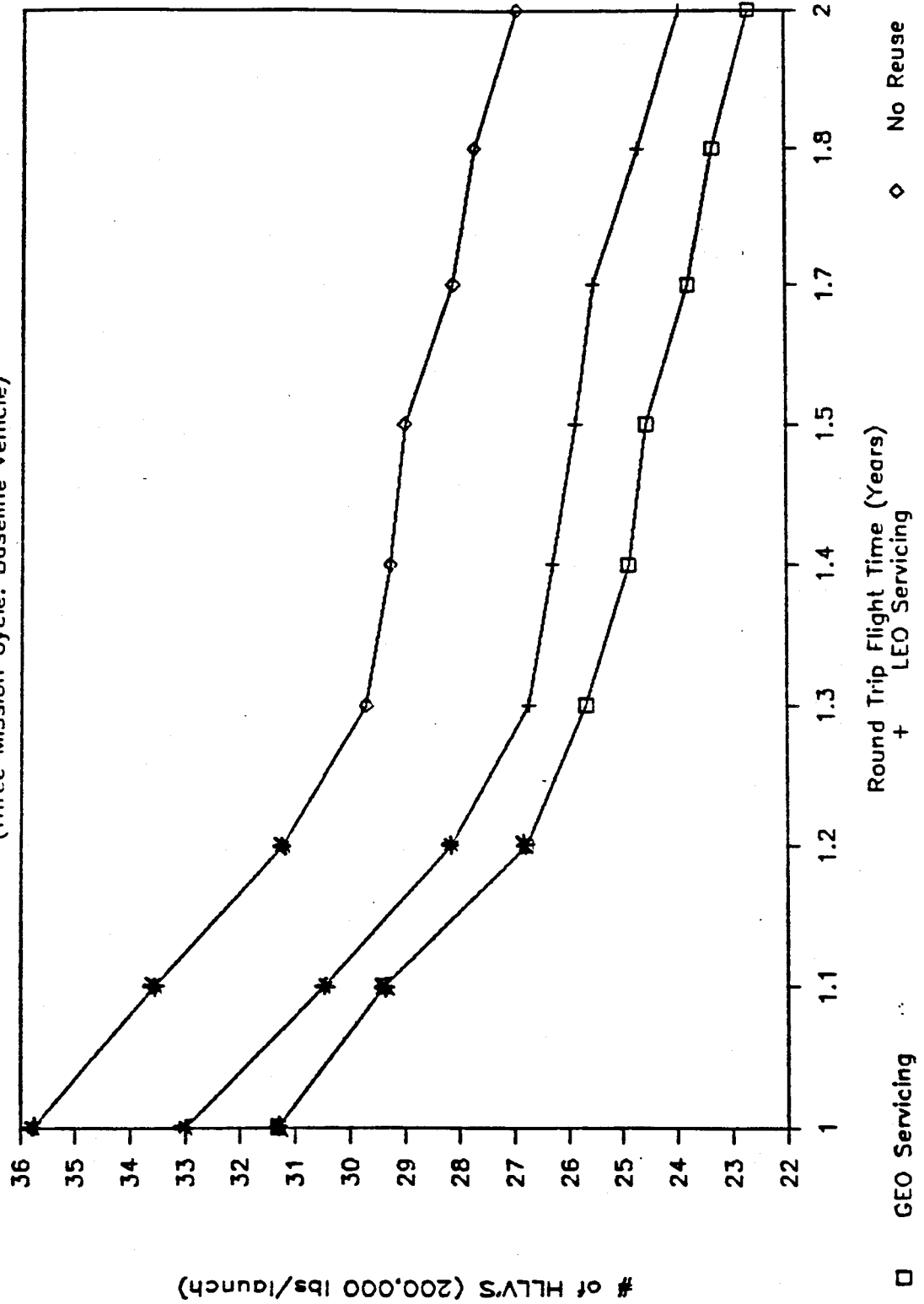
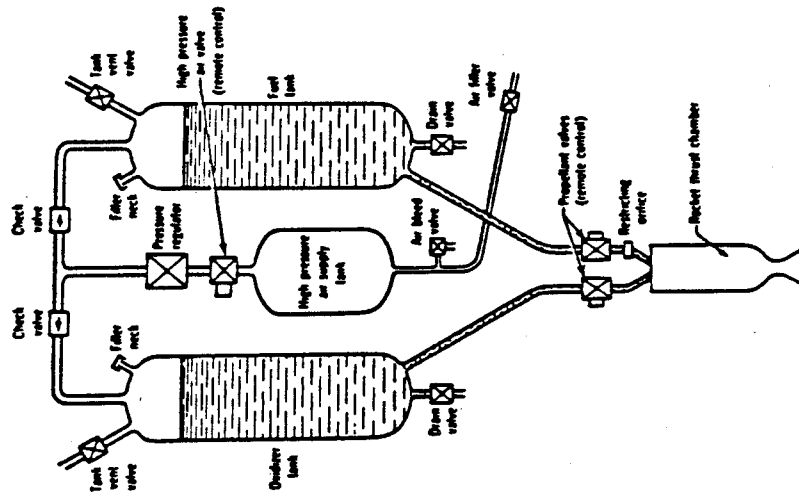


Figure AP3.1
 Multi-use Options
 (Three Mission Cycle: Baseline Vehicle)



* Power raised from 5 MW to 10 MW

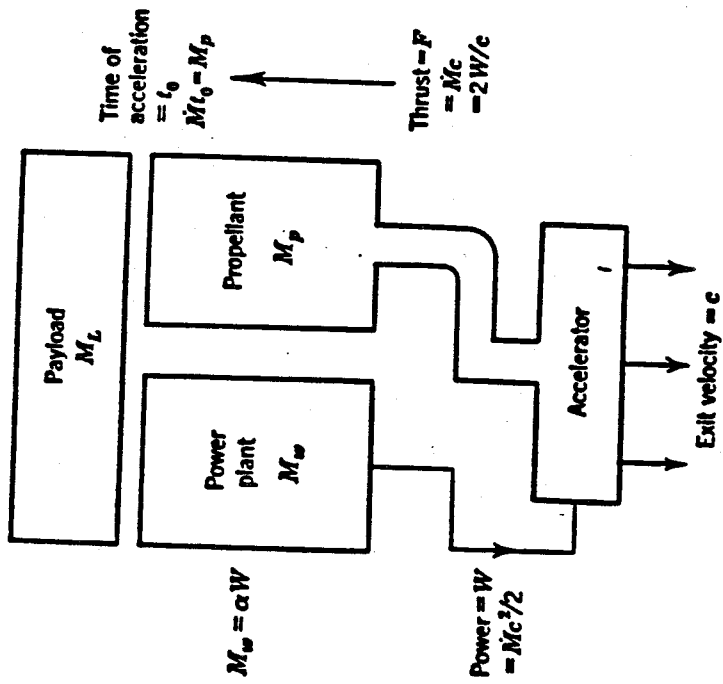
Figure AP4.1
Chemical Propulsion System Diagram



Schematic diagram of a liquid propellant rocket with a gas pressure feed system.

● Reference 3

Figure AP4.2
Electric Propulsion System Diagram

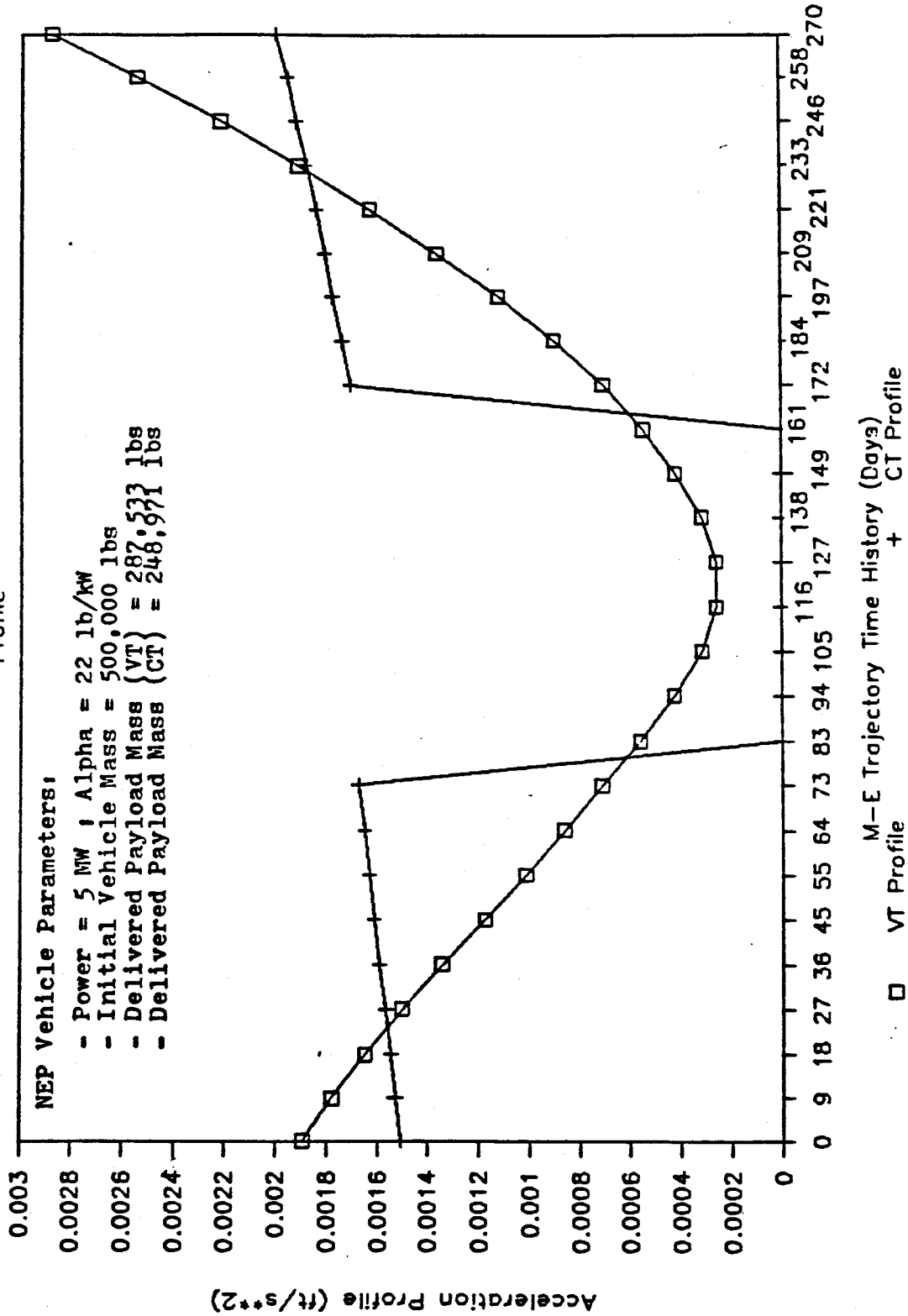


Total mass = $M_0 = M_L + M_p + M_{wp}$

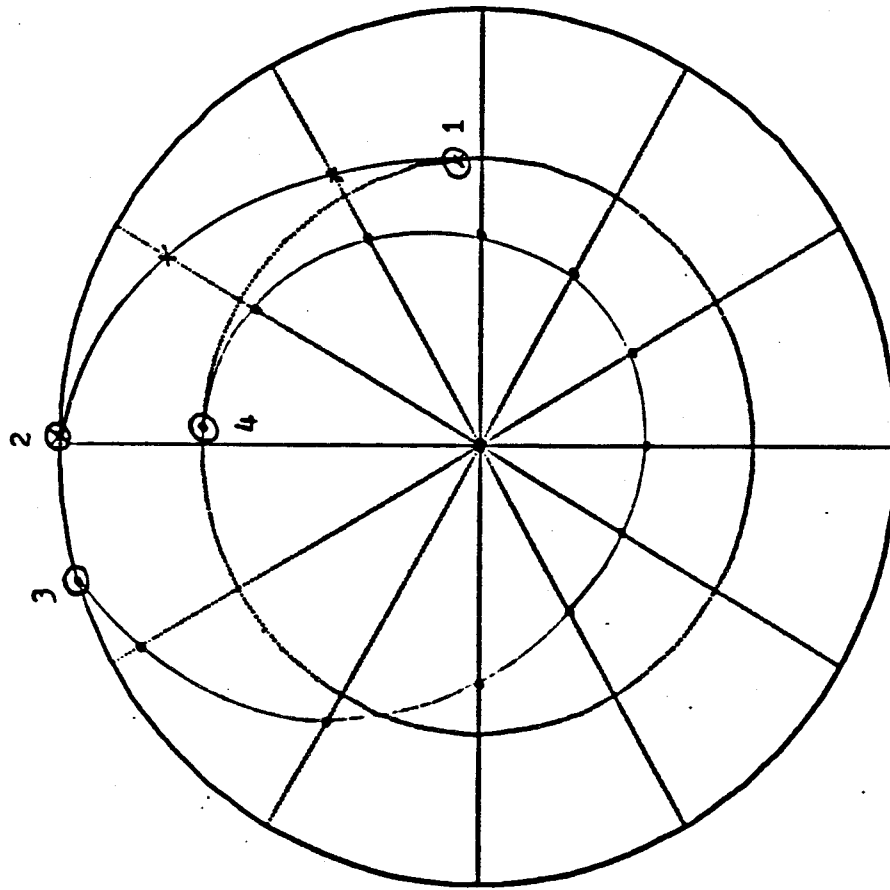
Functional diagram of the separately powered rocket.

Variable Thrust vs. Constrained Thrust Profile

Figure AP4.3



Supplemental Figure IA
 Sample Low Thrust Interplanetary Trajectory Profiles
 (Baseline Vehicle)

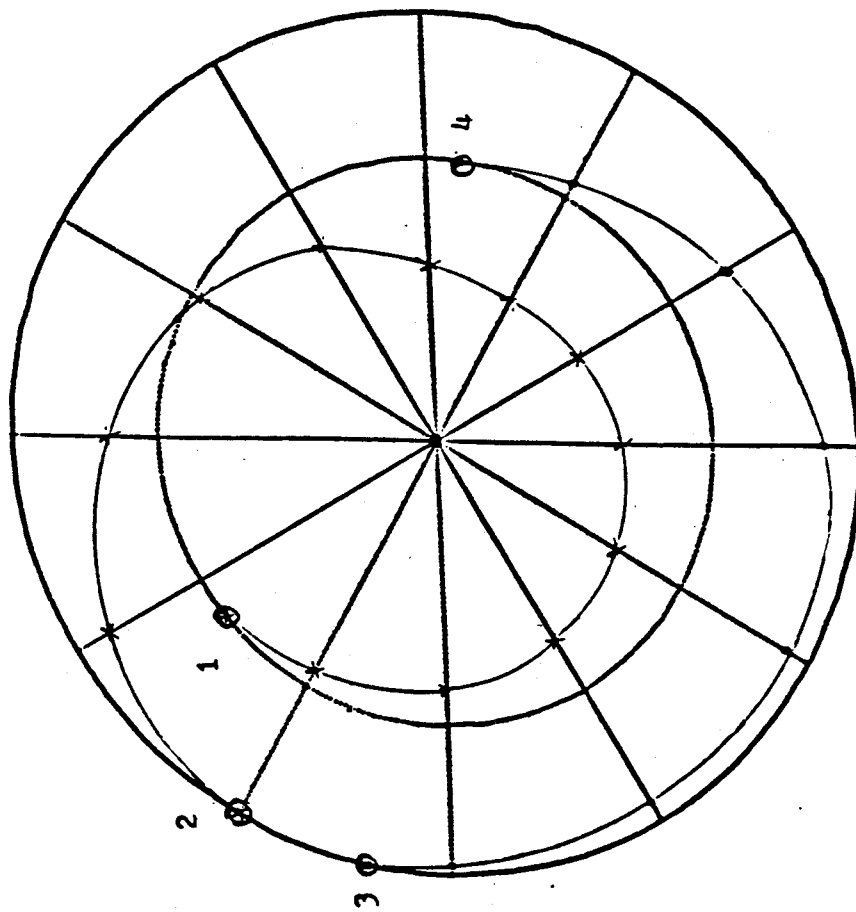


- 1 Earth Departure
- 2 Mars Arrival
- 3 Mars Departure
- 4 Earth Arrival

1.2 Year Round Trip Mission

E-M LEG : 153 Days
 M-E LEG : 225 Days
 Stay Time : 60 Days

Supplemental Figure 1B
 Sample Low Thrust Interplanetary Trajectory Profiles
 (Baseline Vehicle)



- 1 Earth Departure
- 2 Mars Arrival
- 3 Mars Departure
- 4 Earth Arrival

E-M LEG : 360 Days
 M-E LEG : 270 Days
 Stay Time : 60 Days

1.89 Year Round Trip Mission

APPENDIX 1

SPHERE OF INFLUENCE FLIGHT MANEUVER CALCULATION

The following outlines the procedure used to calculate the fuel requirements for the SOI Flight Maneuver. Part I outlines general geometric and astrodynamic constraints. Part II outlines the sizing and fuel calculation relationships utilized.

PART I: GEOMETRIC AND ASTRODYNAMIC CONSTRAINTS

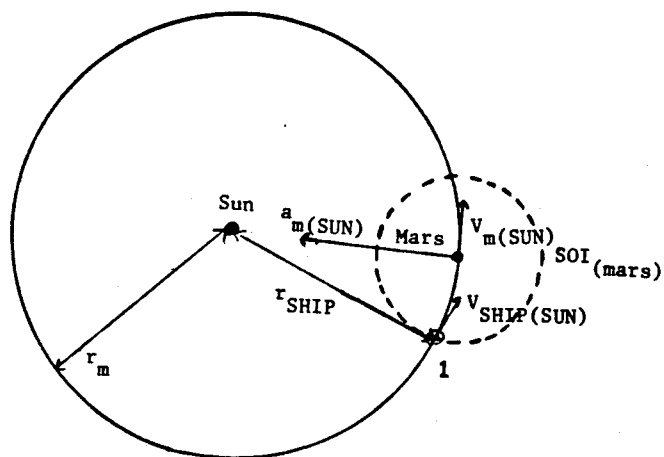


Figure 1: DIAGRAM OF MARS IN ORBIT ABOUT THE SUN

where:

- $a_m(SUN)$ = Acceleration vector of Mars
- $V_m(SUN)$ = Velocity of Mars with respect to the sun
- $SOI(MARS)$ = Mars sphere of influence
- r_m = (Mars-Sun) separation distance
- $V_{SHIP(SUN)}$ = Velocity of ship at point 1 with respect to the sun
- r_{SHIP} = (Ship-Sun) separation distance.

Basic Problem: Determine the thrusting profile required for a spacecraft to maintain itself at point 1 for a time t .

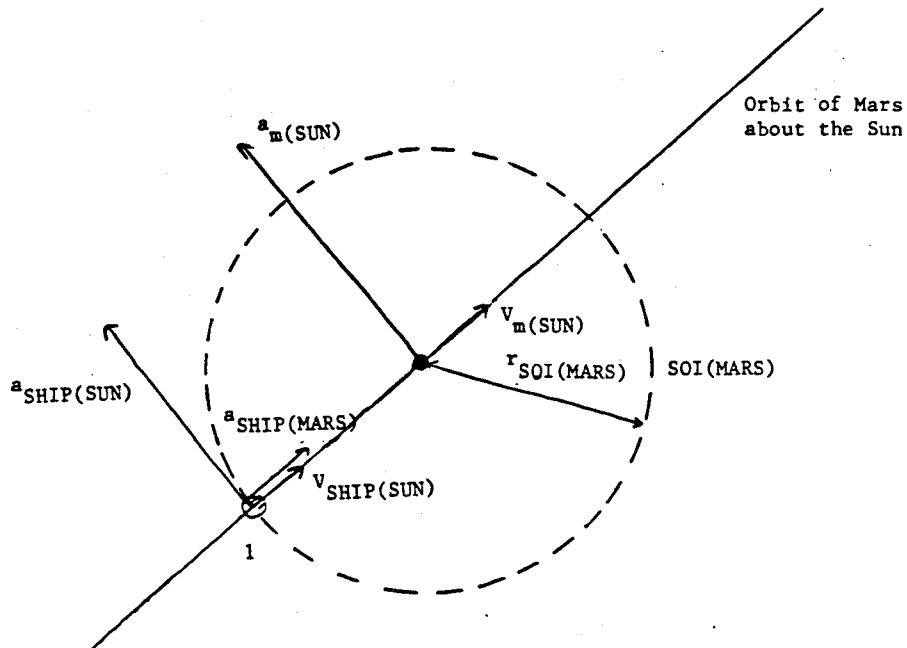


Figure 2: GENERAL GEOMETRY OF SOI FLIGHT MANEUVER

Note: a_{SHIP}

(MARS) - Perturbing acceleration of Mars on spacecraft at point 1.

To initiate the geometry suggested in Figures 1 and 2, a vehicle must arrive at point 1 such that its distance and velocity relative to the Sun match those of Mars (i.e., $r_{SHIP} = r_{Mars}$ and $v_{SHIP} = v_{Mars}$). Relative to Mars this would imply:

$$r_{SHIP} = r_{SOI_M}$$

$$v_{SHIP} = 0$$

where:

r_{SOI_M} = Radius of the Martian sphere of influence.

v_{SHIP} = Hyperbolic entry velocity of the spacecraft at the Martian sphere of influence.

If the gravitational acceleration on the spacecraft due to Mars (a_{SHIP_MARS}) were identically zero at point 1, the ship would require no additional maneuvering fuel to maintain the geometry suggested in Figures 1 and 2.

However, this is not the case. Recalling that the sphere influence is only a convenient fiction used primarily in connection with Patched Conic trajectory work (Reference 22), we must take into account the residual acceleration at point 1 on the vehicle if it is to remain there for any amount of time. This residual acceleration is given simply by Newton's Second Law with Newton's Universal Law of Gravitation:

$$a_{SHIP} = \frac{U_{Mars}}{r^{**2}}$$

where:

a_{SHIP} = The acceleration of the ship due to Mars

U_{Mars} = Gravitational Parameter (MARS)

r = Distance between bodies.

This relationship is used to calculate the force of attraction at point one on the vehicle due to Mars.

At point one the magnitude of this residual acceleration was found to be equal to:

$a_{SHIP} = 4.2316E-4 \text{ ft/sec}^{**2}$

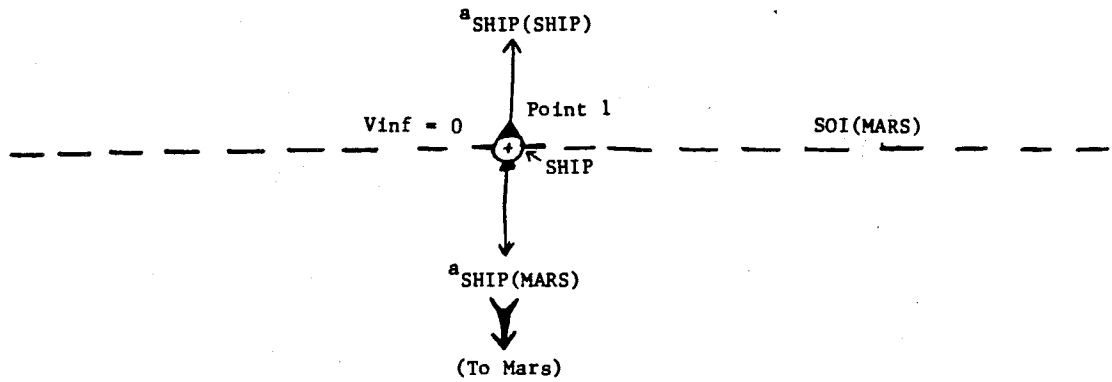


Figure 3: ACCELERATION PROFILE

Thus, to maintain the orbital geometry suggested by Figures 1 and 2, the spacecraft must maintain a constant thrust profile such that the induced acceleration balances the perturbing force produced by Mars. As shown in Figure 3, this acceleration must be equal in magnitude but opposite in direction.

PART II: FEASIBILITY AND PROPELLANT REQUIREMENTS

A. Feasibility

In Part I the acceleration requirement to maintain a spacecraft at point 1 was calculated. The low thrust propulsion system utilized in the thesis work can provide continuous acceleration levels on the order of $2.6E-3 \text{ FT/SEC}^2$ (based on heliocentric transfer data from QUICKTOP). Thus, the minimum acceleration level required to perform the stationkeeping maneuver suggested in Part I can be provided by the NEP system. The SOI Flight Maneuver was therefore deemed feasible from a vehicle standpoint.

B. Propellant Calculation

Once the feasibility of the concept was established, a calculation of the fuel requirements to perform such an operation was in order.

REQUIREMENT: SHIP MUST PROVIDE A CONSTANT ACCELERATION OF $4.2316E-4$ ft/sec**2 FOR TIME T.

CALCULATION:

When calculating the propellant required to perform the maneuver, realize that the net work done by the low thrust rocket is the same as if it were placed in free field space and accelerated at $4.2316E-4$ ft/sec**2 from t_1 to t_2 . From Reference 31 the rocket equation is seen as directly applicable.

$$V_c = V_e * \ln(M_i/M_f) \quad \text{eqn. (a)}$$

Where:

V_c = Characteristic free field velocity

V_e = Propellant exhaust velocity

M_i = Initial mass of spacecraft (t_1)

M_f = Final mass of spacecraft (t_2).

Also note that since the acceleration applied from t_1 to t_2 is constant, V_c may also be found using the simple kinematic relationship:

$$V_c = a_{SHIP} * (t_2 - t_1). \quad \text{eqn. (b)}$$

where:

a_{SHIP} = acceleration of ship (const.)

By defining $(t_2 - t_1)$ as the stay time at Mars V_c was found to equal:

$V_c = 2,193.68 \text{ ft/sec.}$

In general, by substituting in the relationship:

$$M_{SHIP_{t2}} = M_{SHIP_{t1}} - M_{SOI_{FUEL}} \quad \text{eqn. (c)}$$

equation (a) becomes:

$$M_{SOI_{FUEL}} = M_{SHIP_{t2}} * (e^{Vc/Ve} - 1.0). \quad \text{eqn. (d)}$$

In the thesis work V_e was assumed to be equal to 96.96 ft/sec**2. With this, equation (d) reduces to:

$$M_{SOI_{FUEL}} = .02298 * M_{SHIP_{t2}} \quad \text{eqn. (e)}$$

Thus, for a given vehicle mass at time t_2 (Mars departure), the low thrust fuel requirements for the stationkeeping maneuver can be calculated using equation (e).

APPENDIX 2

TRANSFER VEHICLE SIZING

During the early phases of the analysis process, it became apparent that the interplanetary low-thrust vehicle would require two transfer vehicles to be carried on board to provide additional at planet phase transport capability.

The first vehicle is used at Mars to taxi the landing party from the interplanetary vehicle to the surface of the planet and back. The second is needed during earth return (EOM) to safely deliver the crew to low earth orbit while the interplanetary vehicle descends to a geostationary earth orbit.

The following appendix outlines mass and time of flight information for these taxi vehicles. Since several different Mars arrival orbits were studied, a different Mars transfer vehicle was sized for each orbital node. Due to fast transfer constraints (on the order of hours or days), only chemical propulsion or aerobrake schemes were investigated. The total mass of such vehicles is primarily driven by the total energy or delta-v requirements imposed on them by transfer and propulsion system parameters.

Part I of this appendix will draw on Reference 22 and outlines the theoretical development necessary to properly quantify the mass and time of flight requirements for the transfer vehicles. In Part II, the calculated mass and TOF data for each vehicle will be presented.

PART 1: DISCUSSION OF GENERAL SIZING PROCEDURE

A. Trajectory Calculations

In general, a straight forward calculation of transfer requirements based on Hohmann Transfers (see Figure 1) was used to obtain trajectory energy requirements. Only general propellant and structural estimates were desired for the transfer vehicles. By using Hohmann transfers, a relatively simple but reasonable assessment of orbit transfer demands could be obtained (Reference 22). For transfer to and from planetary SOI's, the estimated Hohmann flight time for a particular transfer was deemed to be unacceptably long. In these cases, hyperbolic approach and departure trajectories were utilized (see Figure 2). Reference 22 defines and outlines in detail these two transfer modes. However, the following has been included as a short review.

1. The Hohmann Transfer Between to Coplanar Circular Orbits.

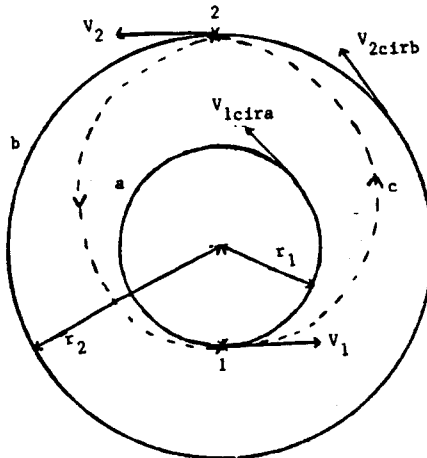


Figure 1

Transfer from orbit A to orbit B is accomplished by traveling along the doubly-tangent transfer ellipse designated as orbit C.

The velocity any object has while traveling in a circular orbit about a central body at a radius r is given by:

$$V_{\text{cir}} = \frac{U_{\text{BODY}}}{r}$$

where: U_{BODY} = the gravitational parameter of the central body. The energy of the transfer orbit C is given by:

$$E = \frac{-U_{\text{BODY}}}{(2 * a)}$$

where: $a = \frac{r_1 + r_2}{2}$

Thus: $E = \frac{-U_{\text{BODY}}}{r_1 + r_2}$.

For this ellipse an objects velocity at point n (with $n = 1, 2$) is given by:

$$V_n = \sqrt{2 \left(\frac{U_{\text{BODY}}}{r/n} + E \right)}$$

For a spacecraft to transfer from orbit A to orbit C , a velocity change at point 1 given by is required.

$$\text{DELTA}V_1 = \|V_{\text{cir}1} - V_1\|.$$

After traveling on orbit C to point 2 the spacecraft can then make another velocity change to transfer to orbit B .

This velocity change is equal to:

$$\text{DELTA}V_2 = \|V_2 - V_{\text{cir}2}\|.$$

Thus, the total velocity change that the spacecraft must impart to perform a Hohmann transfer from orbit A to orbit B is:

$$\text{DELTA}V(\text{tot}) = \text{DELTA}V_1 + \text{DELTA}V_2 .$$

The total transfer time is given by:

$$TOF_{12} = \frac{a^{*3}}{U^{BODY}}$$

To travel from orbit B to orbit A a similar analysis is performed, except the indices and orbital operations would follow a reverse path.

2. Hyperbolic Approach/Departure Trajectory (2-D)

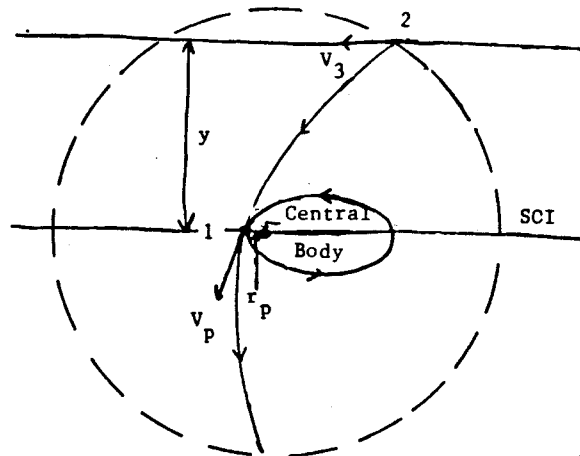


Figure 2

Upon entry into a planetary SOI, a spacecraft's velocity relative to the planet's motion is designated by V_{inf} or V_e .

For a hyperbolic approach trajectory with a specified closest approach distance R_p , the velocity at that point V_p is given by:

$$V_p = \sqrt{V_3^2 + \frac{2 * U}{R_p}}$$

Like the Hohmann analysis, at point 1 a spacecraft must impart a velocity change if it is to assume some type of ellipse about the planet.

Thus, for a desired elliptical orbit with velocity V_{ell} at point 1 the total velocity change is given by:

$$\Delta(TOT) = \|V_p - V_{ell}\|$$

The travel angle θ is given by:

$$\theta = \sin^{-1} \left(\frac{Y}{R_p} \right)$$

where: $Y = \frac{R_p}{V_3} * V_p$

For small θ an approximate estimate for the TOF from point 2 to point 1 is given by:

$$TOF_{21} = \frac{R_{soi}}{V_3}$$

The same mathematical development can also be applied for transfer from point 1 to the SOI if a hyperbolic departure trajectory is desired.

B. Rocket Equation Mass Sizing

With the above velocity requirement information and specified propulsion system parameters, the mass of a transfer taxi can be determined.

Mass sizing was done by utilizing the Rocket Equation (Reference 31) which can be stated as follows:

$$\text{DELTA V} = (V_e) * \ln(\text{MR})$$

where:

DELTA V - The velocity change required by the spacecraft.

V_e - Exhaust velocity given by $V_e = I_{sp} * g_e$

where:

I_{sp} - Specific impulse (sec) [Earth].

g_e - Acceleration at the surface of the Earth

MR - Ratio of M_o/M_{final}

with:

M_o - Initial mass of vehicle (before ignition)

M_{final} - Final mass (after ignition).

In the case of a single stage rocket (see Figure 3), the mass of the structure and fuel required to propel the spacecraft from a velocity V_1 at time t_1 to velocity V_2 at time t_2 is given by:

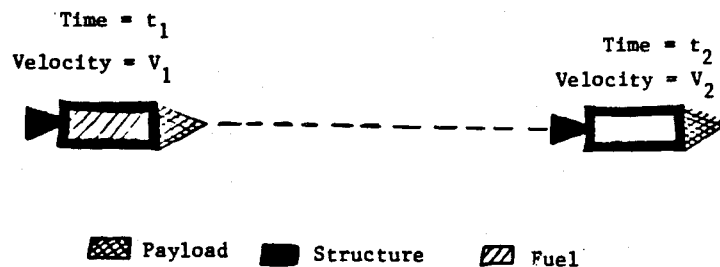


Figure 3

$$M_{\text{fuel}} = \frac{M_{\text{P/L}} [e^{\frac{\Delta V}{v_e}} - 1]}{1 + M_f (1 - e^{-\Delta V/v_e})}$$

and $M_{\text{str}} = M_f * M_{\text{fuel}}$

where: M_f = ratio of structural mass to fuel mass for the rocket stage. $M_f = .1$ assumed in this study.

Thus, the total mass of the rocket at t_1 is given by:

$$M_{\text{tot}} = M_{\text{pl}} + M_{\text{str}} + M_{\text{fuel}}$$

For multiple stage rockets, the analysis is similar. However, for each stage a unique accounting must be made as to what constitutes "payload" for that stage, and what is its specific delta V requirement. The "payload" must include any subsequent stages including fuel and structure for subsequent maneuvers. In this way, by starting with the last delta V maneuver and working backward, an estimate of fuel and structure for each stage can be determined. With each stage properly sized, the total mass estimate for the vehicle results.

C. Aerobrake Sizing Procedure

The following section outlines the procedure used to estimate the potential impact that an aerobrake system could have on the mass requirements of each transfer vehicle.

PROCEDURE:

- 1) Calculate mass requirements for an all propulsive transfer vehicle.
- 2) Use the following sizing rule to calculate the aerobrake mass:

$$M_{\text{ab}} = S_{\text{Fab}} * (M_{\text{prop}} + M_{\text{pl}}) \quad \text{eqn. (C1)}$$

where: M_{ab} = Mass of the aerobrake

S_{Fab} = Sizing Factor (15 percent was assumed in this study)

M_{prop} = Mass of fuel plus structure for the all propulsive stages being replaced by the aerobrake

$M_{P/L}$ = Mass of the "payload" to be carried during AB maneuver (includes fuel, structure, etc., and subsequent impulsive maneuvers)

- 3) Resize any stages previously sized in step 1 with M_{ab} replacing M_{prop} to determine new overall transfer vehicle mass.

Note: Equation (C1) represents a rough scaling relationship between a propulsive stage, a delivered payload mass, and the associated aerobrake stage used to replace the propulsive segment. The 15 percent scaling factor was obtained from Roy Young, SRS Technologies, Huntsville, Alabama.

PART 2: TRANSFER VEHICLE SIZING RESULTS

A. All propulsive calculations:

1. Earth Return Vehicle:

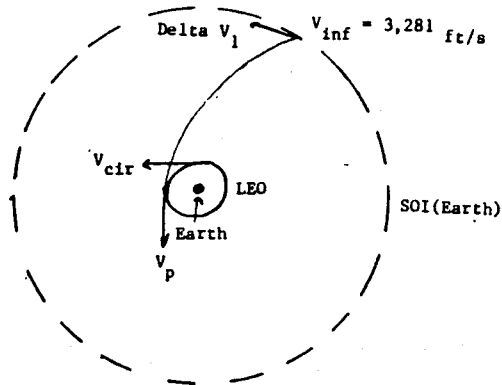


Figure 1

Figure 1 outlines the hyperbolic return trajectory followed by the earth return vehicle. Starting from the SOI of the Earth, the ERV travels to point 2 which is at an altitude of 270 NM from the Earth's

surface. At this point, a circulation maneuver is performed. Using the equations and techniques described in Part I, the following parameters were determined.

a) Trajectory Results

Earth entry velocity	V3 = 3,281 ft/s
Closest approach distance	Rp = 4,275 mi
Velocity at point 2 along hyperbola	Vp = 35,433 ft/s
Circular velocity at point 1	Vc = 24,964 ft/s
Total V requirement (from ERV)	Vtot = 12,106 ft/s

b) Mass Sizing

Payload capsule	Mpl = 31,177 lb
First Stage:	
Propellant mass	Mf1 = 10,011 lb
Structural mass	Mst1 = 1,003 lb
Subtotal Stage 1	11,014 lb
Second Stage:	
Propellant mass	Mf2 = 32,767 lb
Structure mass	Mst2 = 3,278 lb
Subtotal Stage 2	36,045 lb

Total Mass ERV (Chem)	Merv = 78,286 lb
-----------------------	------------------

2. Mars Excursion Module

Five separate vehicles were sized. One vehicle for each Mars arrival/departure node studied. These nodes include:

- o LMO

- o Areo
- o 48 hour Ell
- o SOI Mars
- o Ll Mars.

The following sections describe the trajectory and vehicle specifications. In this analysis, each vehicle was required to transfer to LMO before descending to the surface. Thus, for operations to and from planet's surface, a standard lander could be utilized.

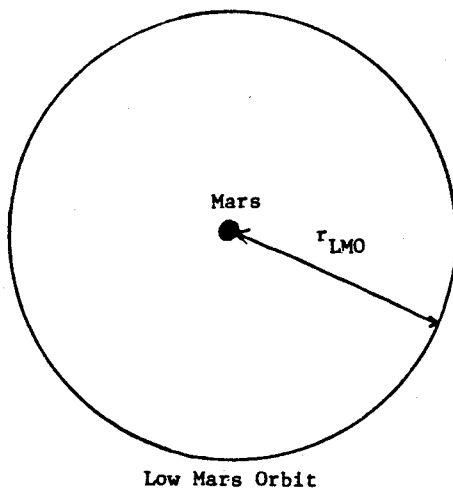


Figure 2

Orbit 1: (Low Mars orbit)

As mentioned above, the LMO orbit represents the staging orbit for the surface landing craft. Thus, the landing craft carried to this orbit by the main interplanetary spacecraft did not require any additional transfer stages. Reference 16 provided the LMO landing craft. Therefore, its mass manifest represents the MEM mass for the LMO node.

MEM (LMO):

Mass manifest (Reference 16: (Four person option)

1) Surface to LMO:

Ascent capsule	5,602 lb
Ascent stage II prop	6,876 lb
Ascent stage I prop	13,479 lb

2) LMO to surface

Descent stage	43,291 lb
Deorbit motor	4,200 lb

3) Thirty percent growth and contingency 22,046 lb

Total MEM (LMO)	95,503 lb
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For the following landers (see below) an orbital ascent transfer stage is left in LMO during surface operations. Thus, the surface ascent capsule must rendezvous with this stage upon return to LMO in order to continue its transfer back to the interplanetary vehicle.

Orbit 2: (Areosynchronous orbit)

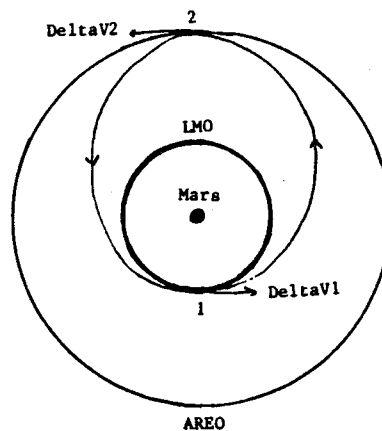


Figure 3

Figure 3 illustrates the MEM transfer requirements from aerosynchronous orbit. A Hohmann transfer to and from AREO and LMO was utilized.

a) Trajectory Results

AERO circular velocity	Vc = 4,757 ft/s
Velocity at point 2 ellipse	V2 = 2,687 ft/s
Delta V1	= 2,070 ft/s
Velocity at point 1 on ellipse	V1 = 14,108 ft/s
LMO circular velocity	Vc = 10,925 ft/s
Delta V2	= 3,182 ft/s
Total delta V	5,256 ft/s
Transfer time (one way)	.235 days

b) Mass Sizing

1) Ascent stage

Capsule mass	Mpl = 5,602 lb
Mass of propellant	Mfa = 2,337 lb
Mass of structure	Mst = 234 lb
Substage total (excluding payload)	Mtota = 2,571 lb

2) Descent stage

Payload	Mpl = 98,073 lb
Mass of propellant	Mfd = 41,433 lb
Mass of structure	Mst = 4,145 lb
Substage total (excluding payload)	Mtotd = 45,578 lb

Total Mass MEM (AREO)	143,651 lb
-----------------------	------------

Orbit 3: (48 hour ellipse)

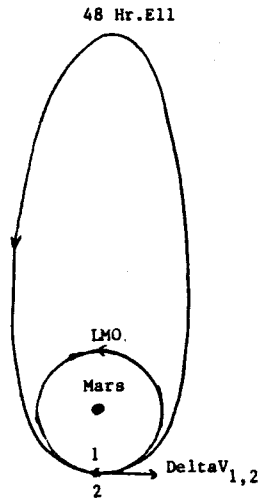


Figure 4

Figure 4 illustrates the MEM requirements from a 48 hour ellipse to LMO. One impulse is required for circularization or to re-establish orbit 3 upon ascent.

a) Trajectory Results

Velocity at point 1 on ellipse	$V_e = 15,092 \text{ ft/s}$
Circular velocity at LMO	$V_c = 10,925 \text{ ft/s}$
Delta V	$= 4,167 \text{ ft/s}$
Time for transfer (one way)	-

b) Mass Sizing

Ascent stage:

Capsule mass	$M_{pl} = 5,602 \text{ lb}$
Mass of propellant	$M_{fa} = 1,790 \text{ lb}$
Mass of structure	$M_{st} = 181 \text{ lb}$
Substage total (excluding payload)	$M_{tot} = 1,971 \text{ lb}$

Descent stage:

Payload	$M_{pl} = 97,474 \text{ lb}$
---------	------------------------------

Mass of propellant	Mf1 = 31,144 lb
Mass of structure	Mst = 3,115 lb
Substage total (excluding payload)	Mtot = 34,259 lb

Total Mass MEM (48 hour Ell)	131,733 lb
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Orbit 4: (Mars SOI Transfer)

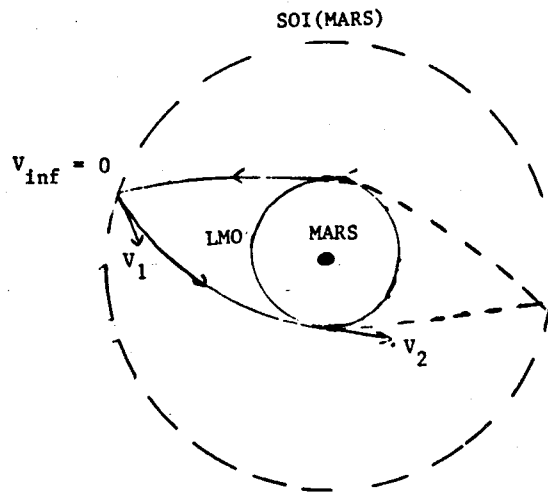


Figure 5

Similar to the ERV sizing procedure, a hyperbolic descent and ascent trajectory was followed.

a) Trajectory Results

SOI entry velocity	V3 = 3,281 ft/s
Delta V1	= 3,281 ft/s
Closest approach distance	Rp = 2,411 mi
Velocity at point 2 along hyper.	Vp = 15,801 ft/s
Circular velocity at LMO	Vc = 10,925 ft/s
Delta V2	= 4,888 ft/s

Total Delta V 8,169 ft/s
 Time of Flight (one way) <6.5 days

b) Mass Sizing

Ascent Stage:

Payload	Mpl =	5,602 lb
Mass of propellant	Mf =	4,112 lb
Mass of structure	Mst =	412 lb
Substage total (excluding payload)	Mtot =	4,524 lb

Descent stage

Payload	Mpl =	100,026 lb
Mass of propellant	Mf =	73,413 lb
Mass of structure	Mst =	7,341 lb
Substage total (excluding payload)	Mtot =	80,754 lb

Total Mass MEM (SOI)	180,780 lb
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Orbit 5: (Mars Libration Point One)

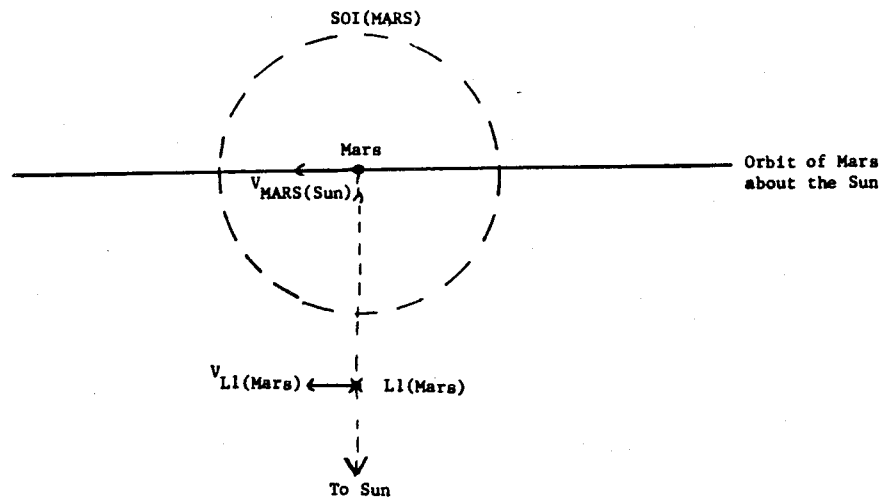


Figure 6

Figure 6 illustrates the geometry of transfer beginning at L1.

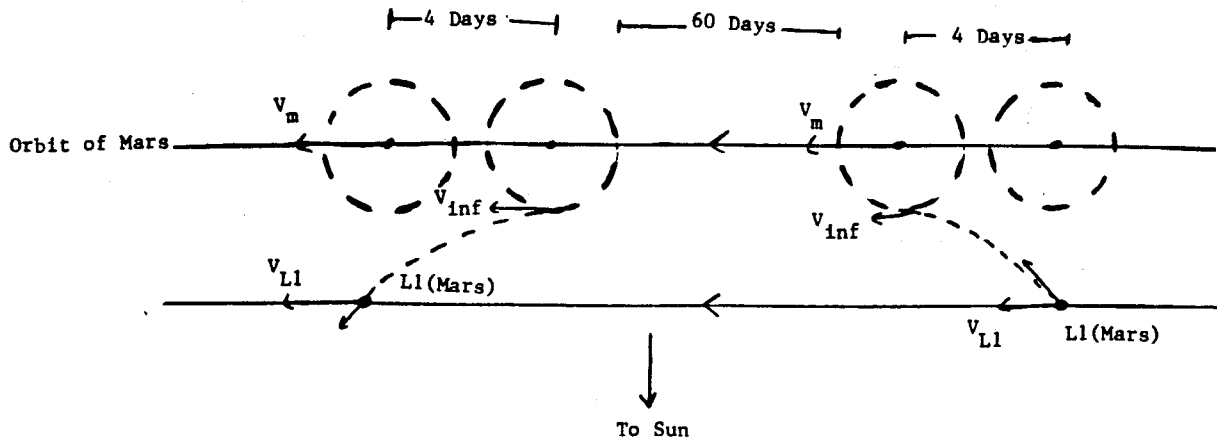


Figure 7

Figure 7 illustrates how the flight trajectory from L1 to SOI Mars and back was modelled. The gravitational acceleration (due to the sun) in this region was found to vary only on the order of $3.72 \times 10^{-5} \text{ft/s}^2$. Therefore, the gravitational acceleration acting on a body in this region was taken to be a constant ($g = 7.296 \times 10^{-3} \text{ft/s}^2$).

a) Transfer Results

Part 1

Time of flight	4.2 days
Initial velocity at L1	78,839 ft/s
Velocity at Mars	79,199 ft/s
Total Delta V	8,005 ft/s

Part 2

Same as those found under trajectory results Mars SOI transfer except vehicle now enters and leaves SOI with $V_3 = 3,281 \text{ft/s}$.

Total Delta V	4,873 ft/s
---------------	------------

b) Mass Sizing

Ascent stage

Part 1

Mass of propellant	Mf = 4,087 lb
Mass of structure	Mst = 410 lb
Substage total	Mtot = 4,497 lb

Part 2

Mass of propellant	Mf = 3,748 lb
Mass of structure	Mst = 375 lb
Substage total	Mtot = 4,123 lb

Descent stage

Part 2

Mass of propellant	Mf = 38,640 lb
Mass of structure	Mst = 3,865 lb
Substage total	Mtot = 42,505 lb
Mass of propellant	Mf = 106,969 lb
Mass of propellant	Mst = 10,699 lb
Substage total	Mtot = 117,670 lb

Total Mass MEM (L1)	264,297 lb
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B. Aerobrake Calculations

1. Earth Return Vehicle

Payload at LEO	Mpl = 31,177 lb
Aerobrake second stage	Mab = 10,084 lb
Resized first stage	Mst = 5,141 lb

Total Mass ERV (AB)	46,402 lb
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2. Mars Excursion Module

Orbit 1: (Low Mars Orbit)

- No Change

Mass = 95,503 lb

Orbit 2: (Aerosynchronous Orbit)

Ascent stage - same

Descent stage

Aerobrake mass for DV2 Mab = 18,370 lb

Revised descent stage Mst = 18,574 lb

Total Mass MEM (AREO) (AB)	135,018 lb
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Orbit 3: (48 hour Ell.)

Ascent stage - same

Descent stage

Aerobrake mass Mab = 1,971 lb

Total Mass MEM (48 hour Ell.) (AB)	112,095 lb
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Orbit 4: (SOI Mars)

Ascent stage - same

Descent stage

Aerobrake mass for DV2 Mab = 21,342 lb

Revised descent stage Mst = 32,443 lb

Total Mass MEM (SOI) (AB)	153,812 lb
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Orbit 5: (Libration Point 1)

Ascent stage - same

Descent stage

Aerobrake mass for DV2 Mab = 21,994 lb

Revised descent stage Mst = 91,877 lb

Total Mass MEM (L1) (AB)	217,996 lb
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APPENDIX 3

MULTIPLE MISSION SIZING

This appendix defines the procedure used in to calculate net multi-use on-orbit mass requirements (referenced to LEO) for the Baseline option. A discussion of potential refitting scenarios will be presented as well as the implications that each option has on a multiple Mars mission program.

SECTION I: DISCUSSION OF VEHICLE REFIT REQUIREMENTS

The mass statement for the baseline vehicle, as discussed in section 3.3, encompasses the initial on-orbit mass requirements for one complete manned Mars mission. Recall that at earth return (EOM) the baseline vehicle is placed in a GEO storage orbit. The following outlines various components of this stored vehicle that must be replaced or refurbished before another Mars exploration mission could be attempted.

A. Manifest of Vehicle Components

Vehicle Components: (Single Mission, BOM)

Hardware:

1. Propulsion system - power plant, radiator, and thruster system
2. Main vehicle structure - basic truss work, artificial gravity structure, etc.
3. Habitation modules
4. Mars Excursion Module
5. Earth Return Vehicle

Propellant Components:

1. High thrust departure stage
2. Low thrust propellant
3. LEO to GEO preflight low thrust transfer fuel
- B. Component Reuse or Replacement Guidelines

The following divides the vehicle components listed in Part A into those that were assumed reusable and those which needed to be replaced if the baseline vehicle is to be refitted for another mission.

1. Components that require replacement:
 - a) Mars Excursion Module
 - b) Earth Return Vehicle Propellant
 - c) High thrust GEO departure stage
 - d) Low thrust propellant
 - e) LEO to GEO pre-flight LT fuel stage
2. Components that are reusable:
 - a) Propulsion system
 - b) Vehicle structure
 - c) Habitation modules
 - d) Earth Return Vehicle Structure

SECTION II: DISCUSSION OF SERVICING OPTIONS

OPTION 1: LEO SERVICING

This option assumes that the baseline vehicle is transferred from the GEO storage orbit back to the LEO assembly facility for refit operations.

The following describes those elements that must be considered when analyzing the effect of refitting the baseline vehicle at LEO.

REFIT ELEMENTS:

R1) GEO to LEO TRANSFER STAGES:

Low thrust propellant must be transported to GEO so that the previously stored baseline vehicle can transfer back to LEO for servicing. (Assume LT OTV with a ratio of payload to total vehicle mass equal to .85).

R2) COMPONENTS THAT REQUIRE REPLACEMENT:

See section IB1 Of this appendix.

The total mass required to refit the baseline vehicle for any mission i , $i > 0$ is then given by:

$$M_{\text{refit1}}(i) = M_{r1} + M_{r2}$$

where:

M_{r1} and M_{r2} represent the total mass of refit elements R1 and R2, respectively.

Thus, for Option 1 the net multi-use on orbit mass (referenced to LEO) for N missions is given by:

$$M_{\text{net1}} = M_0 + \sum_{i=1}^{N-1} [M_{\text{refit1}}(i)]$$

where:

M_0 - mass of first mission

OPTION 2: GEO SERVICING

This option assumes that the baseline vehicle is fully refitted at GEO. Transfer of the vehicle back to LEO is no longer required. However, refit elements and personnel must now be transported to GEO.

The following outlines those elements that must be considered when analyzing refit operations at GEO.

REFIT ELEMENTS:

RR1) LEO to GEO transfer stage for replacement equipment. (Assumes LT OTV with a ratio of payload to total vehicle mass equal to .85).

RR2) Replacement equipment (see section IBl).

Using a similar approach as in Option 1, the total mass required to refit the baseline vehicle for any mission i , $i > 0$ is given by:

$$M_{\text{refit2}}(i) = M_{\text{rr1}} + M_{\text{rr2}}$$

Thus, the net multi-use on orbit mass (ref. LEO) for N missions is given by:

$$M_{\text{net2}} = M_0 + \sum_{i=1}^{N-1} [M_{\text{refit2}}(i)]$$

Note: This option would require an additional servicing facility located at GEO. The mass of such a station has not been included in the above mass estimated and therefore, constitutes an additional infrastructure assumption required by this servicing mode.

OPTION 3: (NO RE-USE)

In this scenario, a new baseline vehicle is constructed in LEO for each Mars mission. At end of mission (EOM) spent vehicles are permanently stored at GEO.

For N missions, the net mass for this option is given by:

$$M_{\text{net3}} = N * M_0$$

SECTION III: RESULTS AND CONCLUSIONS

A. Mass Data

Figure AP3.1 shows the mass requirements for a typical three mission Mars program utilizing each of the options discussed above.

B. Logistical Implications

Option 1 requires an infrastructure environment consistent with that assumed in Chapter 2.0. The possible net mass savings implied by such a servicing node is enhancing. However, a possible draw back inherent in this option would be the time consuming GEO to LEO return necessary before refit operations could begin.

Option 2, which also offers enhancing mass savings in a multi-use environment, also has unique logistical properties. Unlike Option 1, servicing the vehicle at GEO would eliminate delays incurred by requiring the NEP vehicle to continually spiral to and from LEO. After the first mission, the vehicle could immediately begin each succeeding mission without the 200 day LEO to GEO transfer at the beginning of the mission or the 100 day GEO to LEO return transfer end of mission.

However, as discussed earlier, refitting the vehicle at GEO would also require a servicing facility. This higher degree of infrastructure places greater demands on projected on-orbit capability than does Option 1.

Option 3, which was included as a reference case, yielded the highest total net mass requirement. In addition to the magnitude of additional mass required, it is important to recall the type of systems that would be lost if such as option were adopted. The primary

structure, propulsion system, and habitation modules represent the core systems of the baseline vehicle. These systems also represent the most intricate components of the vehicle. The added assembly demands required to replace these systems, as implied in a no-reuse environment, would greatly increase multiple mission logistics. Thus, while no reuse is a plausible option, it would fail to take advantage of important salvageable systems returned to Earth.

APPENDIX 4

BRIEF DISCUSSION ON ELECTRIC PROPULSION

For those not especially familiar with electric propulsion, the following is a very brief overview. First, the terms electric propulsion, low thrust propulsion, continuous thrust propulsion, or ion drive all refer to the same basic propulsion phenomenon that is the subject of this paper and are used by those in the field interchangeably. This form of propulsion is characterized by very high specific impulse (2,000 sec - 10,000 sec) and low acceleration levels (10^{-2} ft/sec² to 10^{-5} ft/sec²) produced over time periods comparable with the total time of flight for a particular transit.

The fundamental difference between electric propulsion and the more conventional chemical propulsive systems (see Figures AP4.1 and AP4.2, is that the former systems provide thrust via exogenous acceleration of the propellant while the latter relies on the endogenous reaction of an oxidizer and fuel to provide thrust. That is, electric propulsion systems use electrical energy to accelerate some type of inert propellant using electrostatic or electromagnetic forces. Because the energy source is independent of the propellant, exhaust velocities can be optimized for a particular mission. The ultimate performance limits of electric propulsion are only limited by the available power source. Chemical systems, as stated above, rely on an endogenous process (i.e., the conversion of stored chemical energy of reaction into kinetic energy) to produce thrust. This type of energy extraction method has a theoretical upper level performance limit of approximately 500 seconds of specific impulse (Reference 3).

Chemical systems typically operate for brief periods of time (seconds to minutes) at very high-thrust levels (millions of Newtons) to propel the ship along a "transfer ellipse" between the departure and arrival points. For most of the time, the ship "coasts" through space much the same way as an arrow upon leaving a bow "coasts" to its intended target. Thus, to effect a given momentum change, chemical systems eject massive amounts of material at relatively low velocity for brief periods of time.

In contrast, electric systems achieve a given momentum change by expelling low density material at very high velocity, thus requiring less mass (propellant) to complete a transfer. However, as mentioned earlier, due to the low density nature of the ejected mass, low acceleration levels are maintained. This means that electric systems must operate for long periods of time (days or months).

The trajectories that electric vehicles follow, also differ vastly from the ballistic trajectories followed by chemical vehicles. Because the thrust period for low thrust transfers are on the order of the transfer time, the vehicle is less dependent upon planetary alignments and assumes more of a powered flight trajectory. Classically, two types of thrusting profiles are used with electric propulsion systems. The first, the "variable thrust mode" (Reference 8), allows the thrust profile for the vehicle to vary continuously throughout the entire flight and yields the most flexible and efficient transfers. The second profile is that of the "constrained thrust mode" (Reference 8). This profile consists of a series of thrusting periods all at

relatively the same magnitude of total thrust with coast periods interspersed in which the thrust level is zero. Constrained thrust trajectories are always slightly less efficient than variable thrust profiles. Figure AP4.3 illustrates (for identical low thrust vehicles) typical variable thrust and constrained thrust profiles for a 270-day Earth-Mars transfer departing Earth and arriving at Mars at the same time, respectively. Note that the constrained thrust trajectory has a thrust/coast/thrust profile while the variable thrust profile varies continuously throughout the 270 days. In addition, also note that although the total time of flight for these two trajectories is the same, the variable thrust profile delivered slightly more terminal mass to the target.

Also, because of powered flight nature of low thrust vehicles, the standard analytic equations (i.e., the rocket equation and ballistic motion equations) usually employed to solve interplanetary trajectory problems are not valid in this case. To overcome this difficulty, some type of computer model is used to integrate the equations of motion given a set vehicle with specified propulsion and payload parameters. In addition, several codes are currently available, like the one used in the thesis work, (also see Reference 8) that go beyond even this level of sophistication and actually optimize the the vehicle and trajectory parameters in what ever direction an analyst may deem appropriate (i.e., mimization of travel time, maximizing delivered payload, determination of power, ISP, or power plant mass, etc.).

Thus, although low thrust trajectory/vehicle analysis is more complicated, it can easily be conducted with current tools and computer hardware.

In summation, electric propulsion can provide a reasonable alternative to chemical propulsion if a mission requires:

- o A large value of total impulse or delta-V (on the order of low earth orbit transfers to lunar distances and beyond)
- o A magnitude of thrust that is moderate (i.e., "in space" operation only, can not be used for earth to orbit transfers)
- o Noncritical thrusting time (i.e., total trip time should be on the order of months).