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MISSION ORIENTED STUDY OF ADVANCED NUCLEAR SYSTEM PARAMETERS

PHASE VI FINAL REPORT

VOLUME II

TECHNICAL REPORT

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TRW SYSTEMS GROUP

Volume II TECHNICAL REPORT

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FOREWORD

This volume, which is the second of a set of two volumes, describes the study tasks, analyses, and results of the Mission Oriented Study of Advanced Nuclear System Parameters, performed under Contract NAS8-5371, for George C. Marshall Space Flight Center, Huntsville, Alabama. This work was performed during the period from July 1967 to June 1968 and covers Phase VI of the subject contract.

The final report has been organized into a set of two separate volumes on the basis of contractual requirements. The volumes in this set are:

01977-6025-R0-00	Volume I	Summary Technical Report
01977-6026-R0-00	Volume II	Technical Report

Volume I summarizes and Volume II presents the details of the basic study guidelines and assumptions, the analysis approach, the analytic techniques developed, the analyses performed, the results obtained, and an evaluation of these results together with specific conclusions and recommendations. Also included in these two volumes are discussions of those areas of research and technology in which further effort would be desirable based on the results of the study.

This study was managed and principally performed by personnel in the Analytical Research Operations of the Systems Laboratories of TRW Systems. The principal contributors to this study were Messrs. G.M. Callies, A.R. Chovit, R.S. Schussler, and L.D. Simmons.

ABSTRACT

The details of the study approach and basic guidelines and assumptions which were used in a series of analyses of manned Mars lander and manned Mars and Venus orbital capture (no manned lander) missions are given. Analyses were performed for Mars missions employing opposition class, Venus swingby and conjunction class trajectories for launch opportunities from 1980 through 1993; the Venus missions employed inferior conjunction class trajectories for launch opportunities from 1980 through 1985. The investigations included comparative analyses of vehicles using cryogenic chemical, liquid storable, and nuclear rocket propulsion systems; nuclear rocket thrust levels of 75,000, 100,000 and 200,000 pounds were analyzed. Both circular and elliptic Mars parking orbits were investigated, and an analysis of Earth and Mars launch window requirements was made for selected missions. The analyses used and results obtained for these study tasks are presented, as well as an evaluation and recommendations based on the results.

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I. INTRODUCTION

This final report presents the details of the mission, trajectory, and vehicle analyses conducted during Phase VI of the Mission Oriented Study of Advanced Nuclear System Parameters performed by TRW Systems for the George C. Marshall Space Flight Center.

Included in this volume are the spacecraft weights, performance parameters, assumptions, and the analytical approaches used, together with the results of investigations of various problems related to the use of nuclear rocket propulsion for manned interplanetary missions. An evaluation of the results is also given together with specific conclusions and recommendations.

STUDY OBJECTIVES

The overall basic objective of this study was to expand and update the mission analyses performed in the earlier study phases of the subject contract. Specifically, the analyses would 1) update the mission evaluations performed in Phase IV for the 1980 to 1986 time period through the inclusion of the latest vehicle and propulsion system design information; 2) extend the mission evaluations to the 1988 to 1993 time period; 3) analyze Mars and Venus orbital capture missions for the 1980 to 1985 time period; and 4) investigate the use of elliptic Mars parking orbits for orbital capture missions.

At the approximate midpoint of the study, a revised set of parameter assumptions and task guidelines was formulated. This study reorientation was dictated by renewed interest in the development of a nuclear engine with a thrust in the range of 75, 000 to 100, 000 pounds; the initial Phase VI study efforts were centered upon the use of a 200, 000-pound thrust nuclear engine. Accordingly, the first half of the study period, in which the 200, 000-pound thrust nuclear engine was analyzed, was viewed as Phase A and the latter half, in which the lower thrust engines were analyzed, as Phase B. Due to the basically different objectives and resulting study tasks for these two phases, the analyses and results are discussed in separate sections of this report.

STUDY TASKS

A brief description of the study tasks performed in each phase is given below. A more detailed description of each task is included at the beginning of each task section in this report. In all cases, minimum vehicle weight refers to weight in Earth orbit prior to launch into the trans-planetary trajectory.

Phase A

The four tasks included under Phase A encompass all analyses performed for the 200,000-pound thrust nuclear engine, chemical cryogenic engines and liquid storable engines.

Task A1. Venus Swingby Missions - This task involved the determination of the minimum vehicle weight requirements in Earth orbit for launch of manned Mars lander missions employing the Venus swingby trajectory profile. These investigations included the analysis of various vehicle configurations using both nuclear and chemical propulsion systems for the launch opportunities occurring in the years 1980 through 1993. Swingby trajectory type 3 and both types 5 were analyzed, as they occurred in this range of opportunities. In addition, each swingby type was combined with both the long and short trajectory type for the direct leg of the round trip mission.

Appropriate trajectory data were generated as required for the SWOP program for these missions. The mission analyses were conducted within specified ETR launch site azimuth constraints. The results for the swingby missions were compared with the analogous results obtained in the subsequent two tasks.

Task A2. Opposition Class Missions - Similar investigations were conducted for opposition class lander missions. The analyses in this task were confined to the type II-B opposition class trajectory except in those cases in which the specified launch azimuth constraints were violated. Launch opportunities from 1980 through 1993 were analyzed.

Task A3. Conjunction Class Mission - The minimum vehicle weight requirements were determined for the type I-A conjunction class lander mission occurring during the 1983 launch opportunity. The analyses were performed for various vehicle configurations and considered the same launch azimuth constraints.

Task A4. Orbital Capture Missions - The minimum vehicle weight requirements were determined for manned Mars and Venus orbital capture missions (unmanned planetary probe) for launch opportunities from 1980 through 1984 for the Mars mission and from 1980 through 1985 for the Venus mission. Both opposition class and Venus swingby trajectories were analyzed for the Mars missions. The investigations included the analysis of various vehicle configurations using both nuclear and chemical propulsion systems.

Phase B

The four tasks included under Phase B encompass all analyses performed for the 75, 000- and 100, 000-pound thrust nuclear engines.

Task B1. Manned Mars Lander Missions - The minimum vehicle weight requirements were determined for manned Mars lander missions for each launch opportunity from 1984 through 1993 for opposition class or outbound or inbound Venus swingby trajectories. Both 75, 000- and 100, 000-pound thrust nuclear engines were investigated and a range of nuclear engine clustering arrangements were considered in order to determine the optimum engine clustering configurations for each engine thrust level. The optimum engine clustering was selected consistent with a specified maximum engine firing time constraint.

Task B2. Mars Orbital Capture Missions - The minimum vehicle weight requirements and optimum nuclear engine clustering arrangements were determined for a 1984 type II-B opposition class Mars orbital capture mission. Both circular and elliptic Mars parking orbits were investigated for both engine thrust levels.

Task B3. Mars Orbital Capture Missions, Aftercooled Engines - The minimum orbital launch vehicle weight requirements were determined for the same mission as in the preceding task, but with the arrive-Mars nuclear engine aftercooled and retained for providing the depart-Mars propulsion. Two aftercooling modes were considered: in the first, the arrive-Mars propellant tankage was jettisoned and the vehicle provided with separate depart-Mars propellant tankage; in the second mode, one propellant tank contained all propellant required for the arrive-Mars retro phase, the nuclear engine aftercooling, and the depart-Mars injection phase. Both circular and elliptic Mars parking orbits were investigated.

Task B4. Launch Window Analysis - An investigation was conducted to determine the effect on initial vehicle weight for Mars lander missions when launch windows are provided both at Earth and at Mars.

The effects of nodal regression of the parking orbits were taken into account and the propellant tanks were sized so as to provide the minimum initial weight vehicle necessary for permitting a launch on any day during the launch windows.

The analysis was conducted for an opposition class trajectory and an outbound and inbound Venus swingby trajectory. ETR launch azimuth constraints and maximum engine firing time constraints were considered in this analysis.

REVIEW OF PREVIOUS STUDY PHASES

Phase VI of the study utilized the mission optimization and vehicle sizing computer program developed during the earlier phases of the study as well as some of the previously developed parametric data and analysis techniques. Therefore, a brief review of Phases I through V is given here in order to provide continuity and background for Phase VI. (References 1 through 6 contain the details of Phases I through V.)

The first major task of Phases I, II and III was to develop a computer program that would permit the rapid determination of the optimum (minimum weight) trajectory for a variety of mission modes, propulsive systems, vehicle configurations, system and payload weights and scaling laws, and performance parameters. This computer program was given the acronym SWOP (Swingby Optimization Program). The SWOP program was then utilized to analyze manned interplanetary missions for various trajectory types, launch opportunities, vehicle configurations, and performance parameters in order to determine the vehicle performance as a function of the nuclear engine thrust level for missions in the 1975 to 1990 time period. Detailed analyses were also made to determine the vehicle and stage weight sensitivity to variations in performance, vehicle, and mission parameters. Concurrently, a nuclear optimization computer program (NOP), also developed in the study, was used for analyzing the detailed engine design parameters in terms of their effect on the engine weight, thrust, and specific impulse.

In this manner it was possible to determine within a narrow range the mission, vehicle, and engine requirements for future manned interplanetary missions. Within this narrow range a more detailed analysis was then performed which related the vehicle and mission requirements to variations in specific engine design parameters. The information obtained from the detailed evaluations then permitted the identification of the optimum engine design requirements and the major vehicle and mission criteria.

Phases IV and V were concerned with expanding the detailed mission and engine analyses performed in the earlier study phases to include trade-off studies of alternative propellant tank configurations, additional mission modes, launch window and abort analyses, and alternative nuclear engine design criteria. In addition, both the SWOP and NOP computer programs were revised to incorporate additional mission, engine, and vehicle parameters that would render the programs more effective.

SWOP DESCRIPTION

The SWOP program was the primary tool utilized in optimizing and analyzing the various missions in this study as well as sizing the vehicle component systems and computing the initial vehicle weights. Therefore, a more or less detailed description of the program is included here to indicate the manner in which the program was utilized and to present the level of detail to which the vehicles were configured.

The SWOP program uses a unique combination of analytic and mathematical optimization techniques, specified curve fit routines and pre-computational processing, selection, and storage of trajectory and performance data to minimize the initial vehicle weight in Earth orbit with respect to all the velocity changes (propulsive and aerodynamic braking), the trip times (life support expendables, and micrometeoroid protection), the propellant boiloff requirements, and the planet passage distance constraints (for swingby missions). The vehicle is configured by the program by means of parameter options and payload specifications. In addition to the variable propulsive or aerodynamic stage weights which make up the vehicle, the program computes or provides for various weight provisions including attitude control, midcourse corrections, planet lander, and Earth lander (after retro or aerodynamic braking). The program also considers the addition or deletion of fixed weights at various points along the mission trajectory on option.

All variable weights are sized using general scaling laws whose coefficients are input. The trajectory data used by the program are preprocessed free flight data and powered flight information. The program has the capability of optimizing a mission for one or more constrained trajectory or velocity parameters. These include the launch or arrival dates at Earth, the target, or the swingby planet; the individual leg or total trip times; one or more of the velocity increments; the perihelion distance; the periapsis distance at the swingby planet; and the propulsion systems' thrust, thrust-to-weight ratio, or percentage gravity loss. When one or more of the independent parameters are constrained, the program optimizes those that are unconstrained; if all are constrained, the vehicle is sized for the fixed trajectory. The vehicle propulsion stages can be selected to be nuclear (aftercooled or non-aftercooled), chemical cryogenic, or storable chemical. The planet braking maneuvers can be propulsive, aerodynamic, or a combination of propulsive and aerodynamic braking.

The computed vehicle weight is the minimum gross spacecraft weight that is required to perform the mission for the specified vehicle, payload, trajectory, and performance constraints. This weight corresponds to the overall vehicle weight at the point just prior to boost out of Earth parking orbit. The vehicle weight in all cases is computed using trajectory characteristics that are optimum for the selected constraints, i.e., the particular launch dates and trip times used (with the corresponding characteristic velocities and perihelion distance) produce the minimum overall vehicle weight. In addition, the program computes and outputs the vehicle weight before and after every powered phase of the mission as well as all propellant, insulation, and tank weights.

The initial vehicle weight data are based on calculations for the propellant weight in which the velocity losses due to operation in a gravity field are taken into account in an exact manner. The gravity losses can be determined by either specifying: a) a fixed engine thrust, b) a fixed percentage increase of the impulsive velocity, or c) a fixed vehicle thrust-to-weight ratio.

Running time for the SWOP program ranges from 0.4 to 2 seconds per case depending upon the computer system used.

REPORT ORGANIZATION

The following section of this report (Section II) presents the vehicle scaling laws, performance parameters, assumptions, and constraints that were used and are applicable to all analyses performed in Phase VI of this study. Sections III and IV present the mission and vehicle mode matrices, the analysis approach, and the results for each task performed during Phases A and B, respectively. Sections V, VI, and VII discuss future research and advanced technology areas for manned interplanetary missions; present a summary of the more salient results for each task; and give a list of references used.

II. MISSION AND VEHICLE PERFORMANCE PARAMETERS

A number of basic guidelines and parameter values were established initially in the study which define: the missions and trajectory types; the performance of the propulsion systems; the velocity losses due to finite thrusting in a gravity field; the vehicle stage weight scaling laws; the mission payloads and expendables; the magnitude of midcourse corrections, orbit adjustments, and rendezvous maneuvers; the weight of cryogenic propellant vaporized; and insulation weight.

MISSIONS AND TRAJECTORY TYPES

Mission Descriptions

Four basic manned interplanetary missions were investigated in this study. These were:

- Opposition Class Lander Mission
- Venus Swingby Lander Mission
- Conjunction Class Lander Mission
- Orbital Capture Mission with Unmanned Probe

A typical Mars opposition class lander mission is shown in Figure II-1, which depicts the major operational phases that occur during the mission and the points along the trajectory at which major velocity and vehicle weight changes occur. Additional vehicle weight allowances are made for life support expendables, propellant boiloff, and attitude control. A propulsive maneuver is also provided for orbit adjustment after capture into the Mars parking orbit. The Earth braking maneuver is accomplished by aerodynamic braking or by a combination propulsive retro followed by aerodynamic braking. The Mars excursion module is a manned vehicle which descends to the Martian surface from the orbiting spacecraft and after surface exploration ascends to rendezvous with the orbiting spacecraft.

A Venus swingby lander mission is essentially the same as the Mars opposition class lander mission depicted in Figure II-1 except the trajectory is constrained to pass in the vicinity of the planet Venus during either the outbound or inbound leg. The vehicle therefore performs

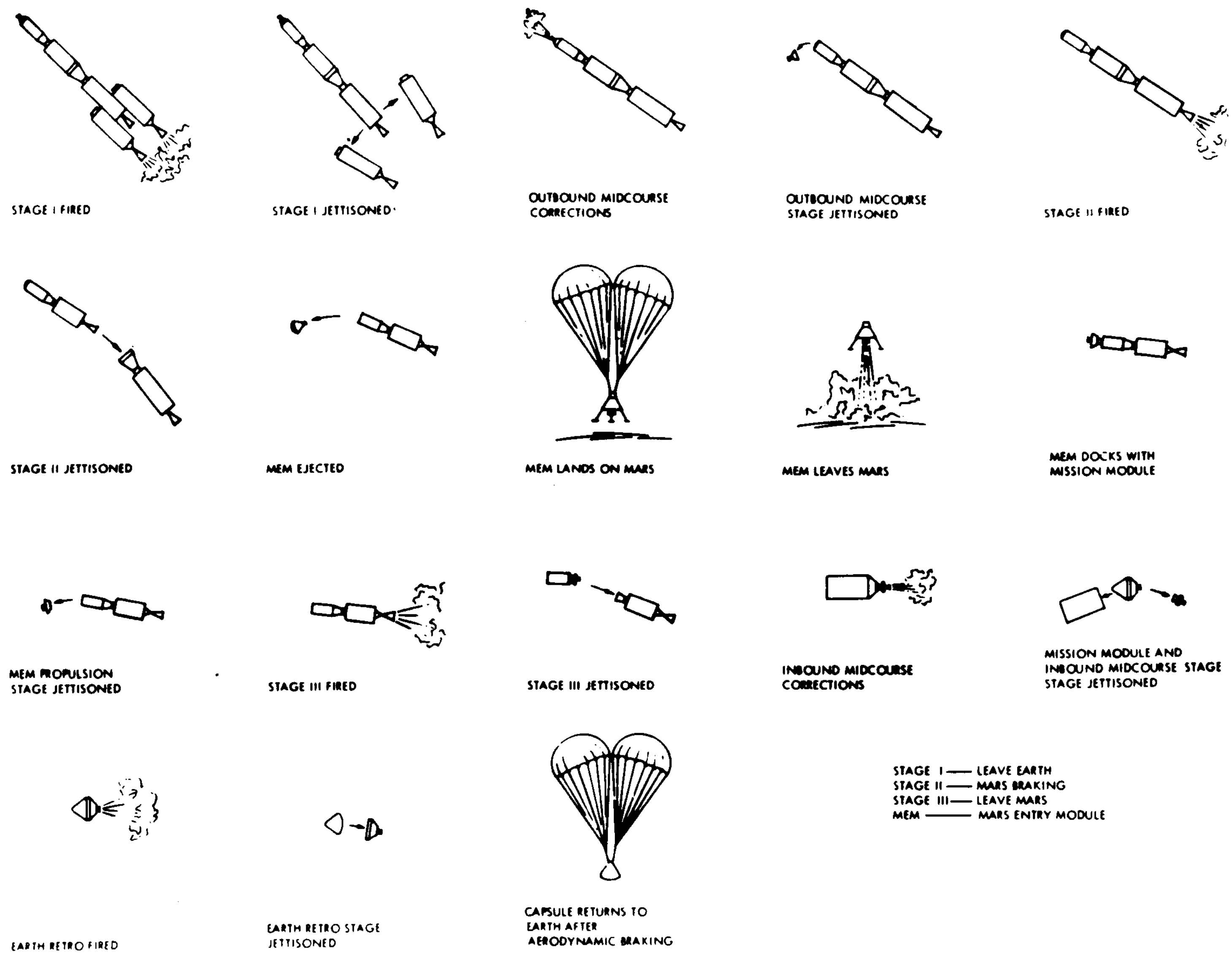


Figure II-1. Mars Lander Mission Sequence a hyperbolic turn about Venus. For a given approach V_∞ the degree of turn is governed by the choice of the periapsis radius. For the swingby mission, a third midcourse correction propulsion maneuver is assumed.

The conjunction class lander mission investigated in this study is the double-Hohmann class of mission. It resembles the opposition class lander mission in that an opposition occurs during the outbound trip. The dwell time at Mars, however, is extended so that the following opposition occurs during the inbound trip, which is then another near Hohmann transfer. The intervening conjunction occurs during the dwell period at Mars, giving rise to the designation of a conjunction class mission. The dwell time at Mars is optimized to yield the minimum weight vehicle and is characteristically about 400 days.

The orbital capture missions are similar to either the opposition class lander or Venus swingby lander mission with the essential difference that in lieu of a manned Mars excursion module, an unmanned planetary probe is separated from the orbiting spacecraft and no payload is subsequently recovered. In this study the orbital capture mission was investigated for both Mars and Venus as the target planets; in the case of the Mars missions, both opposition class and Venus swingby trajectories were analyzed.

Trajectory Types

The IIB round trip trajectory was the basic trajectory considered for the opposition class mission analyses. (Trajectory types I and II refer to the outbound leg; types A and B refer to the inbound leg. The I or B denotes a trajectory leg where the heliocentric angle traversed, θ, is greater than 180^o and less than 360^o; the II or A designates a trajectory leg where $0^o < \theta < 180^o$). It was previously shown in Phase III (Ref. 1) that the IIB trajectory generally produces the minimum initial vehicle weight for all opportunities. For a few opportunities in the Earth-Mars synodic cycle and for certain vehicle mode and performance combinations, the IB trajectory can result in a slightly lower weight vehicle (approximately two percent), but with an attendant increase in total trip time of approximately 13 percent. In certain cases the type IIB trajectory violated specified launch azimuth constraints and the type IB trajectory was analyzed.

Two types of trajectories were considered for the direct leg of each of the swingby missions, types I or B and types II or A. Two types of trajectories were considered for the swingby leg of the swingby missions, types 3 and 5. A detailed discussion of swingby trajectory characteristics is presented in References 5 and 7.

A IA conjunction class mission trajectory was selected for comparing the conjunction class mission with the opposition and swingby class missions in this study. The IA conjunction class trajectory yields a lower weight vehicle than the other three possible trajectories (types IB, IIA, and IIB). The total trip time for the type IA trajectory is within approximately three percent of the minimum trip time obtained for the other types. A full discussion of conjunction class missions is presented in Reference 5.

PROPULSION SYSTEM PERFORMANCE

Three basic propulsion systems were investigated in this study, viz; 1) nuclear rocket engines of the NERVA class, 2) chemical cryogenic stages using liquid oxygen and hydrogen for propellants, and 3) liquid storable propulsion systems. Combinations of these systems were used to formulate the many vehicle configurations analyzed for the various mission modes. The performance parameters for these systems are given in Table II-1.

Table II-1. Propulsion System Performance Parameters

Type	Specific Impulse (sec)	Thrust (lb)	Engine Weight (lb)
Nuclear	850	200,000	30,750
	850	100,000	20,000
	850	75,000	18,000
Cryogenic	460		
Storable	380		

The engine weight and thrust levels for clusters of two or more nuclear engines were taken as direct multiples of the above values.

GRAVITY LOSSES

The computations of the initial vehicle weights are based on calculations for the propellant weight in which the velocity losses due to operation in a gravity field are taken into account in an exact manner. The gravity losses are determined by specifying either a fixed engine thrust for nuclear engines or a fixed percentage increase of the impulsive velocity for chemical propulsion systems.

For vehicles employing nuclear propulsion stages, these losses are based on the required velocity change, the engine specific impulse, and the vehicle thrust-to-weight ratio obtained from the computed vehicle weight and the specified engine thrust.

For vehicles employing chemical propulsion systems, the characteristic velocity is obtained by increasing the required impulsive velocity change by a fixed percentage. The percentage values used are shown in Table II-2; they are valid for both Mars and Venus missions.

Table II-2. Gravity Losses for Chemical Propulsion Systems

Propulsive Phase	Propulsion Mode	Percentage Increase
Depart Earth	Cryogenic	2.3
Arrive Planet	Cryogenic	0
Depart Planet	Cryogenic or Storable	1
Arrive Earth Retro	Storable	0

The impulsive velocities used in computing the propellant requirements are based on the assumption that the spacecraft injects into an interplanetary orbit from a 500-km circular orbit at Earth and a 600-km circular orbit at Mars or Venus; for the braking maneuver at Mars or Venus, the vehicle is decelerated into a 600-km circular orbit. When elliptic parking orbits were utilized at Mars, the periapsis altitude was assumed to be 600-km to coincide with the orbital altitude used for circular orbits. The apoapsis altitude was chosen so that the ratio of apoapsis to periapsis radius would be six (orbital period ≈ 14 hours).

VEHICLE CONFIGURATION AND SYSTEM WEIGHTS

A number of assumptions, constraints, and scaling laws were used concerning the mission payloads, propellant tanks, propulsion systems, secondary spacecraft systems, and operational modes. The values and scaling laws listed in this section are applicable to all of the mission operational modes except when nuclear aftercooling is employed at the target planet. The analysis of this mode required special scaling laws due to its operational uniqueness. For this case the exceptions to the values in this section are given at the point where the analysis is discussed in a subsequent section.

Two inherently different types of vehicle configurations were used for this study; a tanking mode for vehicles with cryogenic propulsion, and a connecting mode for vehicles utilizing nuclear engines.

The tanking mode tends to make full use of the Earth launch vehicle payload volume capacity by initially orbiting empty or partially filled modules. The modules are then filled via propellant transfer from tanker vehicles or from an orbital propellant storage facility. Thus, it is assumed that all propellant tanks are in a completely full condition just prior to Earth injection. The maximum propellant capacity for a given tank is set by the limitations of the Saturn V launch vehicle payload weight capability or overall vehicle length limitation.

In the connecting mode, the modules are orbited fully loaded with propellant, hence, their capacity is also limited by the Earth launch vehicle. Use of the connecting mode gives rise to a specific vehicle design configuration or method of adding tankage to each stage as the propellant requirements increase. An example of a typical vehicle employing the connecting mode design is shown in Figure II-2. For the leave Earth stage, a cluster of three propulsion modules is first assumed, with each propulsion module containing a nuclear engine. This set of three modules is designated tier 1. If these three modules have insufficient capacity to contain the required propellant, a single propellant module, designated tier 2A, is attached above tier 1. If the total propellant capacity of tier 1 and tier 2A is still insufficient, two additional propellant modules are clustered to the single propellant module. The resultant propellant modules are designated tier 2B. Should the total propellant capacity still be insufficient, another single propellant module, designated tier 3, is attached above tier 2B.

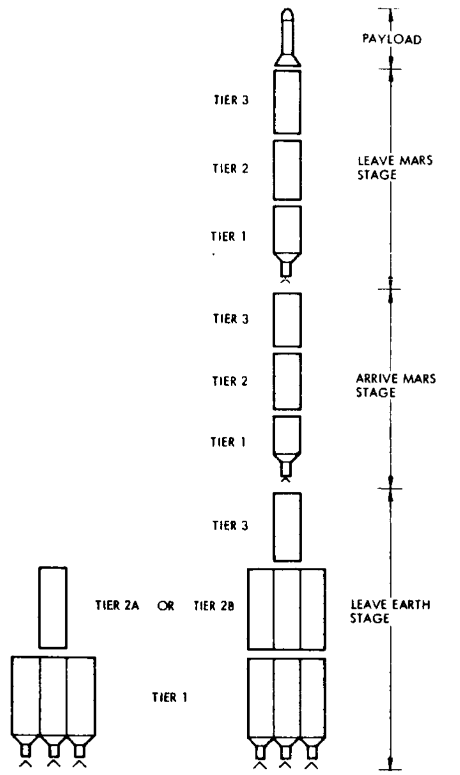


Figure II-2. Typical Connecting Mode Vehicle Configuration

The configurations for the arrive Mars and leave Mars stages are similar to the leave Earth stage except that single propulsion modules or propellant modules are used at each level or tier in the example configuration shown in Figure II-2.

To increase the gross effective thrust for vehicles employing nuclear engines, in an effort to reduce the velocity gravity losses, engine clustering, or the simultaneous use of two or more nuclear engines was investigated in the Phase B Tasks for each of the three main propulsion stages i. e., leave Earth, arrive Mars, and leave Mars, and the optimum number of engines required for the leave Earth and arrive Mars stages was then determined. The optimum number of engines for each stage was taken as the configuration producing the minimum weight vehicle in Earth orbit and in which no nuclear engine firing time exceeded 2700 seconds. Thus, the above schematic illustration (Figure II-2) represents only one of the many possible vehicle connecting mode configuration designs investigated during the course of this study.

Each of the two modes, the tanking mode and the connecting mode, has structural scaling laws for computing the tank and propulsion system weights. The scaling laws given in Tables II-3 and II-4 are in equation form for the chemical propulsion systems and corresponding mass fractions for the nuclear propulsion connecting mode configurations. (Detailed equations for the connecting mode scaling laws are given in Sections III and IV for the 200,000-pound thrust engine, and the 75,000-and 100,000-pound thrust engines, respectively. Due to varying interstage structure requirements for the different engine clustering arrangements, each clustering configuration has a corresponding set of scaling equations.) Included in these tables are the scaling laws used for midcourse correction stages, the planetary orbit adjustment stage, and the arrive Earth retro stage. Table II-4 also includes a propellant shield weight scaling law. For the nuclear connecting mode the propellant tanks are surrounded by a combination meteoroid and heat protection shield to reduce boiloff during the outbound transit and planetary stopover period. This propellant shield is jettisoned just prior to the arrive planet and depart planet propulsive maneuvers.

In addition to the tank and engine weights associated with each stage of the connecting mode configuration, a block weight is assigned to each stage to account for radar, docking and interstage structure, the attachment members, and the separating mechanism. This weight, which is a function of the number of propulsion modules, is designated the stage constant and takes the form of a fixed weight assigned to each stage. Each stage was also sized to include flight performance reserve propellants. Reserve propellant provisions were computed on the basis of a 0.75 percent increase in the required characteristic velocity for each stage.

These scaling laws were based on vehicle design data selected by MSFC for this study and are applicable to both Mars and Venus missions.

Table II-3. Tanking Mode Scaling Laws

Mission Phase	Equation (lb)	Single Tank Max Propellant Capacity (lb)
Earth Depart		
Cryogenic Propulsion	$W_j = .08427\ W_p + 9494$	700,000
Midcourse Correction Outbound		
Storable Propulsion	$W_j = .05732\ W_p + 1442$	
Planet Braking		
Cryogenic Propulsion	$W_j = .08427\ W_p + 9494$	700,000
Storable Propulsion	$W_j = .03121\ W_p + 21{,}997$	800,000
Planet Depart		
Cryogenic Propulsion	$W_j = .08427\ W_p + 7924$	700,000
Storable Propulsion	$W_j = .03121\ W_p + 20{,}427$	800,000
Midcourse Correction Inbound		
Storable Propulsion	$W_j = .03310\ W_p + 888$	
Earth Braking		
Storable Propulsion	$W_j = .05312\ W_p + 3491$	

Notes:

1. Includes micrometeoroid protection
2. Includes insulation for Earth depart stages
3. Does not include insulation for all other stages
4. Includes engine weight for all stages
5. W_j equals stage jettison weight per propellant tank
   W_p equals total propellant required per propellant tank

Table II-4. Connecting Mode Scaling Laws

Mission Phase	Equation or Average Mass Fraction
Earth Depart	
Nuclear Propulsion	
Propulsion Module	0.84
Propellant Module	0.86
Midcourse Correction Outbound	
Storable Propulsion	$W_j = 0.05732\ W_p + 1442$
Planet Braking	
Nuclear Propulsion	
Propulsion Module	0.84
Propellant Module	0.86
Propellant Tank Shield	$W_j = 0.0233\ W_p - 230 + 5140\ m$
Planet Depart	
Nuclear Propulsion	
Propulsion Module	0.84
Propellant Module	0.86
Propellant Tank Shield	$W_j = 0.0260\ W_p - 260$
Midcourse Correction Inbound	
Storable Propulsion	$W_j = 0.03310\ W_p + 888$
Earth Braking	
Storable Propulsion	$W_j = 0.05312\ W_p + 3491$

Notes:
1. Includes micrometeoroid protection
2. Includes engine weight for all non-nuclear stages
3. Does not include engine weight(s) for all nuclear stages
4. W_j equals stage jettison weight per propellant tank

   W_p equals total propellant required for propulsion or propellant tank

   "m" equals number of arrive Mars nuclear engines

PAYLOADS AND EXPENDABLE WEIGHTS

The payloads and expendable weights assigned to the various missions were selected by MSFC for this study. Three sets of payload weights were used. The first set is applicable to the Mars and Venus orbital capture modes in which the various modules and expendables were sized for accommodating a six-man crew. The second set is sized for an eight-man crew and is used only for the Mars stopover lander missions. The third set is applicable only to the Mars conjunction class mission and is sized for a twelve-man crew. The module and expendable weights for this mission are approximately 50 percent greater than the weights for the opposition and swingby class missions to account for the increased crew size and crew and system requirements dictated by the long stay time at Mars. A list of the payloads and the expendable weights is given in Table II-5.

Table II-5. Payloads and Expendable Weights

	Set 1	Set 2	Set 3
Payload	Orbital Capture	Stopover Lander	Conjunction Class
Crew Size (Men)	6	8	12
Earth Return Module (lb)	12,000	12,000	16,000
Mission Module (lb)	80,000	80,000	110,000
Solar Flare Shield (lb)	14,000	14,000	20,000
Mars Excursion Module (lb)	None	100,000	150,000
Mars Orbit Return Weight (lb)	None	1,500	3,000
Drop Weight for Capture Missions (lb)	35,000 (Mars) 20,000 (Venus)		
Life Support Expendables (lb/day)	30	35	50

The Earth recovered payload lands the crew on the Earth's surface after aerodynamic braking has been accomplished. It consists of the crew and the required structure, landing and recovery aids, power supply, communications, guidance and navigation equipment, reaction jets, life support systems, and any space or planetary payloads that may be returned to Earth.

The mission module contains all systems, equipment, and living quarters required during the full duration of the mission. This module is jettisoned just prior to retrobraking at Earth or aerodynamic braking if a retro is not employed. It consists of structure, crew quarters, life support systems, medical supplies and recreation equipment, communication, guidance and navigation systems, power supplies, maintenance facilities and spare parts, and air locks. The solar flare shield is not included in the mission module weight. The shield weight is a separate, fixed weight that is jettisoned along with the mission module.

The planet excursion module for the stopover lander mission is launched from the spacecraft out of the planetary capture orbit. It contains the required systems and equipment for landing the module on the planet surface and subsequently performing scientific and engineering experiments. In addition, the Mars excursion module contains a crew compartment and the ascent or orbit return module which returns the crew and payload to the orbiting spacecraft. The specified weight for the orbit return module includes only that portion of the module which is taken onboard the orbiting spacecraft and subsequently boosted out of planetary orbit. The drop weight for the Mars and Venus orbital capture missions is considered to be a lander probe only which is not to be manned or returned.

The life support expendables include all of the crew's environmental and biological requirements which are expended at an average daily rate for the entire crew complement for the duration of the mission.

AERODYNAMIC BRAKING SCALING LAWS

In the analyses of the mission modes employing aerodynamic braking for the Earth entry module, the weight of the aerodynamic heat shield was expressed as a function of the atmospheric entry velocity at a

100 km altitude. The analysis and derivation of the Earth aerodynamic braking scaling laws were accomplished during Phase III and are fully described in Reference 1.

The scaling laws for aerodynamically braking the Earth return module are given below for the two module weights used in this study.

$W_R = 12,000$

$$W_{ERM} = 50.9\ V_{AE}^2 - 1141\ V_{AE} + 23,160$$

$W_R = 16,000$

$$W_{ERM} = 57.2\ V_{AE}^2 - 1262\ V_{AE} + 28,570$$

where

W_R = Recovered or return payload weight after Earth entry (lbs)

W_{ERM} = Gross vehicle weight or Earth entry module weight (lbs)

V_{AE} = Entry velocity relative to a non-rotating Earth at an altitude of 100 km (km/sec)

SECONDARY SPACECRAFT SYSTEMS

Additional weight expenditures were allowed for secondary spacecraft systems including midcourse corrections, attitude control, and parking orbit adjustment.

It was assumed in all mission calculations that the midcourse corrections were performed with a liquid storable propellant system having a specific impulse of 380 seconds. Separate jettisonable stages were used for the outbound and inbound leg velocity corrections and for a third leg correction for swingby missions. The scaling laws for the jettisonable stages were given previously under Propulsion System Weight Scaling Laws. A midcourse correction of 100 m/sec was used for each outbound and inbound leg as well as for the additional leg of a swingby mission.

The attitude control functions include orientation for midcourse corrections, spinning of the spacecraft or mission module for artificial gravity or thermal control, orientation of communication antennas, sensors, radiators, or solar panels or collectors, and orientation for planetary rendevous or propulsive or aerodynamic braking. Attitude control provisions were computed at the rate of 8 pounds per day during the planetary stopover period and during each leg of the mission, including the third or swingby leg.

The inbound midcourse correction propulsion system was also employed for adjusting the orbit after braking at Mars or Venus. This propulsive maneuver was sized for a characteristic velocity of 50 m/sec.

CRYOGENIC PROPELLANT VAPORIZATION

Due to the basically different design, launch, and assembly philosophies inherent in the two configuration modes, viz, the tanking mode and the connecting mode, two separate computational techniques were employed for determining the propellant vaporized during the interplanetary trip.

For the tanking mode the analysis determines the optimum tradeoff between the thickness or weight of insulation and the weight of vaporized propellant such that a minimum-weight vehicle results. The insulation requirements for each stage are determined separately, resulting in different insulation thicknesses for each stage. The connecting mode assumes that the insulation thickness is the same for all of the stages and is preselected to form the best compromise for all of the mission phases during which propellant is vaporized.

The cryogenic propellant storage analysis for the tanking mode permits the sizing of the required tankage insulation and calculation of the weight of propellant boiled off during the mission to yield a minimum overall vehicle weight. The analysis and derivation of the necessary equations was performed during Phase III and is detailed in Ref 1. The equations form the basis of the insulation/boiloff optimization subroutine in the SWOP program. The assumption of vented tanks was made and insulation requirements were considered and sized only for the conditions and storage durations commencing with the point just prior to boost out of Earth orbit. At this initial point, it was assumed that all tanks were full.

The optimum selection of the insulation requirements for subsequent cryogenic propellant stages is dependent not only on the insulation and thermal parameters (density, conductivity, temperatures, etc.) but also considers the duration of storage and the sizes, number, and times of vehicle propulsive velocity changes. For a multistage vehicle, the relationships between these latter factors has a major influence in the trade-off between insulation and propellant boiloff.

In the optimization analysis for the cryogenic bipropellants, separate equations are employed for the fuel and oxidizer, obtaining separate insulation and boiloff weights for each propellant component. Appropriate tank areas, heats of vaporization, and temperature differences are used in each case.

The following assumptions and values were used for specifying the various insulation and thermal constants in the optimization analysis for the tanking mode.

The insultation was assumed to be National Research Corporation's NRC-2, which consists of layers of crinkled aluminized mylar 0.25 mil thick. The nominal values of the insulation thermal conductivity and density are 7×10^{-5} Btu/hr. ft. ^{o}R and 3 lb/ft^{3}, respectively. In determining the temperature differences across the insulation, a non-spinning tank was assumed and an average temperature difference over the entire tank surface was calculated. No planetary influence or heat sources other than the sun were assumed and an average distance to the sun of 1.2 AU was used. A solar absorptivity of 0.20 and an emissivity equal to 0.80 were used for the tank surface conditions. The average temperature differences across the insulation computed for liquid hydrogen tanks is 160^{o}R and for liquid oxygen tanks, 34^{o}R. The heats of vaporization for hydrogen and oxygen are 186.95 and 91.6 Btu/lb, respectively.

The propellant storage analysis for the nuclear connecting mode determines the weight of propellant vaporized during the various phases of the mission based on specified rates of propellant boiloff, i. e., fixed insulation thickness. The weight of this insulation per tank is, therefore, a fixed quantity and is included in the scaling laws previously listed for the connecting mode. As for the tanking mode, the propellant vaporized or heat absorbed during assembly and checkout in Earth orbit was not considered; i. e., boiloff computations commenced just prior to injection into the interplanetary orbit.

The weight of propellant vaporized for the arrive- and depart-Mars (or Venus) stages during the interplanetary trajectory and stopover period was based on the actual mission durations and propellant tank requirements and was computed for each mission case investigated. The propellant boiloff rates used for these computations are listed in Table II-6.

Table II-6. Propellant Vaporization Rates

Mission Phase	Structural	Tank Wall
Outbound Leg		
Propulsion Module	39.1 lb/day per tank	3.30×10^{-3} lb/day ft^2 of tank area
Propellant Module	24.9	3.30×10^{-3}
Planetary Orbit		
Propulsion Module	75.0	6.38×10^{-3}
Propellant Module	47.6	6.38×10^{-3}

VEHICLE MODE NOMENCLATURE

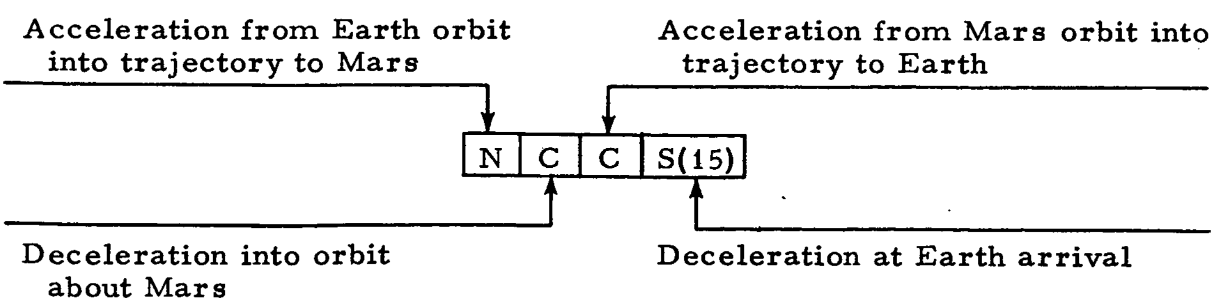
The vehicle configurations considered in the analyses covered a wide range of possible combinations of nuclear, cryogenic, and storable propellant propulsion systems. Reference to the different configurations was simplified by use of a simple nomenclature based on the source of the energy change applied at each discrete point in the mission where an energy change is required. The symbols used to denote the different sources of energy change are listed in Table II-7. The scaling laws and performance parameters for the nuclear aftercooled engines (N_J and N_{NJ})

are given as part of the discussion of these systems in Section IV. The scaling laws and performance parameters for all other propulsion stages were given earlier in this section.

Table II-7. Vehicle Configuration Symbols

Symbol	Means of Energy Change
N	Nuclear engine
N_J	Aftercooled nuclear engine; jettison propellant tankage (100, 000 lb thrust)
N_{NJ}	Aftercooled nuclear engine; retain propellant tankage (100, 000 lb thrust)
C	Chemical cryogenic
S	Storable chemical propulsion
A	Aerodynamic braking
S(15)	Storable chemical retropropulsion to 15 km/sec followed by aerodynamic braking

Vehicle configurations are designated by a code made up of the symbols in Table II-7. A four-element code is used:



Thus, a designation NCCS(15) would refer to a vehicle using the 200,000 lb thrust nuclear engine and a chemical cryogenic engine, respectively, for injection into Earth-to-Mars and Mars-to-Earth trajectories with chemical cryogenic propulsion for braking into a Mars capture orbit and storable chemical propulsion for braking to 15 km/sec at Earth arrival, followed by aerodynamic braking to enter and land.

III. PHASE A MISSION ANALYSIS

Phase A of this study consisted of four mission analysis tasks. These investigations included analyses performed for chemical propulsion systems and the 200,000-pound thrust nuclear engine. The first three tasks involved the mission analysis of Mars stopover lander missions employing Venus swingby, opposition class, and conjunction class trajectory profiles, respectively, for the 1980 to 1993 launch opportunities. The results of these mission evaluations were then compared to illustrate the effect on initial vehicle weight of the variations in launch opportunities, mission and trajectory type, and nuclear engine design criteria.

The final task was an analysis of Mars and Venus orbital capture missions which included the determination of the initial vehicle weight requirements for parametric variations in trajectory types, stopover times, and vehicle and propulsion modes.

The mission analyses for the above tasks included the launch azimuth constraints imposed by range safety restrictions and the physical limits on the departure declination achievable for launches from the ETR.

MISSION AND VEHICLE PERFORMANCE PARAMETERS

The primary mission and vehicle performance parameters that were postulated for this study phase in order to circumscribe the vehicle system weights and performance, vehicle configuration, mission and vehicle operational criteria, and the scope of the analysis were specified in Section II. Included in this section are the details of various scaling laws and constraints that apply primarily to the Phase A analyses.

Vehicle Configuration and Propulsion System Weights

The scaling laws and system weights used to define the mission payloads, propellant tanks, propulsion systems, secondary spacecraft systems, and operational modes are essentially those given in Section II except for the qualifications noted below.

The vehicle modes analyzed in the Phase A tasks centered upon the use of a 200,000-pound thrust nuclear engine and chemical propulsion systems; both the tanking mode and the nuclear connecting mode configurations were assumed as appropriate. In addition to an all nuclear or all chemically propelled vehicle, the use of nuclear and chemical engines combined in a single vehicle but in separate stages was also considered for evaluation and comparison purposes. However, once a chemical stage is introduced into a particular mission (ignoring midcourse corrections and orbit adjustments), all remaining major stages employ chemical propulsion.

The scaling laws used for the chemical propulsion systems were previously given in Section II in Table II-3, while the average mass fractions used for the nuclear propulsion connecting mode configurations were presented in Table II-4 of the same section. A detailed list of the connecting mode scaling laws used to size the 200,000-pound thrust nuclear engine modules is given in Table III-1 for the various mission phases. These equations were derived for a vehicle using a single

Table III-1. Connecting Mode Scaling Laws, 200,000-Pound Thrust Nuclear Engine 3-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = .08085\ W_p + 52{,}410$	732,285
Tier 2A	$W_j = .08085\ W_p + 21{,}826$	274,586
Tier 2B	$W_j = .08085\ W_p + 65{,}478$	823,758
Tier 3	$W_j = .08085\ W_p + 21{,}865$	
Stage Constant	10,719	
Planet Braking		
Tier 1	$W_j = .08085\ W_p + 17{,}470$	244,095
Tier 2	$W_j = .08085\ W_p + 21{,}826$	274,586
Tier 3	$W_j = .08085\ W_p + 21{,}826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = .08085\ W_p + 17{,}470$	244,095
Tier 2	$W_j = .08085\ W_p + 21{,}826$	274,586
Tier 3	$W_j = .08085\ W_p + 21{,}826$	
Stage Constant	5,259	

nuclear engine for each arrive and depart Mars (or Venus) stage and a cluster of three nuclear engines for the depart Earth stage (3-1-1 configuration). It is noted that these scaling laws do not include the nuclear engine weight; the nuclear engine weight and performance parameters are given in Section II.

Launch Azimuth Constraints

Due to safety restrictions imposed on any given launch site, allowable firing sectors are set up and all vehicle launches must be restricted to pass over only these sectors. These sectors are primarily established from the ground rule that during suborbital flight the vehicle must not pass over any inhabited land mass. For any launch site, the allowable firing sector sets the launch azimuth limits which in turn sets the maximum achievable parking orbit inclination. In order to achieve the declination of any departure hyperbolic asymptote for launches out of a parking orbit without resorting to plane change maneuvers, the inclination of the parking orbit must be equal to or greater than the declination of the departure hyperbolic asymptote. Therefore, for the launch azimuth limits set by the ETR allowable firing sector there will be some maximum achievable parking orbit inclination (or declination of the departure hyperbolic excess velocity). The nominal allowable firing sector for ETR is restricted to a region of the Atlantic bounded by the Caribbean Islands and North America. The approximate launch azimuth range associated with this sector is 44^{o} to 114^{o}. However, for most recent launches it has been required that the vehicle not pass over Europe during the launch or first orbit. This restriction reduces the launch azimuth range to approximately 72^{o} to 114^{o}.

Together with the latitude of ETR (approximately 28.4^{o}), the azimuth range defines the range of ascent trajectory and parking orbit inclinations that are achievable. The departure declinations which can be achieved from a given parking orbit without plane change maneuvers range from zero up to the maximum achievable orbit inclination. For the nominal azimuth constraints, the maximum achievable declination is 54.4^{o} (at 44^{o} launch azimuth). For the reduced azimuth range which misses Europe, the maximum achievable declination is 36.6^{o} (at 114^{o} launch azimuth). The latter launch azimuth range was selected as the limiting guideline throughout the analyses. Therefore, for those missions and opportunities for which the optimum (minimum weight) trajectories require Earth departure declinations that exceed the allowable limits, the optimum, opposite type of outbound trip was used, i.e., type I in lieu of type II.

TASKS A1, A2, AND A3 LANDER MISSION ANALYSIS

The first three tasks (A1, A2 and A3) performed during Phase A of the study involved the mission analysis of Mars stopover lander missions employing respectively Venus swingby, opposition class, and conjunction class trajectory profiles. Since the analysis objectives and basic guidelines for each of these tasks were identical and the results obtained in each task were to be compared with the analogous results obtained in the other two tasks, it is appropriate that the discussions for all three tasks be combined in this section of the report.

Task Description

Tasks A1, A2, and A3 involved the determination of the minimum vehicle weight requirements in Earth orbit for launch of manned Mars lander missions employing each of three types of trajectory profiles. The use of 200,000-pound thrust nuclear engines was investigated in a 3-1-1 connecting mode configuration as well as chemical propulsion systems. The minimum weight vehicle was to be determined consistent with the specified engine firing time limit and launch azimuth constraints.

The matrix of mission, trajectory, and vehicle types which were analyzed in the lander mission analysis tasks in order to evaluate and compare the Venus swingby, opposition class, and conjunction class trajectory profiles are shown below. The nomenclature used for the swingby trajectories (types 3 and 5) is based on the work of Ross and Gillespie (Reference 7); the types II and A, direct leg trajectories, are the short trips (less than 180°) and the I and B trajectories are the long trips (greater than 180°, less than 360°). Only circular Mars parking orbits were analyzed in these three tasks and a 30-day stopover time was assumed.

Mars Lander Mission Matrix

Mission Type	Launch Opportunity	Trajectory Type	Vehicle Mode
Venus Swingby	1980-1993	Swingby Leg - 3 and 5 Direct Leg - I, II, A and B	NNNA and CCCA
Opposition Class	1980-1993	IIB	NNNA, NNNS(15), CCCA, and CCCS(15)
Conjunction Class	1983	IA	NNSA and CCSA

Analysis

For each of the combinations of mission, trajectory, and vehicle types shown above, the required minimum initial vehicle weight was determined using the SWOP program. In addition to the initial vehicle weight, the planetary departure and arrival dates, the total trip time, and the relative Earth arrival velocity were tabulated from the program output. (These data are listed later in this section). The engine firing time also was noted for all nuclear engine propelled stages.

Firing Time Constraint - All noted nuclear engine firing times were compared with the 2700 sec limit: none of the nuclear engines in any of the vehicle stages of the missions analyzed required firing times in excess of this limit.

Launch Azimuth Constraint - For each of the missions analyzed, the trajectory characteristics output by SWOP were used to compute the associated declination of the Earth departure hyperbolic asymptote. The declinations were then compared with the 36.6 deg maximum limitation dictated by the ETR launch azimuth constraints to determine the missions which violate this constraint. Table III-2 presents those missions for which the declinations exceeded 36.6° or were very close to this limit. The range of declination values given for each mission is the range of values obtained for the various vehicle configurations analyzed for each mission type.

Table III-2. Missions Which Violate Launch Azimuth Constraints

Mission Type	Trajectory Type	Launch Opportunity	Declination (deg)
Opposition Class	IIB	1980	+39.9 to +55.3
		1984	-35.9 to -37.0
		1986	-50.1 to -51.5
		1990	+37.6 to +41.6
		1993	+43.7 to +49.9
Venus Swingby	II5	1984	-36.0
	5A, 5B	1990	-64.0

For each of the above opposition class missions that violated the launch azimuth constraints, the opposite type of outbound trajectory, i.e, type I, was analyzed. In all cases, this alternative trajectory type satisfied the launch azimuth constraints. For the Venus swingby missions that violated the constraints the opposite type of outbound trajectory already was specified in the matrix of cases to be analyzed.

Results and Discussion

The data obtained in the analyses of lander missions employing Venus swingby trajectories are given in Table III-3. The data are given successively for each launch opportunity from 1980 to 1993 and for the various trajectory types and vehicle configurations analyzed for each opportunity. The data include, in addition to the initial vehicle weight, the total trip time, the relative Earth arrival velocity, and the planetary departure and arrival dates.

Table III-3. Venus Swingby Lander Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE	INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1980 Venus Outbound Swingby, Type 3A	CCCA	4.123	559	11.49	3838	4179	4397
	NNNA	2.136	549	11.52	3839	4172	4388
Type 3B	CCCA	3.074	673	14.18	3840	4203	4514
	NNNA	1.905	673	14.17	3840	4202	4513
1982 Venus Inbound Swingby, Type I3	CCCA	2.835	618	11.90	4931	5249	5549
	NNNA	1.597	618	11.90	4931	5249	5549
Type II3	CCCA	3.321	569	11.93	4978	5235	5546
	NNNA	1.785	568	11.92	4979	5236	5546
1984 Venus Outbound Swingby, Type 5A	CCCA	7.416	511	14.42	5513	5834	6024
	NNNA	3.103	502	14.26	5512	5835	6014
Type 5B	CCCA	5.336	575	14.76	5514	5847	6089
	NNNA	2.545	574	14.73	5514	5845	6087
1984 Venus Inbound Swingby, Type I5	CCCA	4.827	559	11.58	5650	5938	6209
	NNNA	2.222	571	11.46	5645	5934	6215
** Type II5	CCCA	5.236	486	11.46	5733	5944	6219
	NNNA	2.238	480	11.56	5731	5941	6211
1986 Venus Outbound Swingby, Type 3A	CCCA	2.910	601	11.75	6148	6550	6750
	NNNA	1.825	599	11.72	6148	6547	6747
Type 3B	CCCA	2.931	644	12.35	6149	6554	6793
	NNNA	1.829	642	12.30	6149	6551	6790
1988 Venus Inbound Swingby, Type I3	CCCA	2.756	609	11.75	7292	7546	7901
	NNNA	1.574	605	11.75	7296	7546	7901
Type II3	CCCA	2.593	563	11.75	7338	7551	7901
	NNNA	1.527	563	11.75	7338	7549	7901
**1990 Venus Outbound Swingby, Type 5A	CCCA	6.545	457	11.24	7845	8130	8302
	NNNA	2.875	455	11.30	7843	8126	8298
** Type 5B	CCCA	5.463	586	12.54	7849	8140	8435
	NNNA	2.532	584	12.47	7848	8138	8433
1993 Venus Outbound Swingby, Type 3A	CCCA	4.028	560	11.65	8507	8846	9067
	NNNA	2.113	559	11.66	8507	8844	9066
Type 3B	CCCA	3.895	611	11.15	8507	8843	9118
	NNNA	2.089	614	11.25	8506	8839	9120

*JULIAN DATE MINUS 2,440,000 DAYS

**THESE MISSIONS VIOLATE LAUNCH AZIMUTH CONSTRAINTS

Table III-4 presents the same results for the opposition class lander missions employing a type IIB trajectory. As indicated in this table and in Table III-2, the missions for certain launch opportunities violate the launch azimuth constraints; in these cases the mission was analyzed for a type IB trajectory and the results are given in Table III-5.

Table III-4. Opposition Class IIB Lander Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE	INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
**1980 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	2.181	488	20.67	4184	4465	4672
	NNNS(15)	2.971	476	20.49	4186	4455	4662
	CCCA	4.731	484	20.67	4188	4465	4672
	CCCS(15)	7.798	486	20.64	4184	4464	4670
1982 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	1.950	461	18.05	4968	5214	5429
	NNNS(15)	2.379	462	17.98	4960	5206	5422
	CCCA	4.358	462	18.06	4968	5215	5430
	CCCS(15)	5.244	463	18.02	4964	5211	5426
**1984 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	1.848	456	16.10	5749	5975	6205
	NNNS(15)	1.920	453	16.03	5747	5972	6200
	CCCA	3.790	456	16.10	5749	5975	6205
	CCCS(15)	4.098	454	16.04	5747	5973	6201
**1986 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	1.631	460	16.10	6541	6750	7001
	NNNS(15)	1.782	446	15.34	6534	6734	6981
	CCCA	3.153	462	16.23	6541	6753	7004
	CCCS(15)	3.384	452	15.60	6536	6741	6988
1988 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	1.679	464	18.94	7338	7547	7802
	NNNS(15)	2.052	474	16.77	7297	7513	7771
	CCCA	3.376	465	19.05	7338	7550	7803
	CCCS(15)	4.178	468		7314	7527	7782
**1990 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	2.056	459	22.05	8131	8354	8590
	NNNS(15)	3.389	456	19.48	8103	8312	8559
	CCCA	4.765	460	22.06	8131	8355	8591
	CCCS(15)	8.479	452	20.05	8113	8323	8565
**1993 Mars Opposition, Type IIB (30 Day Stopover)	NNNA	2.413	460	22.56	8901	9139	9361
	NNNS(15)	4.266	445	20.98	8889	9106	9334
	CCCA	5.577	464	22.82	8902	9147	9366
	CCCS(15)	12.639	449	21.41	8892	9115	9342

*JULIAN DATE MINUS 2,440,000 DAYS

**THESE MISSIONS VIOLATE LAUNCH AZIMUTH CONSTRAINTS

Table III-5. Opposition Class IB Lander Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE	INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1980 Mars Opposition, Type IB (30 Day Stopover)	NNNA	2.211	508	20.74	4168	4471	4676
	NNNS(15)	3.072	504	20.46	4157	4452	4661
	CCCA	4.777	509	20.78	4170	4474	4679
	CCCS(15)	8.104	506	20.53	4158	4456	4664
1984 Mars Opposition, Type IB (30 Day Stopover)	NNNA	1.854	527	16.15	5682	5978	6209
	NNNS(15)	1.959	523	16.05	5679	5974	6202
	CCCA	3.715	528	16.19	5683	5981	6211
	CCCS(15)	3.951	524	16.09	5681	5976	6205
1986 Mars Opposition, Type IB (30 Day Stopover)	NNNA	1.944	534	14.98	6455	6737	6989
	NNNS(15)	1.944	534	14.98	6455	6737	6989
	CCCA	3.806	534	14.99	6460	6742	6994
	CCCS(15)	3.806	534	14.99	6460	6742	6994
1990 Mars Opposition, Type IB (30 Day Stopover)	NNNA	2.510	528	21.64	8057	8348	8586
	NNNS(15)	3.395	534	16.58	7987	8249	8521
	CCCA	5.507	527	21.62	8058	8346	8585
	CCCS(15)	7.961	522	18.10	8020	8281	8542
1993 Mars Opposition, Type IB (30 Day Stopover)	NNNA	2.726	527	22.61	8835	9141	9362
	NNNS(15)	4.824	536	19.39	8776	9064	9312
	CCCA	6.293	528	23.17	8844	9160	9372
	CCCS(15)	12.620	526	20.47	8802	9091	9328

*JULIAN DATE MINUS 2,440,000 DAYS

Finally, analogous results are presented for the conjunction class lander mission in Table III-6.

Table III-6. Conjunction Class IA Lander Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE	INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1983 Mars Conjunction, Type IA (Stopover time 416 Days)	NNSA CCSA	1.638 2.496	956 956	12.16 12.16	4937 4937	5654 5654	5893 5893

*JULIAN DATE MINUS 2,440,000 DAYS

A graphical presentation of the vehicle weight data obtained for the lander missions using Venus swingby trajectories is given in Figure III-1 for the two vehicle modes analyzed. Of all the missions (trajectory types) shown, only the 1984 II5 and the 1990 5A and 5B violated the launch azimuth constraints established in this study. The use of the opposite type of Venus swingby trajectory (type 5 inbound) for the 1990 launch opportunity was not possible since it was not feasible to match this inbound swingby trajectory with either the type I or II direct outbound trajectories.

In all but two launch opportunities the minimum weight vehicle is obtained when the swingby trajectory (type 3 or 5) is combined with the long direct leg (type I or B); in the years 1986 and 1988 the use of the short direct leg (type A and II, respectively) results in a mission with both a lower weight vehicle and a shorter total trip time. The minimum vehicle weights required for each of the acceptable launch opportunities in the 1980 to 1993 time period range from about 1.53 to 2.22 million pounds.

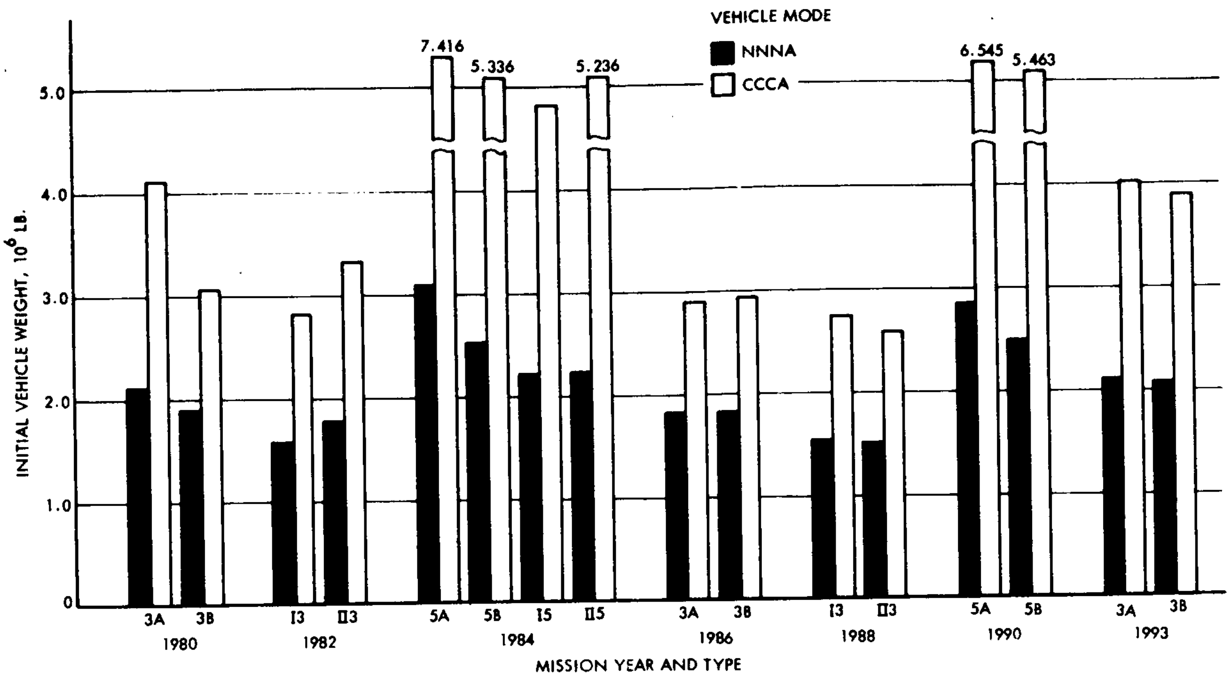


Figure III-1. Venus Swingby Lander Mission

Figure III-2 graphically presents the results obtained in the opposition class lander mission analysis for the acceptable launch opportunities in the period from 1980 to 1993. A type IB trajectory was employed for the 1980, 1984, 1986, 1990, and 1993 launch opportunities since the type IIB trajectories required depart Earth declinations which were outside of the -36.6 to +36.6 degree allowable range.

These results are typical of those obtained in earlier analyses in which similar mission and vehicle parameters were assumed, viz: 1) the assumption of a maximum of 15 km/sec Earth aerodynamic braking capability increases the initial vehicle weight requirements by a factor of 1.5 to 2 over the weight required for unlimited aerodynamic braking for the unfavorable launch opportunities in the synodic cycle; 2) the use of the 200,000-pound thrust nuclear engine results in vehicles weighing approximately 50 percent of those utilizing chemical cryogenic stages; and 3) the weights for the nuclear engine propelled vehicles range from a minimum of 1.68 million pounds to over 3 million pounds depending upon the assumed Earth aerodynamic braking capability and mission year.

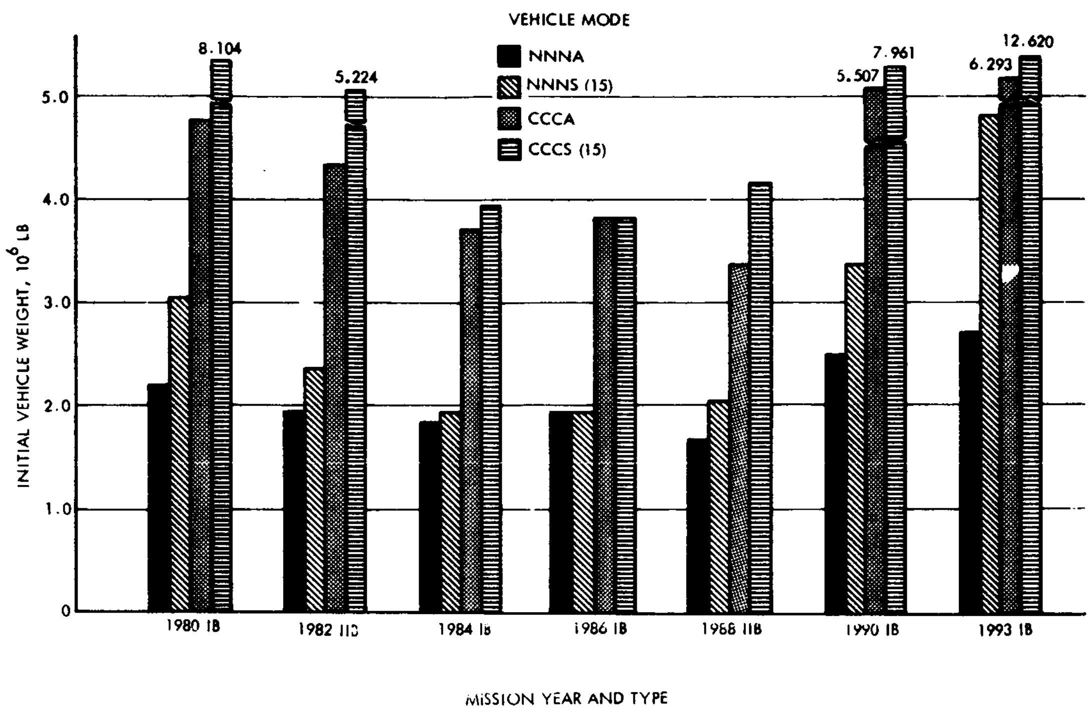


Figure III-2. Mars Opposition Class Lander Mission

The data on Figure III-3 compare the NNNA vehicle configuration weights required for missions using opposition class and Venus swingby trajectories in each of the launch opportunities from 1980 to 1993; a 1983 IA NNSA conjunction class mission is included for comparison. The trajectory types used for the opposition class missions are either type IB or IIB as shown previously on Figure III-2. The trajectory types for the swingby missions are those which yield the minimum weight vehicle in each opportunity; however, it should be noted that the minimum swingby mission result for 1990 corresponds to a trajectory that violates the launch azimuth constraints (type 5B).

In all years except 1984 and 1990 the Venus swingby mission requires a lower weight vehicle than does the acceptable opposition class mission; the minimum vehicle weights range from 1.53 to 2.51 million pounds. The conjunction class mission requires a vehicle weighing slightly more (1.64 million pounds) than that for the minimum Venus swingby mission (1988, 1.53 million pounds) although, it must be recalled that the vehicle for the conjunction class mission employs a storable propulsion system (NNSA configuration) for departing Mars.

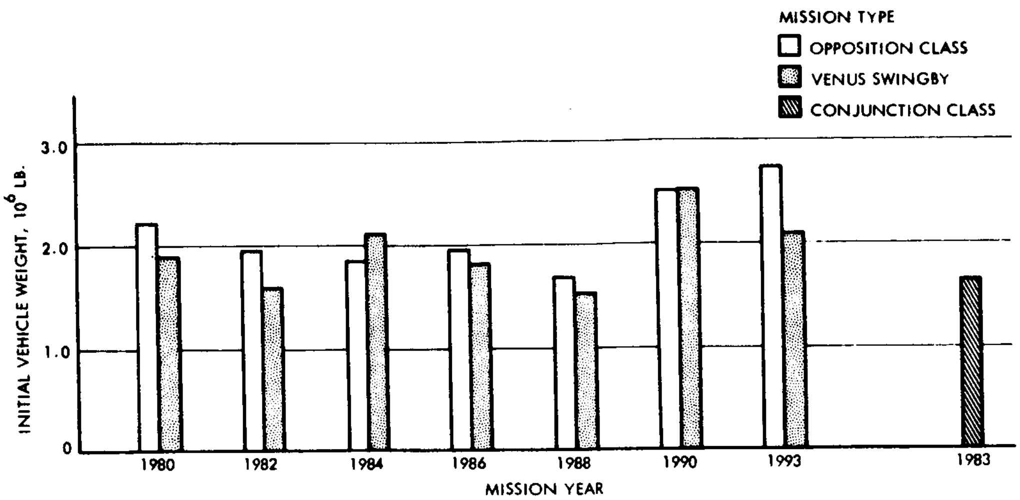


Figure III-3. Lander Mission Mode Comparison

TASK A4 ORBITAL CAPTURE MISSION ANALYSIS

Task Description

This final task in the first phase of this study involved the determination of the minimum vehicle weight requirements in Earth orbit for launch of manned orbital capture missions to either Mars or Venus. As in the initial three tasks in Phase A, the use of a 200,000-pound thrust nuclear engine was investigated in a 3-1-1 connecting mode configuration as well as chemical propulsion systems. The minimum weight vehicle was to be determined consistent with the specified engine firing time limit and launch azimuth constraints.

The matrix of mission, trajectory, and vehicle types which were analyzed in the orbital capture mission analysis task is shown in Table III-7. Only circular Mars parking orbits were analyzed in this task and the planetary stopover time was varied from 20 to 40 days for all missions.

Table III-7. Orbital Capture Mission Matrix

Target Planet	Mission Type	Launch Opportunity	Trajectory Type	Vehicle Mode
Mars	Opposition Class	1980-1984	IIB	NNNA, NNSA, NNNS(15), CCCA, and CCCS(15)
	Venus Swingby	1980-1984	Swingby - 3 and 5 Direct Leg - I, II, A and B	NNNA, NNSA, NCCA, CCCA, and CCSA
Venus	Inferior Conjunction	1980-1985	Outbound - I and II Inbound - A and B	NNNA, NNSA, NCCA, NSSA, CCCA, and CCSA

Analysis

For each of the combinations of mission, trajectory, and vehicle types in Table III-7, the required minimum initial vehicle weight was determined for planetary stopover periods of 20, 30, and 40 days using the SWOP program. As in the initial three tasks of this study phase, the vehicle weight, various trajectory parameters, and nuclear engine firing time were tabulated from the program output.

Firing Time Constraints - All of the noted nuclear engine firing times were compared with the 2700 sec limit. The results showed that the firing time exceeded this limit by a few hundred seconds in only two mission cases, the NNSA and the NCCA vehicle configurations for the 1984 5A Venus swingby trajectory.

Launch Azimuth Constraint - As in the case of the opposition class IIB lander missions, the Earth departure declinations for the opposition class IIB orbital capture missions in the years 1980 and 1984 exceeded 36.6° or were very close to this limit. In these cases the mission was re-analyzed for the opposite type (IB) trajectory.

Also, as in the case of the Venus swingby lander mission, the declination for the 1984 II5 Venus swingby orbital capture mission was very close to the 36.6° limit. None of the minimum weight Venus orbital capture missions (type IIB) violated the launch azimuth constraints.

Results and Discussions

The data obtained for the Mars orbital capture mission analyses are given in Tables III-8 through III-10. Tables III-8 and III-9 are for the Mars missions using respectively Venus swingby and opposition class IIB trajectories; Table III-10 is for the 1980 and 1984 opposition class IB trajectory missions.

Table III-8. Venus Swingby Orbital Capture Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE		INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1980 Venus Outbound Swingby, Type 3A, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.834	538	11.72	3839	4162	4376
		(30)	1.895	548	11.53	3839	4171	4387
		(40)	1.963	559	11.49	3839	4180	4398
	NNSA	(20)	1.977	538	11.73	3839	4161	4376
		(30)	2.098	546	11.55	3838	4169	4385
		(40)	2.255	555	11.50	3835	4173	4390
	NCCA	(20)	2.154	545	11.57	3838	4169	4383
		(30)	2.323	558	11.49	3838	4176	4397
		(40)	2.607	563	11.51	3838	4185	4401
	CCCA	(20)	3.151	545	11.58	3838	4168	4383
		(30)	3.445	559	11.49	3838	4176	4397
		(40)	3.756	562	11.50	3838	4184	4401
	CCSA	(20)	3.682	541	11.66	3838	4164	4379
		(30)	4.099	550	11.51	3839	4172	4389
		(40)	4.491	560	11.50	3839	4182	4399
1980 Venus Outbound Swingby, Type 3B, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.632	667	13.89	3841	4192	4508
		(30)	1.552	673	14.15	3841	4201	4513
		(40)	1.564	678	14.44	3841	4211	4519
	NNSA	(20)	1.682	667	13.89	3841	4192	4508
		(30)	1.700	673	14.16	3841	4201	4513
		(40)	1.717	678	14.45	3841	4212	4519
	NCCA	(20)	1.714	668	13.91	3840	4193	4508
		(30)	1.735	673	14.17	3840	4203	4513
		(40)	1.751	679	14.48	3841	4212	4519
	CCCA	(20)	2.504	668	13.90	3840	4192	4508
		(30)	2.539	673	14.16	3840	4202	4513
		(40)	2.572	679	14.47	3840	4212	4519
	CCSA	(20)	2.838	668	13.90	3840	4192	4508
		(30)	2.881	673	14.17	3840	4202	4513
		(40)	2.921	679	14.46	3840	4212	4519
1982 Venus Inbound Swingby, Type 13, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.421	615	11.90	4934	5246	5549
		(30)	1.436	618	11.90	4931	5248	5549
		(40)	1.453	621	11.89	4929	5252	5550
	NNSA	(20)	1.905	615	11.90	4933	5246	5549
		(30)	1.853	618	11.90	4931	5246	5549
		(40)	1.950	622	11.89	4928	5250	5549
	NCCA	(20)	1.799	616	11.90	4933	5246	5549
		(30)	1.827	619	11.90	4930	5248	5549
		(40)	1.864	622	11.89	4928	5252	5550
	CCCA	(20)	2.436	615	11.90	4934	5247	5549
		(30)	2.484	618	11.89	4931	5249	5549
		(40)	2.565	621	11.89	4929	5253	5550
	CCSA	(20)	2.906	615	11.90	4933	5246	5549
		(30)	2.963	618	11.89	4931	5248	5549
		(40)	3.033	622	11.89	4928	5252	5550

*JULIAN DATE MINUS 2,440,000 DAYS

Table III-8. Venus Swingby Orbital Capture Mission Analysis Results (Continued)

MISSION DESCRIPTION	PROPULSION MODE	INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1982 Venus Inbound Swingby, Type II3, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA (20)	1.550	564	11.93	4982	5232	5546
	(30)	1.536	568	11.92	4978	5236	5547
	(40)	1.526	575	11.92	4972	5237	5547
	NNSA (20)	2.153	564	11.92	4983	5235	5546
	(30)	2.121	568	11.91	4979	5238	5547
	(40)	2.102	574	11.91	4973	5239	5547
	NCCA (20)	2.120	563	11.92	4984	5235	5546
	(30)	2.098	567	11.91	4980	5239	5547
	(40)	2.100	572	11.91	4976	5241	5548
	CCCA (20)	2.998	565	11.94	4981	5231	5545
	(30)	2.936	569	11.92	4978	5236	5547
	(40)	2.901	575	11.92	4971	5236	5547
	CCSA (20)	3.585	565	11.93	4981	5232	5546
	(30)	3.481	569	11.92	4978	5236	5547
	(40)	3.438	575	11.92	4972	5237	5547
1984 Venus Outbound Swingby, Type 5A, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA (20)	2.687	496	13.99	5513	5829	6009
	(30)	2.782	502	14.20	5512	5834	6013
	(40)	2.789	510	14.20	5493	5825	6003
	NNSA (20)	4.015	494	13.91	5513	5827	6007
	(30)	4.131	500	14.11	5495	5815	5995
	(40)	4.953	514	14.44	5497	5825	6011
	NCCA (20)	4.693	497	14.06	5513	5830	6010
	(30)	4.218	512	14.44	5513	5834	6025
	(40)	5.561	514	14.47	5497	5825	6012
	CCCA (20)	7.290	497	14.07	5513	5831	6010
	(30)	6.507	511	14.42	5512	5834	6024
	(40)	8.942	515	14.44	5497	5825	6012
	CCSA (20)	8.737	497	14.02	5513	5829	6009
	(30)	9.540	502	14.24	5512	5835	6014
	(40)	9.355	509	14.24	5492	5825	6001
1984 Venus Outbound Swingby, Type 5B, Circular Parking Orbit (20, 30, 40 Day Stopover).	NNNA (20)	2.221	565	14.59	5514	5834	6079
	(30)	2.250	573	14.72	5514	5844	6087
	(40)	2.281	581	14.84	5514	5854	6095
	NNSA (20)	2.425	565	14.58	5513	5833	6078
	(30)	2.482	572	14.71	5513	5843	6086
	(40)	2.545	580	14.83	5513	5853	6093
	NCCA (20)	2.715	567	14.63	5514	5837	6081
	(30)	2.790	575	14.75	5514	5847	6089
	(40)	2.871	583	14.87	5514	5857	6097
	CCCA (20)	4.265	567	14.62	5514	5837	6081
	(30)	4.388	575	14.75	5514	5847	6088
	(40)	4.521	583	14.87	5514	5856	6096
	CCSA (20)	5.039	567	14.61	5514	5836	6080
	(30)	5.229	574	14.74	5514	5846	6088
	(40)	5.395	582	14.86	5514	5855	6096
1984 Venus Inbound Swingby, Type I5, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA (20)	1.876	560	11.51	5652	5930	6212
	(30)	2.006	571	11.51	5641	5931	6212
	(40)	2.244	583	11.47	5632	5933	6215
	NNSA (20)	2.419	567	11.45	5649	5926	6215
	(30)	2.610	579	11.45	5636	5927	6215
	(40)	2.841	593	11.45	5623	5927	6215
	NCCA (20)	2.530	568	11.53	5642	5930	6211
	(30)	2.765	584	11.48	5629	5931	6214
	(40)	3.049	597	11.48	5617	5931	6214
	CCCA (20)	3.682	558	11.47	5657	5934	6215
	(30)	4.163	564	11.52	5648	5935	6212
	(40)	4.874	579	11.48	5636	5937	6215
	CCSA (20)	4.373	559	11.51	5653	5930	6212
	(30)	5.056	572	11.46	5643	5932	6215
	(40)	5.819	581	11.52	5631	5934	6212
** 1984 Venus Inbound Swingby, Type II5, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA (20)	1.829	476	11.55	5734	5936	6210
	(30)	1.977	481	11.56	5729	5939	6210
	(40)	2.291	490	11.52	5724	5943	6213
	NNSA (20)	2.513	479	11.55	5731	5931	6210
	(30)	2.854	488	11.50	5725	5934	6213
	(40)	3.219	498	11.48	5717	5936	6214
	NCCA (20)	2.943	475	11.57	5735	5939	6210
	(30)	3.419	484	11.52	5729	5942	6213
	(40)	4.146	487	11.62	5722	5945	6209
	CCCA (20)	4.006	475	11.56	5736	5939	6210
	(30)	4.693	482	11.52	5732	5943	6213
	(40)	5.695	484	11.60	5726	5946	6210
	CCSA (20)	4.778	478	11.52	5734	5936	6212
	(30)	5.645	486	11.48	5729	5940	6215
	(40)	6.968	489	11.54	5724	5944	6212

* JULIAN DATE MINUS 2,440,000 DAYS

** THESE MISSIONS VIOLATE LAUNCH AZIMUTH CONSTRAINTS

Table III-9. Opposition Class IIB Orbital Capture Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE		INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE *	TARGET PLANET DEPART DATE *	EARTH ARRIVAL DATE *
** 1980 Mars Opposition, Orbital Capture, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.932	476	20.51	4187	4454	4663
		(30)	2.027	483	20.63	4186	4462	4669
		(40)	2.068	495	20.79	4184	4474	4679
	NNNS(15)	(20)	2.602	466	20.23	4184	4436	4650
		(30)	2.813	467	20.38	4190	4447	4657
		(40)	3.039	469	20.46	4191	4453	4660
	NNSA	(20)	2.858	470	20.38	4187	4443	4657
		(30)	3.016	478	20.51	4186	4454	4664
		(40)	3.286	479	20.58	4188	4458	4667
	CCCA	(20)	4.015	477	20.54	4188	4455	4665
		(30)	4.361	484	20.66	4187	4464	4671
		(40)	4.610	494	20.76	4185	4475	4679
	CCCS(15)	(20)	6.785	470	20.38	4187	4445	4657
		(30)	7.521	472	20.50	4191	4455	4663
		(40)	8.181	493	20.76	4184	4474	4677
1982 Mars Opposition, Orbital Capture, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.731	455	17.96	4965	5202	5420
		(30)	1.771	463	18.03	4964	5211	5427
		(40)	1.815	471	18.11	4964	5220	5435
	NNNS(15)	(20)	1.927	453	17.92	4962	5198	5415
		(30)	2.244	461	17.98	4960	5205	5421
		(40)	2.313	469	18.06	4960	5215	5429
	NNSA	(20)	2.522	456	17.90	4958	5195	5414
		(30)	2.768	462	17.99	4961	5205	5423
		(40)	2.929	472	18.05	4958	5214	5430
	CCCA	(20)	3.666	456	17.97	4965	5203	5421
		(30)	3.955	464	18.04	4965	5213	5429
		(40)	4.192	472	18.12	4964	5222	5436
	CCCS(15)	(20)	4.527	454	17.94	4963	5200	5417
		(30)	4.806	462	18.01	4963	5210	5425
		(40)	5.155	470	18.08	4962	5218	5432
** 1984 Mars Opposition. Orbital Capture, Type IIB. Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.636	446	15.95	5748	5964	6194
		(30)	1.669	456	16.08	5748	5974	6204
		(40)	1.702	465	16.23	5748	5983	6213
	NNNS(15)	(20)	1.699	443	15.90	5747	5961	6190
		(30)	1.741	452	16.00	5746	5970	6198
		(40)	1.788	460	16.11	5746	5978	6206
	NNSA	(20)	2.269	443	15.87	5746	5958	6189
		(30)	2.356	451	15.96	5744	5964	6195
		(40)	2.459	460	16.08	5744	5974	6204
	CCCA	(20)	3.210	447	15.96	5748	5964	6195
		(30)	3.371	456	16.09	5748	5974	6204
		(40)	3.541	466	16.24	5748	5984	6214
	CCCS(15)	(20)	3.423	445	15.91	5746	5962	6191
		(30)	3.618	454	16.02	5746	5971	6200
		(40)	3.914	463	16.16	5745	5980	6208

*JULIAN DATE MINUS 2,440,000 DAYS

**THESE MISSIONS VIOLATE LAUNCH AZIMUTH CONSTRAINTS

Table III-10. Opposition Class IB Orbital Capture Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE		INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1980 Mars Opposition, Orbital Capture, Type IB, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.989	501	18.57	4166	4457	4667
		(30)	2.063	507	20.64	4163	4462	4670
		(40)	2.124	514	20.76	4163	4473	4677
	NNNS(15)	(20)	2.782	497	20.21	4152	4435	4649
		(30)	2.888	504	20.40	4154	4448	4658
		(40)	3.040	511	20.51	4152	4455	4663
	NNSA	(20)	2.928	501	20.31	4154	4439	4655
		(30)	3.107	507	20.43	4153	4448	4660
		(40)	3.380	516	20.56	4150	4456	4666
	CCCA	(20)	4.058	501	20.64	4169	4462	4670
		(30)	4.372	509	20.75	4168	4472	4677
		(40)	4.778	516	20.83	4165	4478	4681
	CCCS(15)	(20)	7.038	499	20.40	4159	4448	4658
		(30)	7.717	506	20.50	4157	4454	4663
		(40)	8.494	513	20.60	4155	4461	4668
1984 Mars Opposition, Orbital Capture, Type IB, Circular Parking Orbit (20, 30, 40 Day Stopover)	NNNA	(20)	1.570	515	15.99	5682	5967	6197
		(30)	1.602	525	16.12	5682	5977	6207
		(40)	1.636	535	16.29	5681	5986	6216
	NNNS(15)	(20)	1.636	512	15.89	5677	5960	6189
		(30)	1.680	521	15.97	5675	5968	6196
		(40)	1.797	531	16.12	5675	5978	6206
	NNSA	(20)	2.257	514	15.88	5675	5958	6189
		(30)	2.348	523	15.95	5672	5964	6195
		(40)	2.449	532	16.05	5670	5972	6202
	CCCA	(20)	3.157	517	16.01	5682	5969	6199
		(30)	3.314	527	16.14	5681	5978	6208
		(40)	3.483	536	16.31	5681	5987	6217
	CCCS(15)	(20)	3.354	514	15.95	5680	5965	6194
		(30)	3.549	523	16.05	5679	5973	6202
		(40)	3.790	533	16.17	5677	5981	6210

*JULIAN DATE MINUS 2,440,000 DAYS

Table III-11 is for the Venus type IIB missions. (The type IIB Venus missions yielded the minimum weight vehicle of all possible combinations of short and long, outbound and inbound trajectory types for all launch opportunities from 1980 to 1985.)

Table III-11. Venus IIB Orbital Capture Mission Analysis Results

MISSION DESCRIPTION	PROPULSION MODE		INITIAL VEHICLE WEIGHT (10^6 LB)	TOTAL TRIP TIME (DAYS)	EARTH ARRIVAL SPEED (KM/SEC)	EARTH DEPART DATE*	TARGET PLANET DEPART DATE*	EARTH ARRIVAL DATE*
1980 Venus Inferior Conjunction, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stop-over)	NNNA	(20)	1.629	424	13.91	4343	4472	4767
		(30)	1.647	441	13.97	4342	4481	4783
		(40)	1.673	459	13.41	4343	4492	4802
	NNSA	(20)	2.053	424	13.90	4342	4447	4766
		(30)	2.091	440	13.96	4341	4480	4781
		(40)	2.153	457	14.20	4340	4489	4797
	NCCA	(20)	2.273	427	13.92	4346	4474	4771
		(30)	2.333	441	13.99	4345	4483	4786
		(40)	2.408	460	13.49	4344	4492	4804
	NSSA	(20)	3.397	425	13.93	4348	4475	4773
		(30)	3.478	441	14.00	4347	4484	4788
		(40)	3.611	457	13.27	4346	4492	4803
	CCCA	(20)	3.245	425	13.91	4343	4473	4768
		(30)	3.327	441	13.98	4342	4482	4783
		(40)	3.446	461	13.27	4342	4492	4803
	CCSA	(20)	3.958	425	13.91	4343	4472	4768
		(30)	4.063	441	13.98	4342	4481	4783
		(40)	4.209	460	13.46	4342	4492	4802
1982 Venus Inferior Conjunction, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stop-over)	NNNA	(20)	1.568	427	14.00	4928	5057	5355
		(30)	1.590	447	14.22	4929	5068	5375
		(40)	1.609	467	14.44	4930	5080	5397
	NNSA	(20)	2.002	426	13.97	4926	5055	5352
		(30)	2.059	444	14.17	4927	5065	5371
		(40)	2.116	466	14.41	4929	5078	5394
	NCCA	(20)	2.145	430	14.05	4930	5059	5360
		(30)	2.210	449	14.26	4930	5070	5379
		(40)	2.257	469	14.48	4932	5082	5401
	NSSA	(20)	3.159	431	14.07	4931	5061	5362
		(30)	3.249	450	14.28	4932	5072	5382
		(40)	3.312	470		4933	5084	5403
	CCCA	(20)	3.008	429	14.03	4928	5058	5358
		(30)	3.110	448	14.23	4929	5069	5377
		(40)	3.186	468	14.45	4930	5081	5398
	CCSA	(20)	3.586	428	14.01	4928	5057	5356
		(30)	3.805	447	14.22	4928	5068	5376
		(40)	3.897	468	14.45	4930	5080	5398
1983 Venus Inferior Conjunction, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stop-over)	NNNA	(20)	1.517	452	13.72	5497	5645	5949
		(30)	1.451	480	13.97	5493	5657	5974
		(40)	1.456	499	14.19	5494	5668	5993
	NNSA	(20)	1.942	461	13.76	5492	5646	5952
		(30)	1.957	481	13.97	5492	5657	5974
		(40)	1.955	500	14.19	5493	5668	5993
	NCCA	(20)	2.055	463	13.76	5490	5647	5953
		(30)	2.073	483	13.97	5490	5657	5973
		(40)	2.075	502	14.18	5490	5667	5992
	NSSA	(20)	2.990	466	13.76	5487	5647	5953
		(30)	3.012	486	13.96	5487	5657	5973
		(40)	3.010	505	14.18	5487	5667	5992
	CCCA	(20)	2.791	461	13.76	5492	5647	5953
		(30)	2.817	481	13.97	5493	5657	5974
		(40)	2.819	500	14.18	5493	5667	5993
	CCSA	(20)	3.329	461	13.76	5492	5647	5953
		(30)	3.357	481	13.97	5493	5657	5974
		(40)	3.355	500	14.19	5493	5668	5993
1985 Venus Inferior Conjunction, Type IIB, Circular Parking Orbit (20, 30, 40 Day Stop-over)	NNNA	(20)	1.573	442	14.08	6080	6223	6521
		(30)	1.578	459	14.25	6079	6233	6539
		(40)	1.584	476	14.38	6079	6243	6555
	NNSA	(20)	1.990	443	14.06	6076	6222	6519
		(30)	1.991	459	14.25	6080	6233	6539
		(40)	1.992	477	14.38	6079	6243	6555
	NCCA	(20)	2.267	442	14.07	6078	6222	6521
		(30)	2.271	459	14.25	6080	6233	6539
		(40)	2.275	476	14.39	6081	6243	6556
	NSSA	(20)	3.354	441	14.09	6082	6224	6523
		(30)	3.354	457	14.27	6085	6235	654
		(40)	3.358	474	14.40	6085	6245	655
	CCCA	(20)	3.082	443	14.07	6078	6222	6521
		(30)	3.089	460	14.25	6079	6233	6539
		(40)	3.094	477	14.38	6078	6242	6555
	CCSA	(20)	3.746	443	14.07	6078	6222	6521
		(30)	3.749	460	14.25	6079	6233	6539
		(40)	3.751	477	14.38	6078	6242	6555

*JULIAN DATE MINUS 2,440,000 DAYS

Stopover Time Variation - Figure III-4 graphically illustrates the typical variation in initial vehicle weight obtained for the opposition class orbital capture missions when the Martian stopover period is varied from 20 to 40 days. With each 10-day increase in stopover time the initial weight is increased by approximately six to nine percent for each of the vehicle modes analyzed. Similar numerical results were obtained for the opposition class missions in 1980 and 1984.

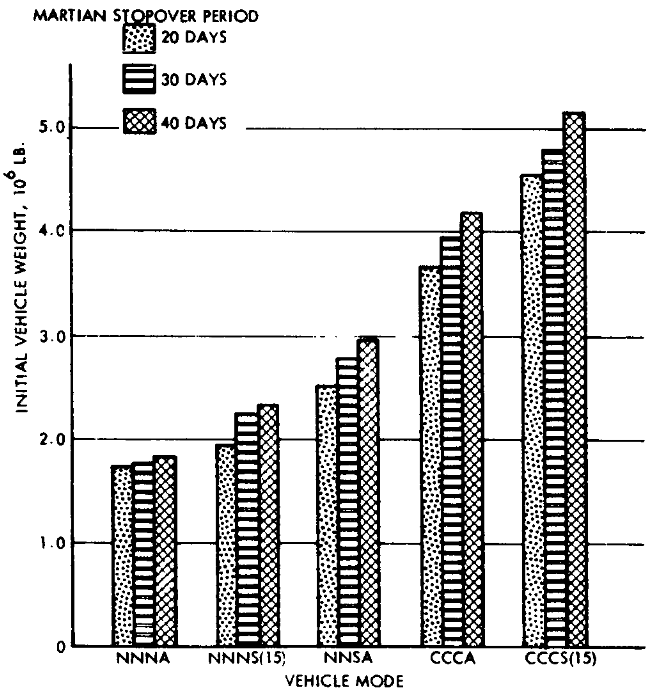


Figure III-4. Mars Orbital Capture Mission—Opposition Class 1982 IIB

This identical numerical initial vehicle weight variation with varying stopover time was obtained also for the type I5 Venus swingby trajectory in 1984 (the type I5 trajectory was the minimum weight Venus swingby trajectory for the 1984 launch opportunity). The minimum weight Venus swingby trajectories in 1980 and 1982 (type 3B and I3, respectively) produced either a very slight increase or a decrease in vehicle weight with varying Martian stopover time; less than one percent change in vehicle weight for a 10-day variation in stopover time.

The variation of initial vehicle weight with varying Venusian stopover time for the Venus orbital capture missions is illustrated on Figure III-5 for a NSSA vehicle configuration. The comparative numerical results for other vehicle configurations employing various combinations of nuclear and chemical propulsion systems are identical to those shown here. In 1980 and 1982 the initial vehicle weight increases by approximately two to four percent with each 10-day increase in stopover time; in 1983 and 1985 there is no significant variation in initial vehicle weight for Venusian stopover periods from 20 to 40 days.

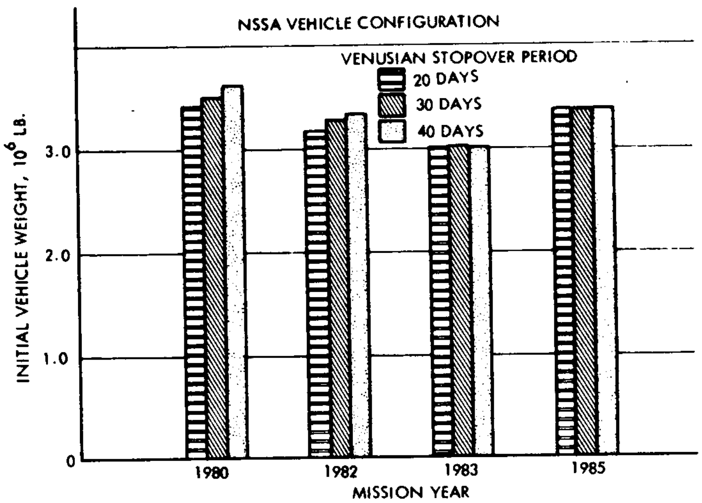


Figure III-5. Venus Orbital Capture Mission Type IIB

<u>Vehicle Mode Variation</u> - Shown in Figure III-6 are the initial vehicle weights for the acceptable opposition class orbital capture missions required for vehicles employing various propulsion systems and Earth aerodynamic braking capabilities. A 30-day stopover period is used in all cases. The initial weight required for the vehicle modes varies in the same proportions as for the opposition class lander missions previously shown under task A2; i. e., the constraint of a 15 km/sec Earth aerodynamic braking capability increases the vehicle weight by a factor of 1.5 to 2, and the nuclear engine propelled vehicles weigh approximately 50 percent less than the chemical cryogenic propelled vehicles.

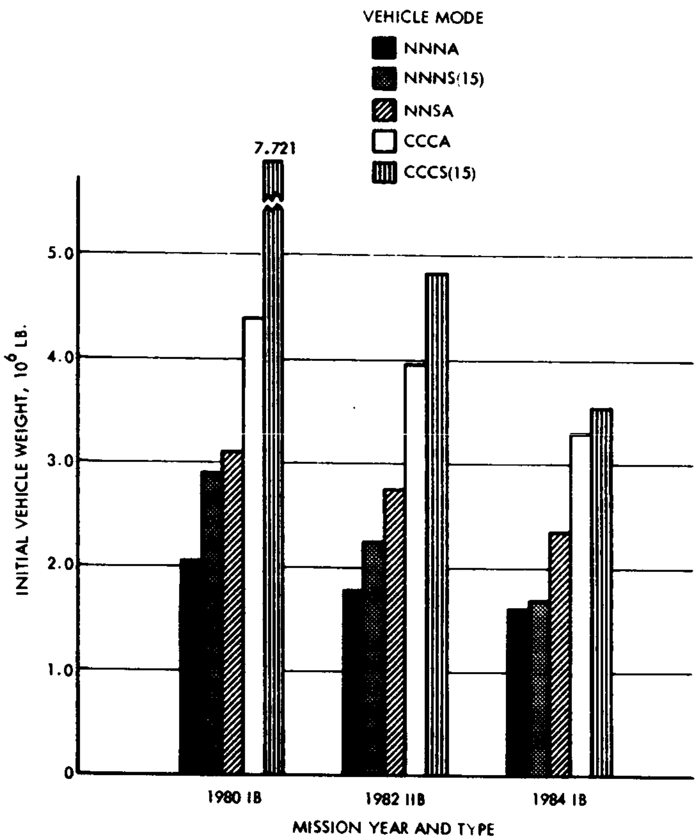


Figure III-6. Mars Orbital Capture Mission—Opposition Class, 30-Day Stopover Period

Figure III-7 presents the same results for Mars orbital capture missions as on the preceding figure except that the data are given for the minimum weight, Venus swingby trajectories for the years 1980 to 1984. In all cases, the minimum weight trajectories are those in which a long direct leg (type I or B) is combined with the swingby leg. The minimum vehicle weights for these three launch opportunities range from about 1.44 to 2.01 million pounds.

Figure III-8 extends the initial vehicle weight comparisons of various vehicle modes to the Venus orbital capture missions. The data are given for a type IIB inferior conjunction trajectory which yields the minimum vehicle weights for all mission opportunities from 1980 to 1985. The vehicle weights for a given vehicle mode vary only slightly in the launch period from 1980 to 1985 with minimum weights obtained for the NNNA vehicle mode ranging from 1.45 to 1.65 million pounds.

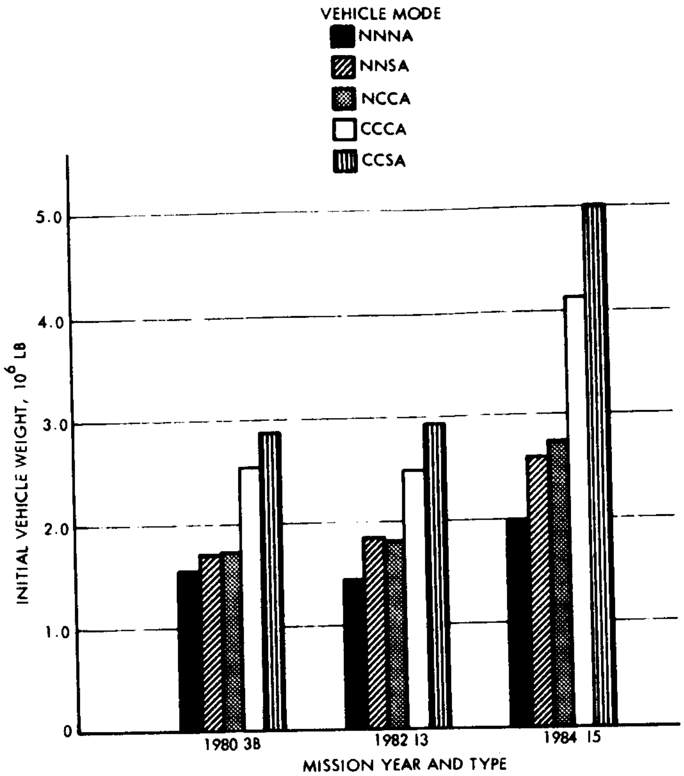


Figure III-7. Mars Orbital Capture Mission—Venus Swingby, 30-Day Stopover Period

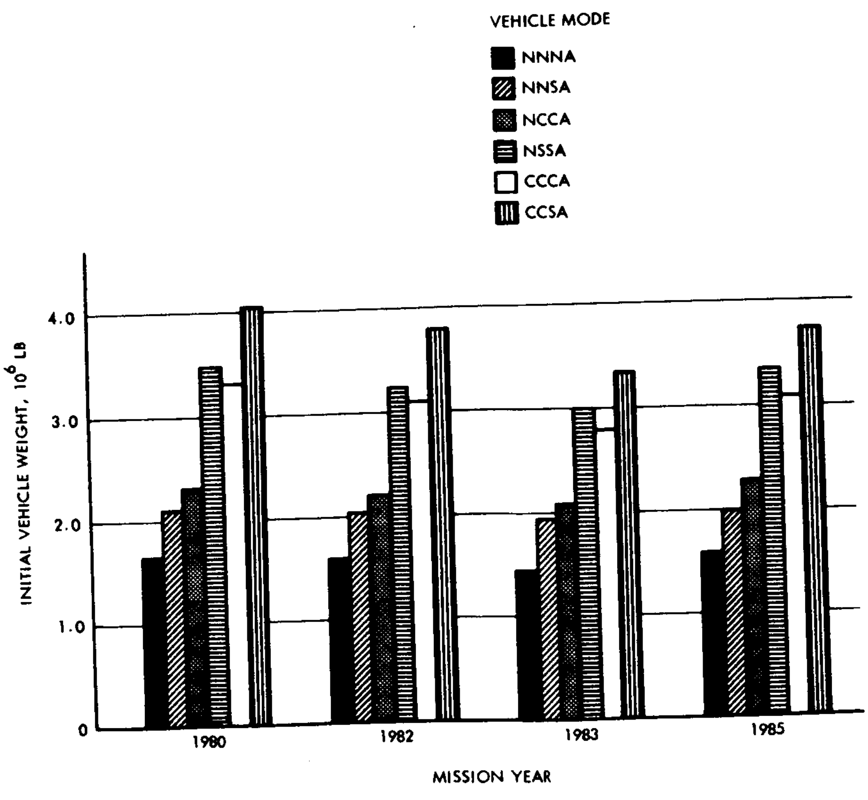


Figure III-8. Venus Orbital Capture Mission Type IIB, 30-Day Stopover Period

Mission Mode Variation - The final graph is this section, Figure III-9, compares the minimum initial vehicle weights obtained for the Mars orbital capture missions employing opposition class and Venus swingby trajectories and the Venus orbital capture missions. For the Mars missions the results are identical to those obtained for the lander mission comparison, i.e., the Venus swingby trajectory yields lower vehicle weights than does the opposition class trajectory in all years except 1984. Furthermore, the minimum vehicle weights required for performing an orbital capture mission to Mars or Venus during each respectively available launch opportunity range from 1.44 to 1.65 million pounds. A comparison of these minimum weight results for the Mars orbital capture missions with the results previously given for the Mars lander missions shows that the lander missions require approximately 170,000 to 350,000 pounds more vehicle weight than the orbital capture missions in a given launch opportunity.

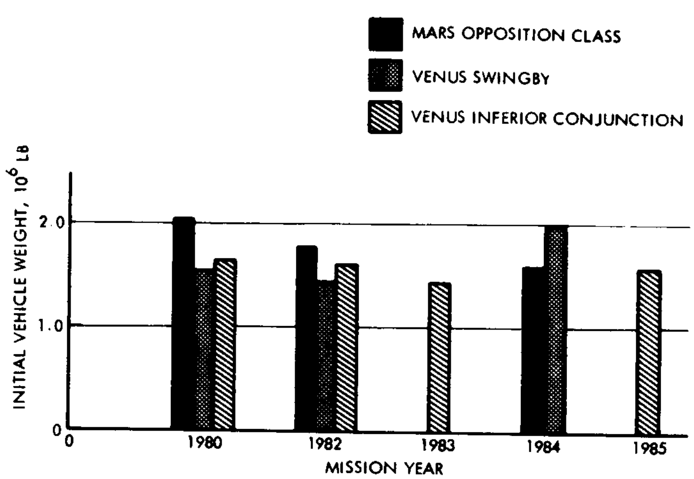


Figure III-9. Orbital Capture Mission Mode Comparison—NNNA Vehicle Configuration

IV. PHASE B MISSION ANALYSES

Phase B, or the second half of the study, was concerned with the analysis of interplanetary missions for vehicles using 75,000- and 100,000-pound thrust nuclear engines. The four tasks in Phase B consisted of the analysis of; 1) Mars lander missions, 2) Mars orbital capture missions, 3) orbital capture missions employing aftercooled nuclear engines, and 4) launch windows.

The overall objective of the Phase B analyses was to determine the vehicle weight associated with the optimum configuration, i.e., the minimum initial weight vehicle required to perform the missions consistent with a maximum engine firing time of 2700 sec and the launch azimuth constraints used in Phase A. Both circular and elliptic parking orbits were investigated for selected mission types. Only missions with Mars as the target planet were analyzed and a Martian stopover time of 30 days was used in all cases.

MISSION AND VEHICLE PERFORMANCE PARAMETERS

The primary mission and vehicle performance parameters that were postulated for this study phase in order to circumscribe the vehicle system weights and performance, vehicle configuration, mission and vehicle operational criteria, and the scope of the analysis were specified in Section II. Included in this section are the details of various scaling laws and constraints that apply primarily to the Phase B analysis.

Vehicle Configuration and Propulsion System Weights

The scaling laws and system weights used to define the mission payloads, propellant tanks, propulsion systems, secondary spacecraft systems, and operational modes are essentially those given in Section II except for the qualifications noted below.

The vehicle modes analyzed in Phase B centered upon the use of 75,000- and 100,000-pound thrust nuclear engines in a connecting mode configuration. In addition to all nuclear propelled vehicles, the use of nuclear and chemical engines combined in a single vehicle but in separate stages was also considered for evaluation and comparison purposes.

The scaling laws used for the chemical propulsion systems were given in Table II-3, of Section II while the average mass fractions used for the nuclear propulsion connecting mode configurations were presented in Table II-4 of the same section. The detailed connecting mode scaling laws used to size the 75,000- and 100,000-pound thrust nuclear engine modules are given in Appendix A. These equations were derived for each of the engine clustering arrangements analyzed in this study phase. The nuclear engine weight and performance parameters are given in Section II.

Launch Azimuth Constraints

A discussion of the launch azimuth constraints used in this study phase was given in Section III for the Phase A analysis.

Elliptic Orbit Parameters

Use of elliptic Mars parking orbits as well as circular Mars parking orbits was investigated in tasks B2 and B3. The periapsis altitude for elliptic orbits was assumed to be 600 km to coincide with the orbital altitude used for circular orbits. The apoapsis altitude was chosen so that the apsidal ratio was six (orbital period $\approx$ 14 hours).

Nuclear Engine Aftercooling Criteria

In task B3 an investigation was made of the use of aftercooled nuclear engines in which the 100,000-pound thrust arrive Mars engine was retained and aftercooled to provide the propulsion system for departing Mars. The required aftercooling propellant was computed with the following equation, provided by MSFC:

$$W_p = 6.76 \times 10^{-7} t_f^3 - 3.10 \times 10^{-3} t_f^2 + 10.14 t_f + 74.9$$

where

W_p - required aftercooling propellant (lbs)

t_f - full power (100,000 lb thrust) firing time (sec)

TASK B1 LANDER MISSION ANALYSIS

Task Description

This task involved the determination of the optimum nuclear engine clustering arrangement for manned lander missions in the 1984 to 1993 time period. The optimum clustering arrangement was selected for each mission consistent with minimum initial vehicle weight and the maximum engine firing time of 2700 sec.

The matrix of mission, trajectory, and vehicle types analyzed is given in Table IV-1. Each of the lander missions in Table IV-1 was analyzed for the use of both 75,000- and 100,000-pound thrust nuclear engines. The mission and trajectory types employed for each opportunity in the period from 1984 to 1993 are those that were determined in Phase A to yield the minimum weight vehicle within the specified launch azimuth constraints. Only the NNNA vehicle mode and circular Mars parking orbits were investigated and a 30-day stopover time at Mars was assumed.

Table IV-1. Mars Lander Mission Matrix

Launch Opportunity	Mission Type	Trajectory Type
1984	Opposition Class	IIB
1986	Outbound Venus Swingby	3A
1988	Inbound Venus Swingby	II3
1990	Opposition Class	IB
1993	Outbound Venus Swingby	3A

Analysis

In order to determine the optimum engine clustering arrangement for each lander mission shown on Table IV-1, the number of nuclear engines used for the depart Earth and arrive Mars stages was varied successively, and the SWOP program was employed to determine the minimum vehicle weight for each combination of numbers of depart Earth and arrive Mars engines. In all cases, one nuclear engine was used for the depart Mars stage; in no cases did the engine firing time exceed 2700 sec for the single depart Mars engine.

The vehicle weight and engine firing time results obtained from this matrix of engine clustering arrangements were noted and the optimum configuration was selected. The optimum configuration was the vehicle that satisfied the combined requirements of 1) minimum initial weight, 2) minimum number of clustered engines in the depart Earth and arrive Mars stages, and 3) engine firing time not exceeding 2700 seconds.

Results and Discussion

Figures IV-1 through IV-5 graphically present the results for each launch opportunity from 1984 to 1993. In each figure is shown the required initial vehicle weight and ranges of engine firing times as a function of the number of engines clustered in the depart Earth and arrive Mars stages. Also each figure presents the results for both engine thrust levels investigated. The optimum engine clustering arrangement is indicated by the circled point in the map for each thrust level. A succeeding table is used to summarize and compare the optimum configurations required for the various lander missions and engine thrust levels analyzed.

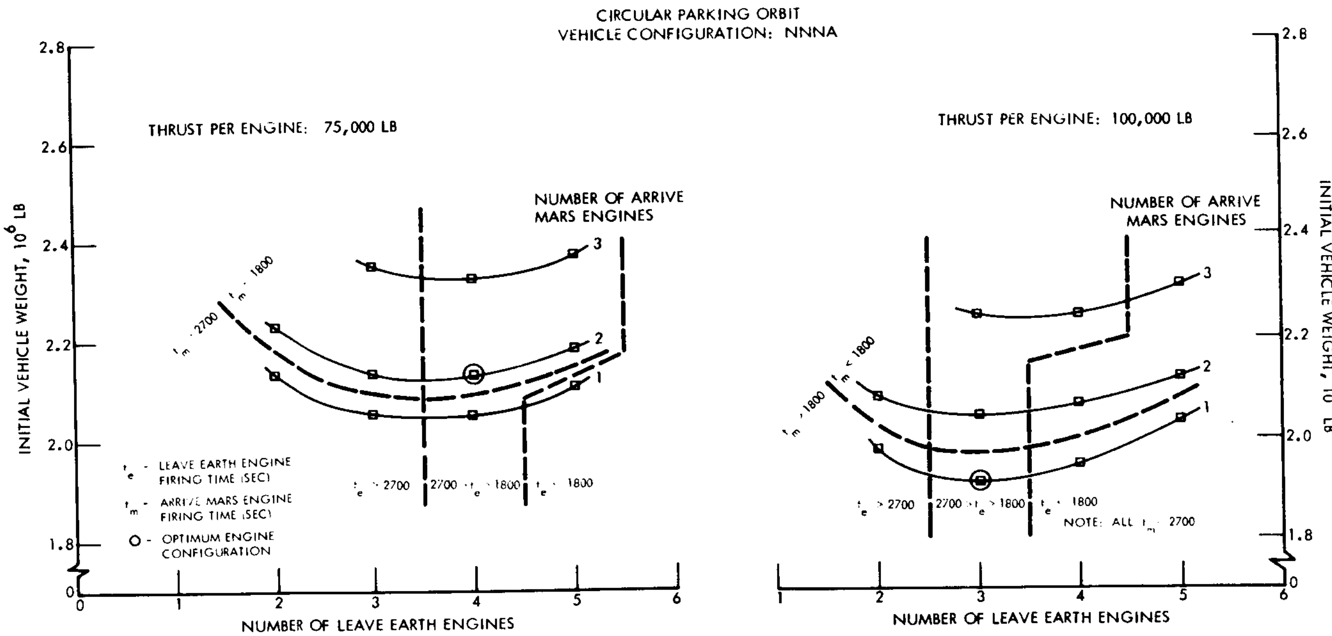


Figure IV-1. 1984 IIB Opposition Class Lander Mission

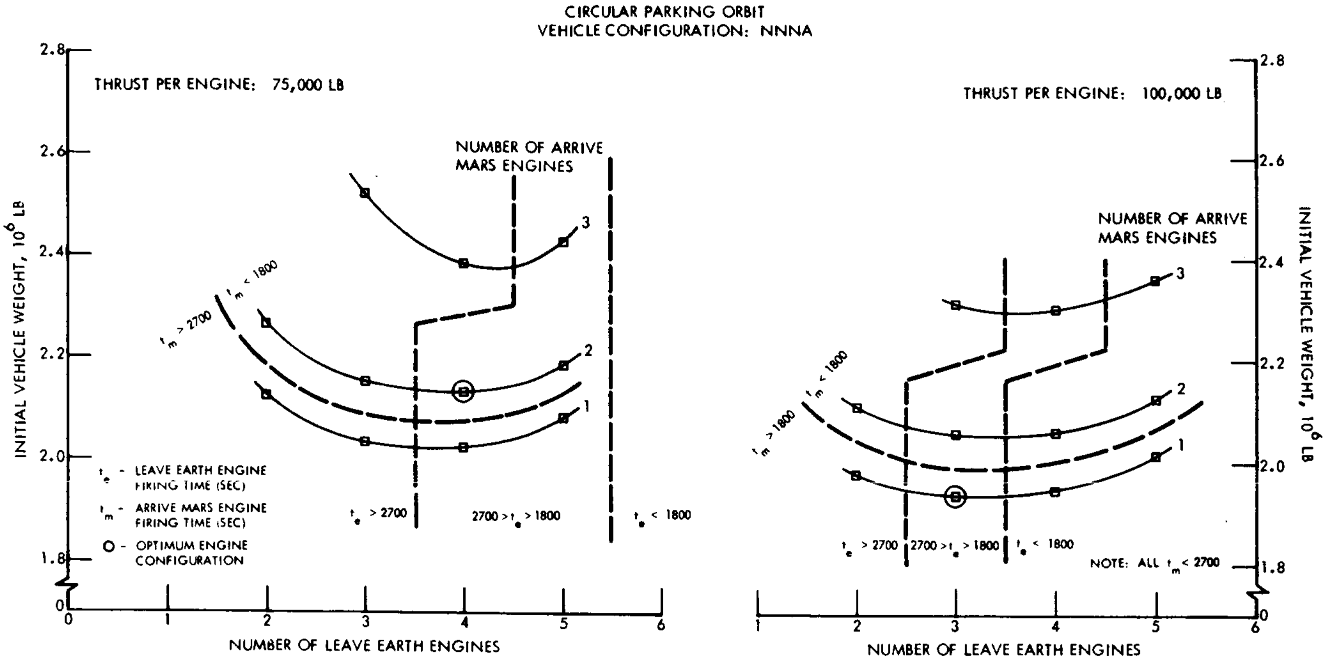


Figure IV-2. 1986 3A Outbound Venus Swingby Lander Mission

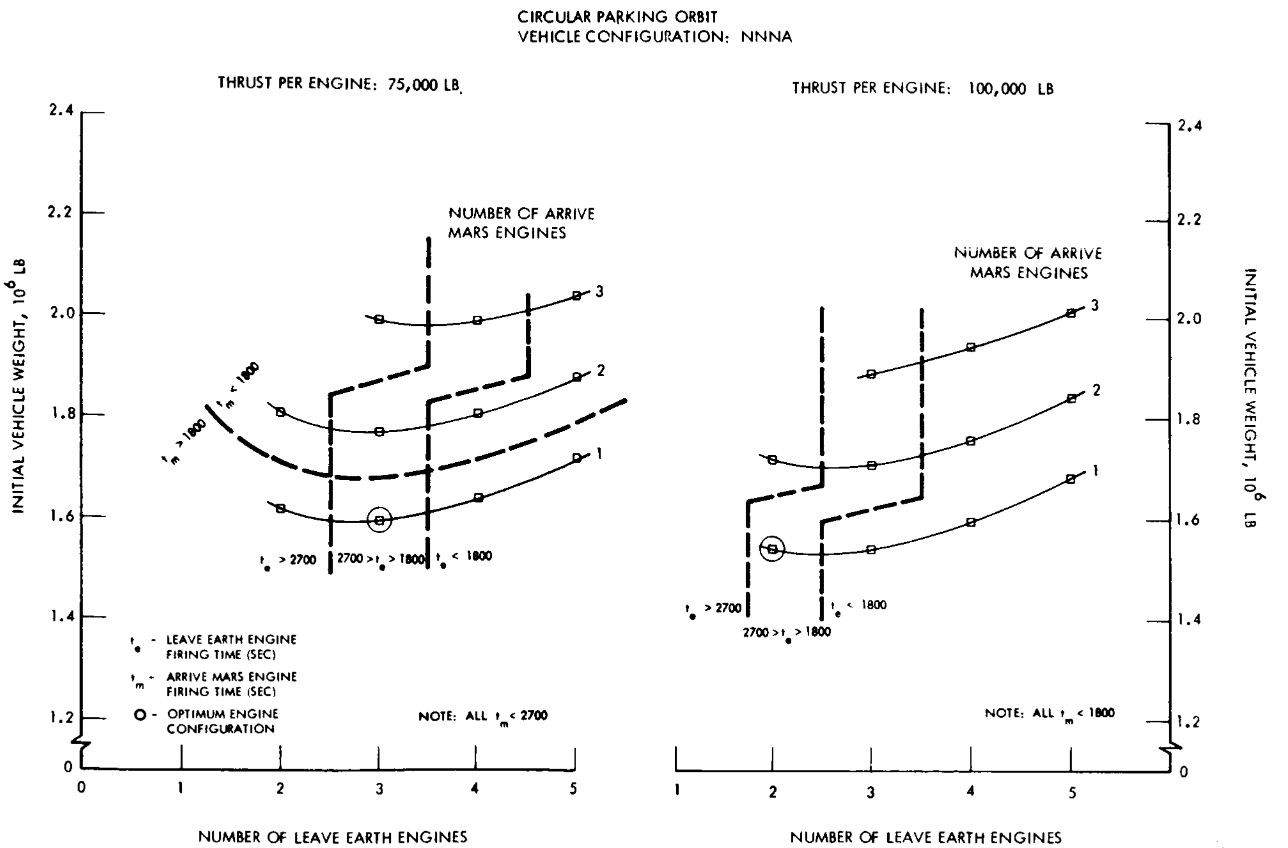


Figure IV-3. 1988 II3 Inbound Venus Swingby Lander Mission

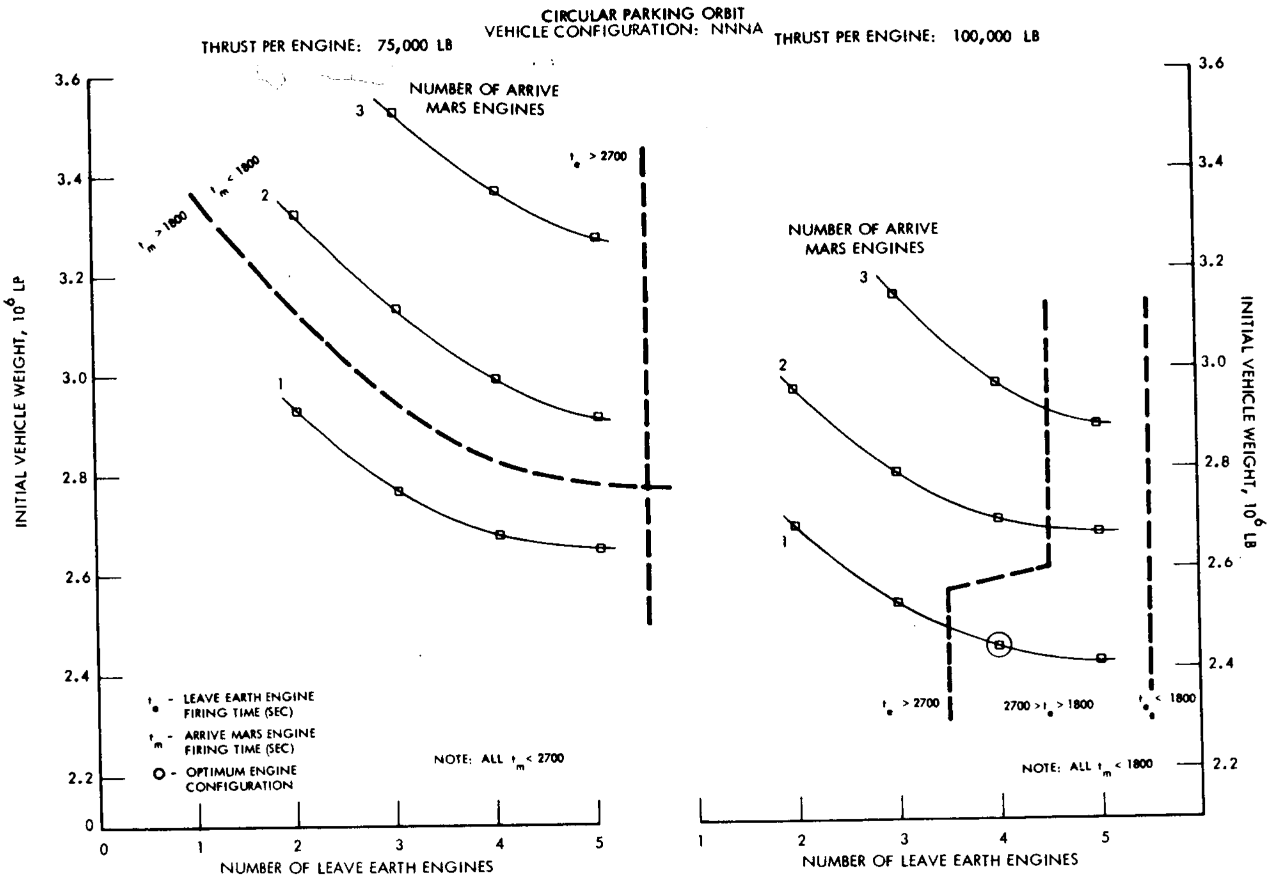


Figure IV-4. 1990 IB Opposition Class Lander Mission

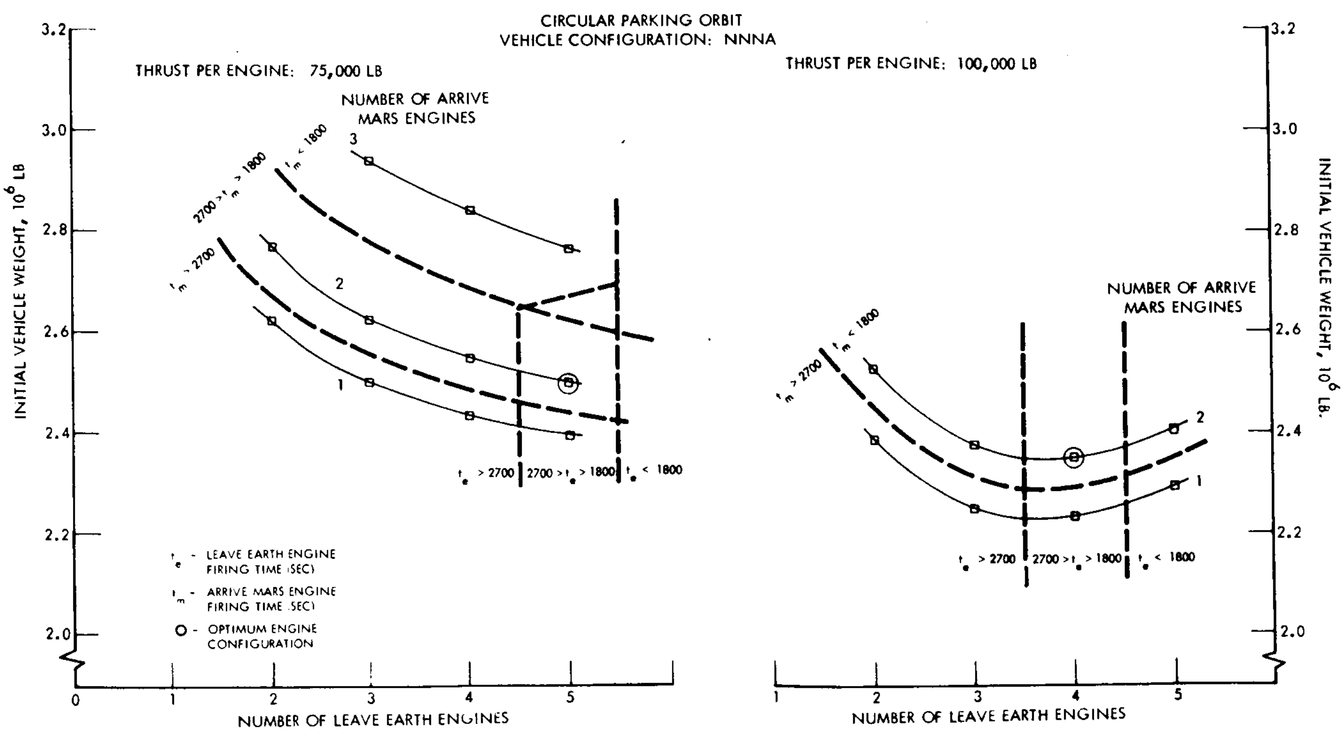


Figure IV-5. 1993 3A Outbound Venus Swingby Lander Mission

Figures IV-1 and IV-2 present the results for the 1984 IIB opposition class and the 1986 3A outbound swingby lander missions when the number of clustered engines is varied. The optimum configurations are identical for both of these lander missions, i. e., a 4-2-1 configuration is required for the 75, 000-pound thrust engines, and a 3-1-1 configuration for the 100, 000-pound thrust engines.

The results for the 1988 II3 inbound swingby lander mission on Figure IV-3 show that the optimum configurations are a 3-1-1 configuration for 75, 000-pound thrust engines and a 2-1-1 configuration for the 100, 000-pound thrust engines. Figure IV-4 for the 1990 IB opposition class lander mission indicates that over five clustered engines would be required for the depart Earth stage if 75, 000-pound thrust nuclear engines were used for this mission. The use of the 100, 000-pound thrust engines permits the use of a 4-1-1 configuration to meet the combined requirements of a 2700 sec engine firing time limitation and minimum vehicle weight and number of engines.

Finally, results for the 1993 3A outbound swingby lander mission are presented in Figure IV-5. The optimum configurations are 5-2-1 and 4-2-1 engine clustering arrangements for the 75, 000- and 100, 000-pound thrust engines, respectively.

Table IV-2 summarizes the results for the optimum configurations of each of the lander missions analyzed, permitting a comparison of the vehicle weight, number of engines, and number of modules required as a function of the assumed nuclear engine thrust level. The results for missions using 200, 000-pound thrust engines are included to extend the comparison to the engine analyzed in Phase A of the study.

First, a vehicle weight reduction of 3 to 10 percent is obtained by employing a 100, 000-pound thrust engine in lieu of a 75, 000-pound thrust engine. The greatest percentage weight reduction is obtained for the 1980 mission; the least for the lowest weight mission occurring in 1988. The use of the 200, 000-pound thrust engine in lieu of the 100, 000-pound thrust engine results in a somewhat greater weight reduction, ranging from 6 to 13 percent. In this case the minimum percentage weight reduction occurs

Table IV-2. Mars Lander Missions, Vehicle Mode—NNNA, Circular Mars Parking Orbit

Mission	75,000-lb Thrust: Vehicle Weight (10^6 lb)	75,000-lb Thrust: Configuration	75,000-lb Thrust: Number Modules	100,000-lb Thrust: Vehicle Weight (10^6 lb)	100,000-lb Thrust: Configuration	100,000-lb Thrust: Number Modules	200,000-lb Thrust: Vehicle Weight (10^6 lb)	200,000-lb Thrust: Configuration	200,000-lb Thrust: Number Modules
Opposition 1984 IIB	2.129	4-2-1	8	1.911	3-1-*1	7	1.798	2-*1-*1	8
Venus Swingby 1986 3A	2.134	4-2-1	8	1.936	3-*1-*1	8	1.790	2-*1-*1	8
Venus Swingby 1988 II3	1.599	3-1-1	6	1.545	2-*1-1	7	1.455	**1-1-1	6
Opposition 1990 IB	-	-	-	2.423	4-*1-1	9	2.208	**2-1-1	9
Venus Swingby 1993 3A	2.507	5-2-1	9	2.355	4-2-1	8	2.046	2-*1-*1	8

*Tier 2A required

**Tiers 2 and 3 required

in 1984 and 1988 and the maximum in 1993. By employing 200,000-pound thrust engines in lieu of the 75,000-pound thrust engines, the weight reductions range from 9 to 18 percent; the least in 1988, the greatest in 1993.

Second, the use of the 100,000-pound thrust engine required at least one engine and in two cases (1984 and 1986) two engines less than the engine requirements for the 75,000-pound thrust engine. By employing the 200,000-pound thrust engine instead of the 100,000-pound thrust engine, the engine requirements can be reduced by one to three engines. The greatest reduction in required engines occurs in 1993 where, it also should be noted, four fewer engines are required by increasing the engine thrust level from 75,000 to 200,000 pounds.

Finally, these results indicate that an increase in engine thrust level may or may not reduce the number of required spacecraft modules. (The spacecraft modules consist of the required propulsion and propellant modules, and one module containing the payload and secondary systems.) However, these results are not altogether significant, since the assumption was made for the connecting mode configuration that all modules in a given stage would be "in line". For example, if two propulsion modules were used in tier 1, tier 2, if required, must consist of two propellant modules; if three propulsion modules were used in tier 1, tier 2 must consist of either one or three propellant modules; and if four propulsion modules were used in tier 1, tier 2 must consist of either two or four propellant modules. This identical "in line" constraint also applied to the propellant modules added in tier 3. Accordingly, in cases such as in 1980, 200,000-pound thrust; 1988, 100,000-pound thrust; and 1990, 200,000-pound thrust; the in line modular concept yields an inefficient application of the connecting mode concept in terms of the minimum number of propellant modules required.

TASKS B2 and B3 ORBITAL CAPTURE MISSION ANALYSIS

The second and third tasks of Phase B both involved the analysis of orbital capture missions. The primary differences between these tasks were the vehicle modes analyzed in each; Task B2 was concerned with vehicles employing nonaftercooled nuclear engines while under task B3, the use of aftercooled engines was investigated. Since the analysis objectives and mission assumptions for these two tasks were identical, and the results obtained in both tasks were to be compared, it is appropriate that the discussions for these two tasks be combined in this section of the report.

Task Description

Tasks B2 and B3 had the same objective as did the preceding task, B1; i.e., the determination of the optimum nuclear clustering arrangement for each mission analyzed. The use of the 75,000-pound thrust nuclear engine was investigated only for the NNNA vehicle mode while the 100,000-pound thrust nuclear engine was investigated for the NNNA, NNSA, NN_JNA, and $NN_{NJ}NA$ vehicle modes. Only the 1984 IIB opposition class orbital capture mission was analyzed but both circular and elliptic Mars parking orbits were investigated for all mission cases. A 30-day stopover period was assumed.

Analysis

As in task B1, the optimum engine clustering arrangement for each orbital capture mission was determined by varying successively the number of depart Earth and arrive Mars clustered nuclear engines and employing the SWOP program to determine the minimum vehicle weight for each combination of numbers of depart Earth and arrive Mars engines. In all cases except for the NNSA vehicle mode, one nuclear engine was used for the depart Mars stage. The aftercooled modes employed 2-1-1 configurations, i.e., two nuclear engines for departing Earth and one engine for arriving at Mars with this latter engine aftercooled and reused to provide the depart Mars propulsive maneuver.

Two basically different staging arrangements were investigated in the use of the aftercooled engine. In the first aftercooling mode, NN_JNA, the arrive Mars stage was provided with a propellant module that contained the required propellant for braking into the Mars parking orbit and subsequently aftercooling the nuclear engine. After the engine was aftercooled the propellant module was jettisoned but the engine retained. A separate propellant module was provided for the depart Mars phase of the mission. In the second aftercooling mode, $NN_{NJ}NA$, one propulsion module contained all propellant required for the arrive Mars retro phase, the nuclear engine aftercooling, and the depart Mars injection phase. This propulsion module was retained during the stopover period and jettisoned after the depart Mars injection maneuver.

An iterative computer technique was employed for analyzing the missions using elliptic parking orbits about Mars. A basic problem in the analysis lies in the fact that the arrival and departure velocity vectors are separated by a substantial turn angle such that a vehicle cannot arrive and depart near the periapsis of a given elliptic parking orbit. Hence, the orientation of the line of apsides of the ellipse must be selected so as to minimize the penalties associated with arriving at or departing from a nonperiapsis point. In some cases, the vehicle may be 90 degrees or more away from the periapsis point when making an arrival or departure maneuver; hence a substantial penalty can be incurred compared to the ideal case of cotangential periapsis transfer. The problem is further complicated by the rotation of the line of apsides due to planet oblateness. The orientation of the ellipse must be selected on the basis of minimizing the total vehicle weight ratio involved in the capture and subsequent departing maneuvers.

The following computational procedure was established for the analysis of the elliptic parking orbits. From a selected set of departure dates and leg durations, the interplanetary transfer trajectories were computed giving the asymptotic velocities of arriving and departing the target planet, and the total angle between them. For a given orientation angle of the ellipse, the location of the ellipse which yields the minimum ΔV's to effect both capture and departure from the ellipse were then computed in a minimum velocity routine (MVR). This procedure was repeated for a range of ellipse orientation angles and the results entered into a weight ratio routine (MWR) which computed the weight ratio from capture through departure. From the results an orientation angle can be selected which minimizes the vehicle gross weight. The ΔV's corresponding to the optimum orientation angle were input into the trajectory-vehicle weight optimization program, SWOP to compute the detailed vehicle weights and propellant requirements. This process was repeated for a number of trajectory dates to find the set of dates that minimizes the total vehicle weight.

As previously stated, the periapsis altitude for the elliptic parking orbits at Mars was assumed to be 600 km to coincide with the orbital altitude used for circular orbits. For each mission and vehicle, a tradeoff analysis of energy requirements and lander weight would yield the best apoapsis altitude. Such an analysis, with its interactions with optimizing orientation, trip times, dates, etc., would be very complex and would not be warranted except for missions for which elliptic parking orbits had been predetermined to be advantageous. Therefore, for these preliminary mission analyses one fixed value was assumed for all cases. The apoapsis altitude was chosen so that the ratio of apoapsis to periapsis radius would be six. The parking orbit orientation; launch dates, and trip times were optimized in each case to minimize total initial weight.

Additional details of the techniques used to analyze elliptic parking orbits are given in Appendix A of Reference 8.

Results and Discussion

As in the first task of Phase B, the required initial vehicle weights and ranges of engine firing times are presented in Figures IV-6 through IV-8 as a function of the number of engines clustered in the depart Earth and arrive Mars stages. Figure IV-6 presents the results of the 1984 orbital capture mission for a NNNA vehicle configuration employing 75,000-pound thrust nuclear engines. If a circular Mars parking orbit is used, a 4-2-1 configuration is optimum; if an elliptic Mars parking orbit is used, a 3-1-1 configuration is optimum and the vehicle weight is reduced by approximately 600,000 pounds.

Figure IV-7 presents the results for the same mission and vehicle configuration as on the preceding figure except that 100,000-pound thrust engines are used and the nuclear engine aftercooled cases are included. For the NNNA vehicle and circular Mars parking orbit, a 3-1-1 configuration is optimum; for the elliptic Mars parking orbit, a 2-1-1 configuration is optimum.

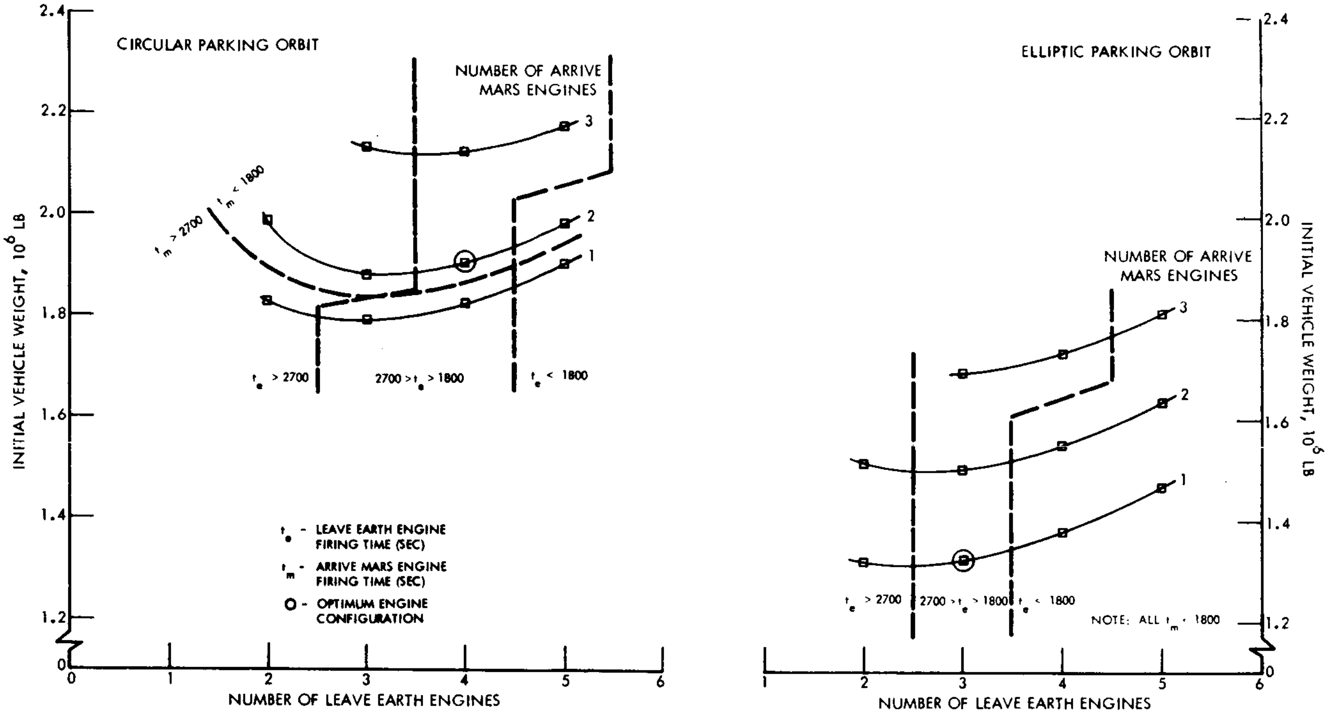


Figure IV-6. 1984 IIB Orbital Capture Mission, NNNA, 75,000-Pound Thrust

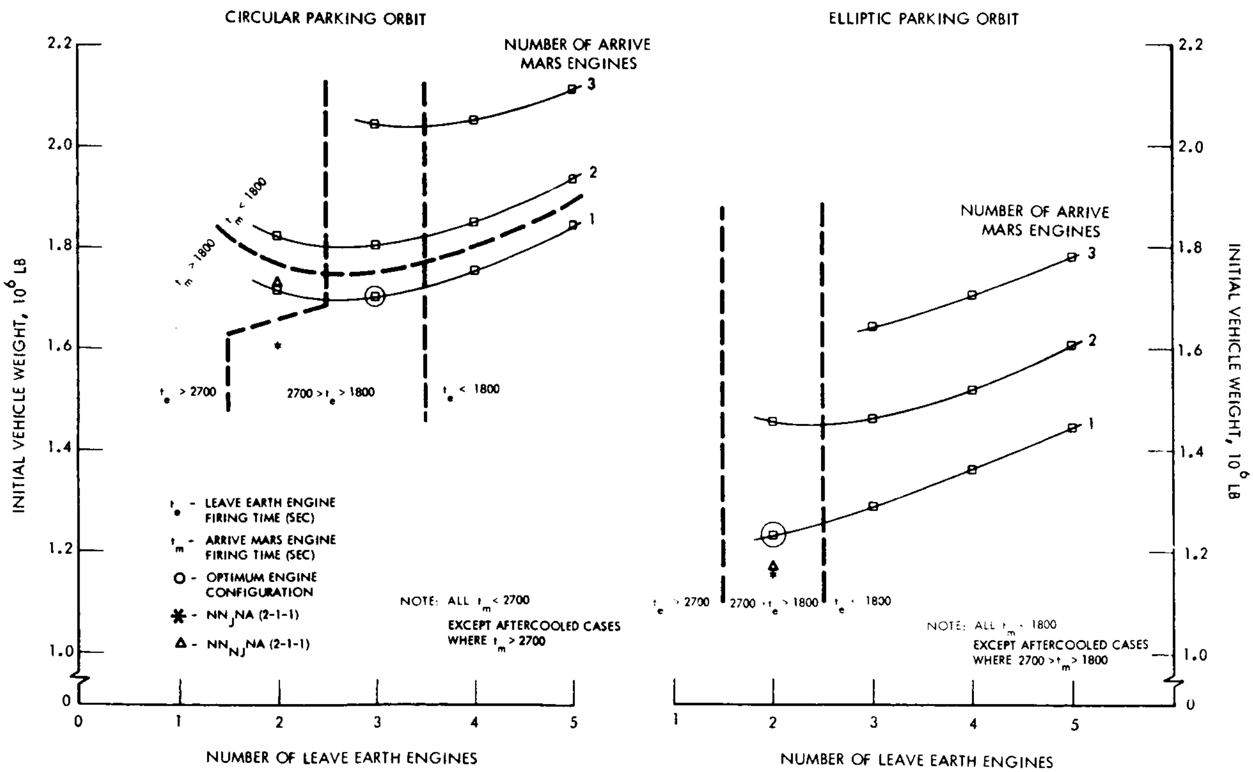


Figure IV-7. 1984 IIB Orbital Capture Mission, NNNA, 100,000-Pound Thrust

Figure IV-8 presents the results for the 1984 IIB mission using a NNSA vehicle mode and 100,000-pound thrust nuclear engines for departing Earth and arriving Mars. The use of an elliptic Mars parking orbit provides a very large vehicle weight reduction for this mode; the vehicle weight is reduced by approximately one million pounds from that required for the circular Mars parking orbit mission. Furthermore, the use of the elliptic orbit required one half the number of nuclear engines required by the circular orbit; 2-1-0 and 4-2-0 configurations, respectively.

Table IV-3 summarizes the results for the optimum configurations of each of the 1984 IIB orbital capture missions analyzed permitting a comparison of the vehicle weight, number of engines, and number of modules required as a function of the assumed nuclear engine thrust level, type of Mars parking orbit, and nuclear engine aftercooling. The results for missions using a 200,000-pound thrust engine are included to extend the comparison to the engine analyzed in Phase A of the study.

First, a vehicle weight reduction of six to ten percent is obtained for the NNNA configuration by employing a 100,000-pound thrust engine in lieu of a 75,000-pound thrust engine, or a 200,000-pound thrust engine in lieu of a 100,000-pound engine for either the circular or elliptic orbit cases. By employing 200,000-pound thrust engines in lieu of the 75,000-pound thrust engines, the weight reductions are 16 to 13 percent for the circular and elliptic orbit cases, respectively. For the same NNNA vehicle configuration, two fewer engines are required in the circular orbit case as the engine thrust is increased from 75,000-pounds to 100,000-pounds and also from 100,000-pounds to 200,000-pounds. In the elliptic orbit case only one less engine is required for each incremental increase in thrust level.

Second, this summary chart indicates that a sizeable 28 to 37 percent decrease in initial vehicle weight is obtained if elliptic Mars parking orbits are employed instead of circular orbits. The greatest weight reduction is obtained for the NNSA case; the least reduction for the NNNA, 200,000-pound and the $\mathrm{NN_JNA}$ cases.

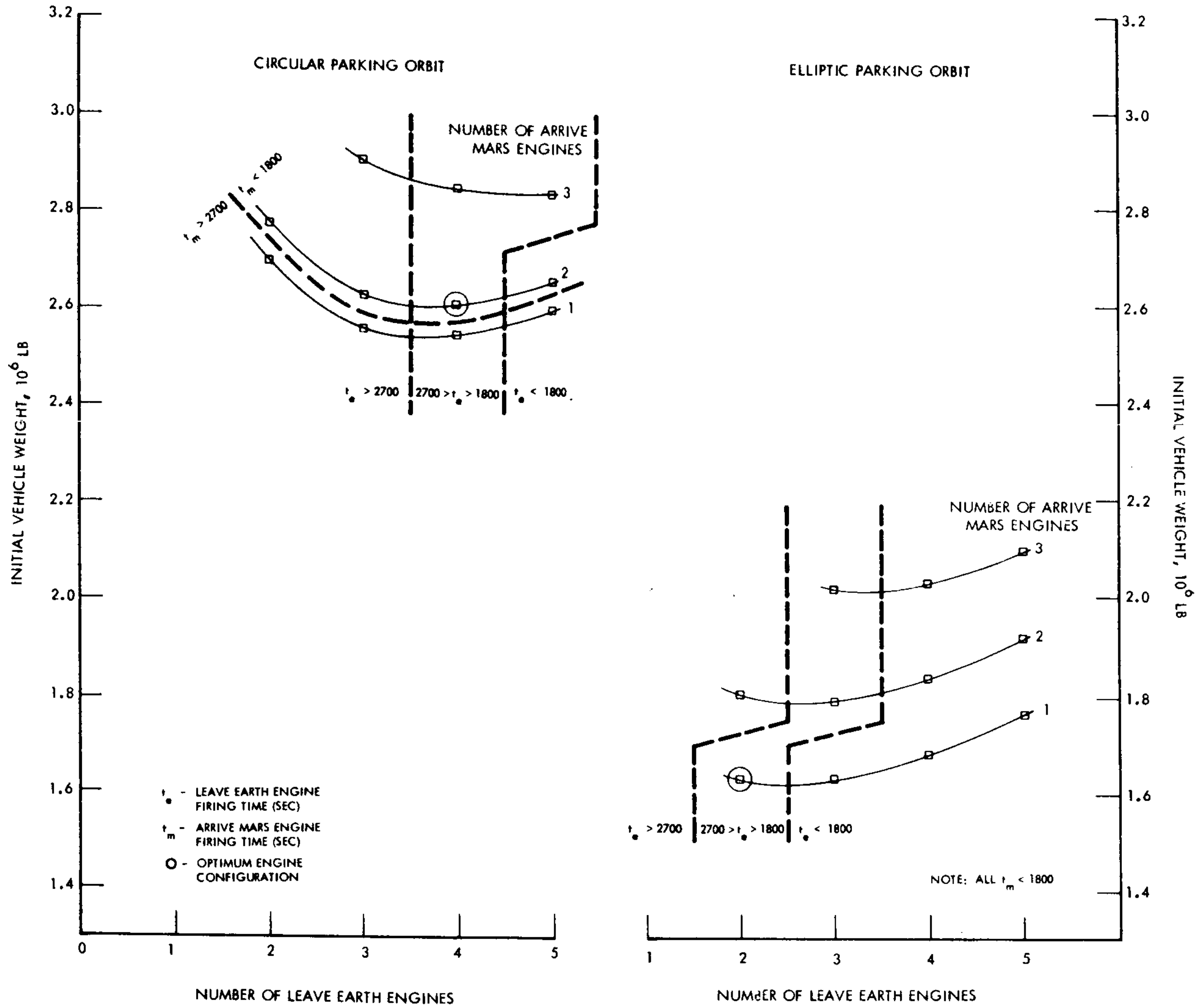


Figure IV-8. 1984 Orbital Capture Mission, NNSA, 100,000-Pound Thrust

When circular Mars parking orbits are combined with the aftercooling modes (2-1-1 configuration assumed in both cases), total firing times of the aftercooled engines exceed the 2700 sec limits by approximately 750 sec for the NN_JNA mode and approximately 1400 sec for the $NN_{NJ}NA$ mode. In addition, the depart Earth engine firing time for the 2-1-1 configuration of the $NN_{NJ}NA$ mode exceeds 2700 sec by 145 sec.

The use of elliptic Mars parking orbits with the aftercooled modes reduces the vehicle weight sufficiently so that all nuclear engines operate under the 2700 sec limit. The total firing times of the aftercooled engines for the NN_JNA and $NN_{NJ}NA$ modes are 2058 sec and 2332 sec, respectively.

Table IV-3. Mars Orbital Capture Missions Opposition Class — 1984 IIB

Thrust (lb)	75,000		100,000								200,000	
Vehicle Mode	NNNA		NNNA		NN_JNA		$NN_{NJ}NA$		NNSA		NNNA	
Mars parking Orbit	C	E	C	E	C	E	C	E	C	E	C	E
Vehicle Wt (10^6 lb)	1.91	1.32	1.71	1.23	1.61	1.61	1.73	1.17	2.60	1.63	1.60	1.15
Configuration	4-2-1	3-1-1	3-1*-1	2-1-1	2*-1-1	2-1-1	2*-1-1	2-1-1	4-2-0	2*-1-0	1**-1*-1	1*-1-1
Number Modules	8	6	7	5	7	5	6	4	8	7	7	5

*Tier 2A required
**Tiers 2 and 3 required
C Circular parking orbit
E Elliptic parking orbit

Third, a maximum reduction of only six percent is available by resorting to engine aftercooling; the vehicle weight for the $NN_{NJ}NA$ case, in fact, increases slightly over the nonaftercooled NNNA case. In addition, the arrive Mars engine firing times increase substantially for the aftercooled cases, actually exceeding the 2700 sec limit for the circular orbit cases.

Finally, as was discussed earlier for the lander missions, a comparison of the number of modules required is not significant due to the "in line" configuration assumption. However, these results indicate that the use of higher thrust levels or elliptic orbits will in general reduce the number of spacecraft modules by one or two.

TASK B4 LAUNCH WINDOW ANALYSIS

Task Description

An investigation was conducted to determine the effect on initial vehicle weight for Mars lander missions when launch windows are provided at Earth and at Mars. Nodal regression of the parking orbits was taken into account, and the propellant tanks were sized to provide the minimum initial weight vehicle for a launch on any day during the launch windows.

The analysis was conducted for a representative lander mission for each of three classes of Mars mission trajectories, viz, an opposition class trajectory and an outbound and inbound Venus swingby trajectory. These missions were made consistent with established ETR launch azimuth constraints and the vehicles were configured within the constraint of maximum firing time for the nuclear engine. The three missions and corresponding launch opportunities and vehicle configurations are given in Table IV-4. All vehicles were sized using the appropriate connecting mode scaling laws presented in the first portion of this section. The nuclear engine thrust level was 100,000-pounds per engine and the specific impulse was 850 seconds.

Table IV-4. Mission and Vehicle Mode Matrix

Mission Type	Year	Trajectory Type	Vehicle Mode
Opposition	1984	IB	NNNA
Outbound Venus Swingby	1986	3A	NNNA
Inbound Venus Swingby	1988	II3	NNNA

A IB trajectory was selected for the 1984 opposition class mission, even though a IIB trajectory results in a slightly lower weight vehicle for the optimum launch dates. The reason is that the declination associated with the optimum IIB trajectory is approximately -36 deg, which is approximately the maximum declination (orbit inclination) limit due to launch azimuth constraints. Due to this limit, the plane change and interplanetary energy requirements required for the Earth launch window lead to prohibitive vehicle weights if the launch azimuth constraint is not to be violated. Consequently, a IB opposition class trajectory was selected as the alternative for the single impulse injection, 1984 opposition class mission.

In order to establish logical stage size requirements for the modular vehicle and at the same time limit the analysis to a level consistent with the scope of the study, the vehicles were configured on the basis of certain guidelines. That is, a determination was made of the minimum weight vehicle that would permit the specified launch window at Earth and Mars consistent with the following operation and configuration guidelines:

- A fixed, 20-day launch window was provided at Earth and at Mars. Therefore, the launch window at Mars occurred during a stopover period varying from 30 to 50 days.
- The injection of the vehicle into the interplanetary trajectory from the parking orbit both at Earth and Mars was confined to a single maneuver. Therefore, no analysis was made of staging possibilities or dual burn maneuvers.
- Circular parking orbits were assumed at Earth and Mars.

- Whenever a vehicle stage contained an amount of propellant greater than that which would subsequently be required (due to an early launch from Earth within the launch window), the excess propellant was jettisoned as appropriate, either in the Earth parking or interplanetary orbit. The alternative (which was not investigated) to this mode of operation would be to transfer the propellant to a stage that required a greater amount of propellant due to the early launch.
- All propulsion or propellant modules for a given vehicle were sized for the propellant capacity that would be required for a minimum weight vehicle with the capability of a launch on any day during the launch window. That is, no attempt was made to limit the modules of a given vehicle to a specified number of equal size modules.
- The minimum number of nuclear engines was to be used for each stage consistent with minimum vehicle weight and with a maximum engine firing time of 2700 seconds.

Analysis Approach

In designing a single vehicle which can be launched on any day during a specified launch window period both at Earth and at Mars, it is necessary to have every stage of the vehicle sized properly so that it will be able to perform its assigned maneuver regardless of when during the window the launch actually occurs. However, the ultimate goal in designing such a vehicle is to plan the launch window and the mission so that the vehicle has the lowest possible total weight after its component stages have been sized. The complexity of this sizing problem is compounded by the requirement that the vehicle will leave from an assembly/parking orbit. Ideally, the assembly/parking orbit should be selected in advance so that the vehicle can depart the planet in the plane of the parking orbit (eliminating costly plane change maneuvers). This is impossible for two reasons: it is not known in advance on which day the vehicle will be able to depart, and the orbit plane does not remain fixed in space but precesses due to the oblateness perturbation by the planet.

Parking Orbit and Trajectory Analysis. The vehicle for a Mars stopover mission is composed of four primary stages to perform the four major maneuvers; viz, leave Earth, arrive Mars, leave Mars, and arrive Earth. Launch windows must be provided in case of launch delays leaving

Earth or leaving Mars. In the process of selecting the parking orbits and interplanetary trajectories, at least three approaches can be taken:

- For each depart date (at Earth or Mars) use the trajectories from Earth to Mars and from Mars back to Earth which give the best possible combination of ΔV's disregarding any plane changes that may be required leaving Earth or Mars, i. e. , the "optimum" interplanetary trajectory for each launch date in the windows. Then choose the parking orbits at Earth and Mars so that the effect of the plane change requirements, when added to the departure ΔV's, will be minimized over the whole window.

- Choose the parking orbit so that the vehicle can depart in the plane of the parking orbit on one day in the launch window. Then on all other dates in the window, depart in the plane of the parking orbit even though it requires using "non-optimum" interplanetary trajectories. This approach compared to the first has the effect of reducing the total departure ΔV's (because there is no plane change) at the expense of increased arrival ΔV's at Mars and Earth.

- Combine the first two approaches so that interplanetary trajectories and parking orbits are chosen which give the best possible combination of ΔV's <u>including</u> plane change requirements. This approach would result in using "non-optimum" interplanetary trajectories (but closer to "optimum" than the second approach), with small plane changes (smaller maximum plane changes than the first approach).

The third approach is the most desirable since it will yield the lowest weight vehicle of all three approaches. However, this approach would have involved development of a computer program to carry out the indicated optimization and was beyond the scope of the contract. A preliminary analysis of the first two approaches showed them approximately competitive, with the first showing slightly greater promise. Therefore, the approach used for the remainder of the launch window studies was the first of the three approaches.

A series of manual analyses and computerized steps were used to determine the minimum weight vehicle configuration for the 20-day Earth and Mars launch windows. First, the SWOP program was used to determine the "optimum" interplanetary trajectories for a range of depart

Earth dates and a range of stopover times at Mars from 30 to 50 days (to allow for the Mars launch window). These computations resulted in a vehicle weight curve similar to that shown in Figure IV-9. A depart Earth date (sometimes but not always, the lowest point on this curve) was selected as the nominal depart Earth date or center of the Earth launch window. From the trajectory data output by SWOP, the impulsive velocities and the right ascensions and declinations of the departure hyperbolic excess velocities were determined and noted for a 10-day range of Earth depart dates on either side of the nominal depart date.

Ideally, it would be desirable to select the parking orbit (for the noted departure right ascensions and declinations) so that the plane change velocity requirements would be zero on or close to the first and last day of the launch window and maximum at the point approximately midway between them, as depicted in Figure IV-9. This choice of parking orbit has the effect of minimizing the plane change propellant requirements

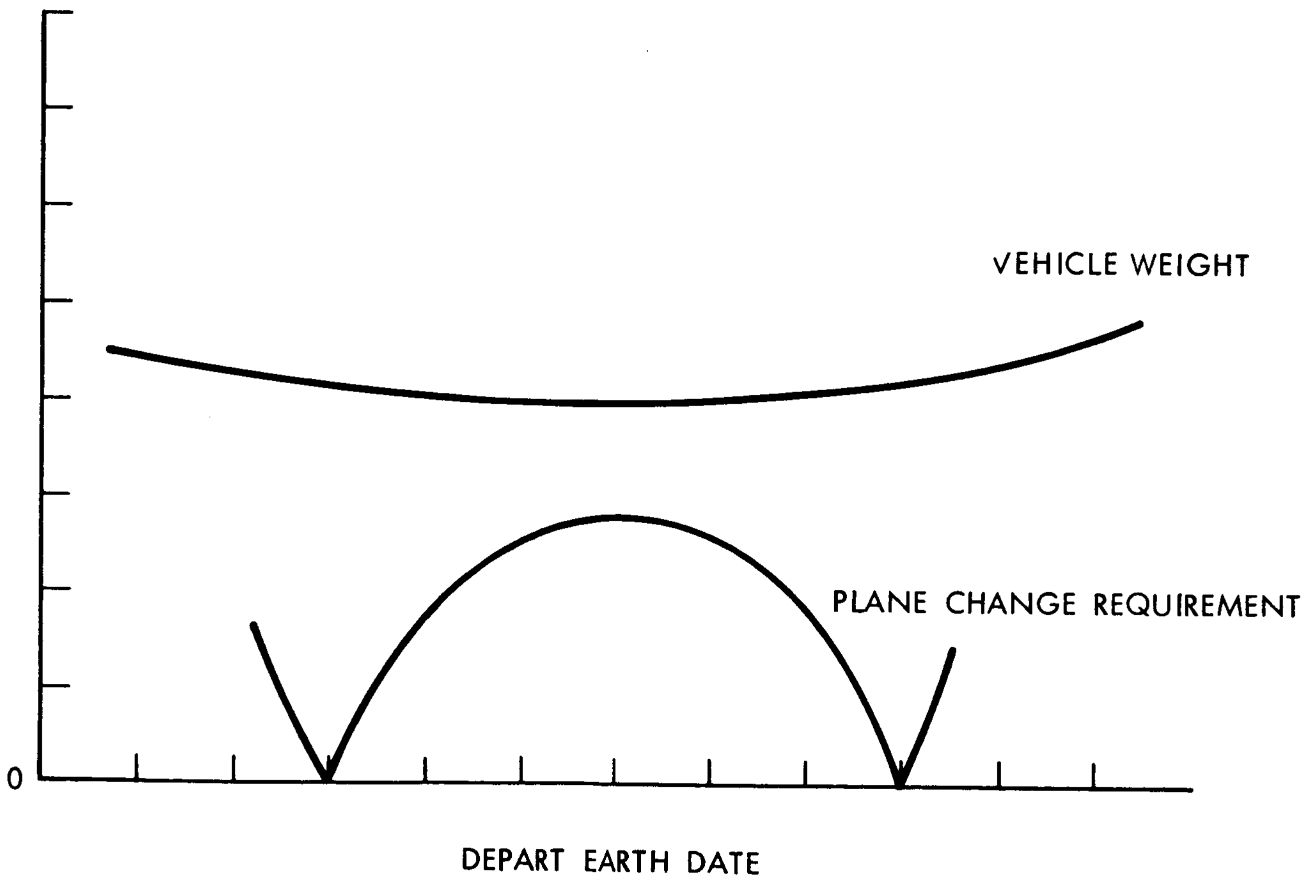


Figure IV-9. Vehicle Weight and Plane Change Requirements at both ends of the launch window when the total vehicle weights are greatest. Similarly, the plane change propellant requirements near the nominal launch date are maximum when the total vehicle weight requirements are minimum. The expected overall effect is a flattening out of the vehicle weight curve as the vehicle is subsequently sized. The geometrical characteristics of an ideally selected parking orbit are illustrated in Figure IV-10.

The oblateness of the planet perturbs the parking orbit, resulting in a precession of the orbit plane, i.e., the longitude of the node, Ω, of the orbit decreases. (For illustrative purposes the orbit plane is held stationary while the hyperbolic excess velocity vector, $\overline{V}_\infty$, is advanced away from the reader.) The time rate of precession, $d\Omega/dt$, is negative and proportional to the cosine of the inclination, i, of the parking orbit. On the first day of the launch window (in the ideal case), $\overline{V}_\infty$ is in the plane of the parking orbit (position closest to the reader). On that day no plane change would be required. It is assumed for the moment that

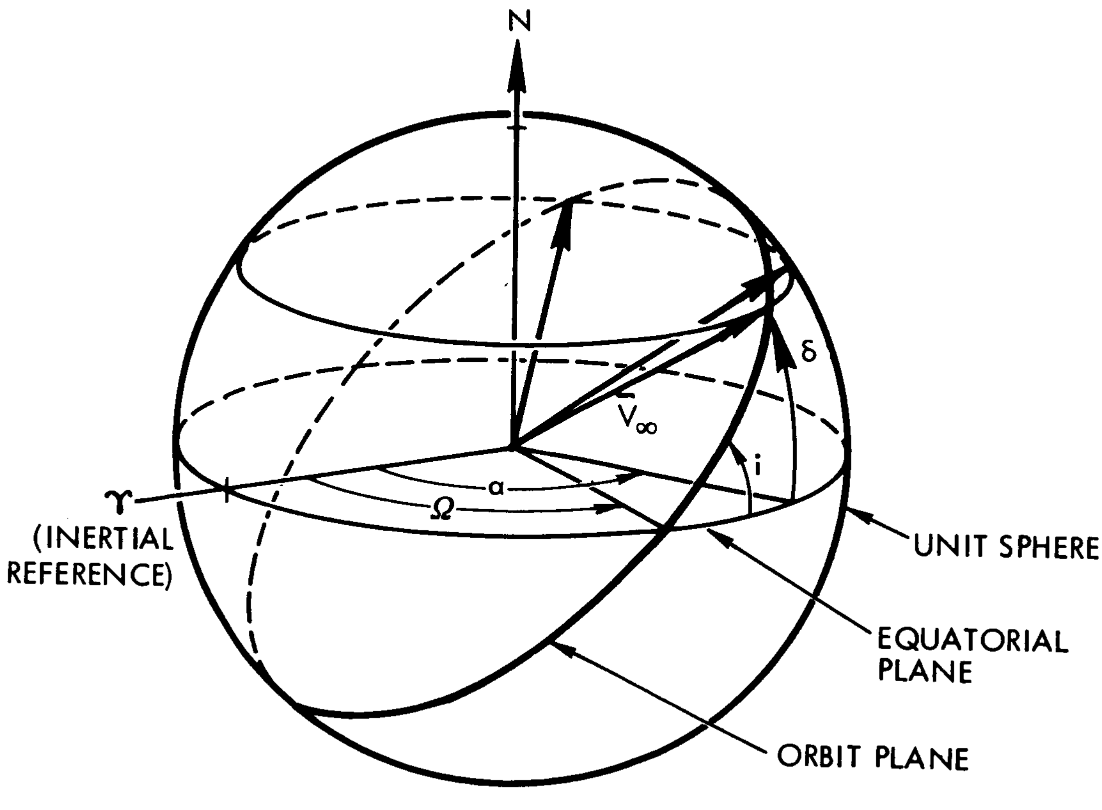
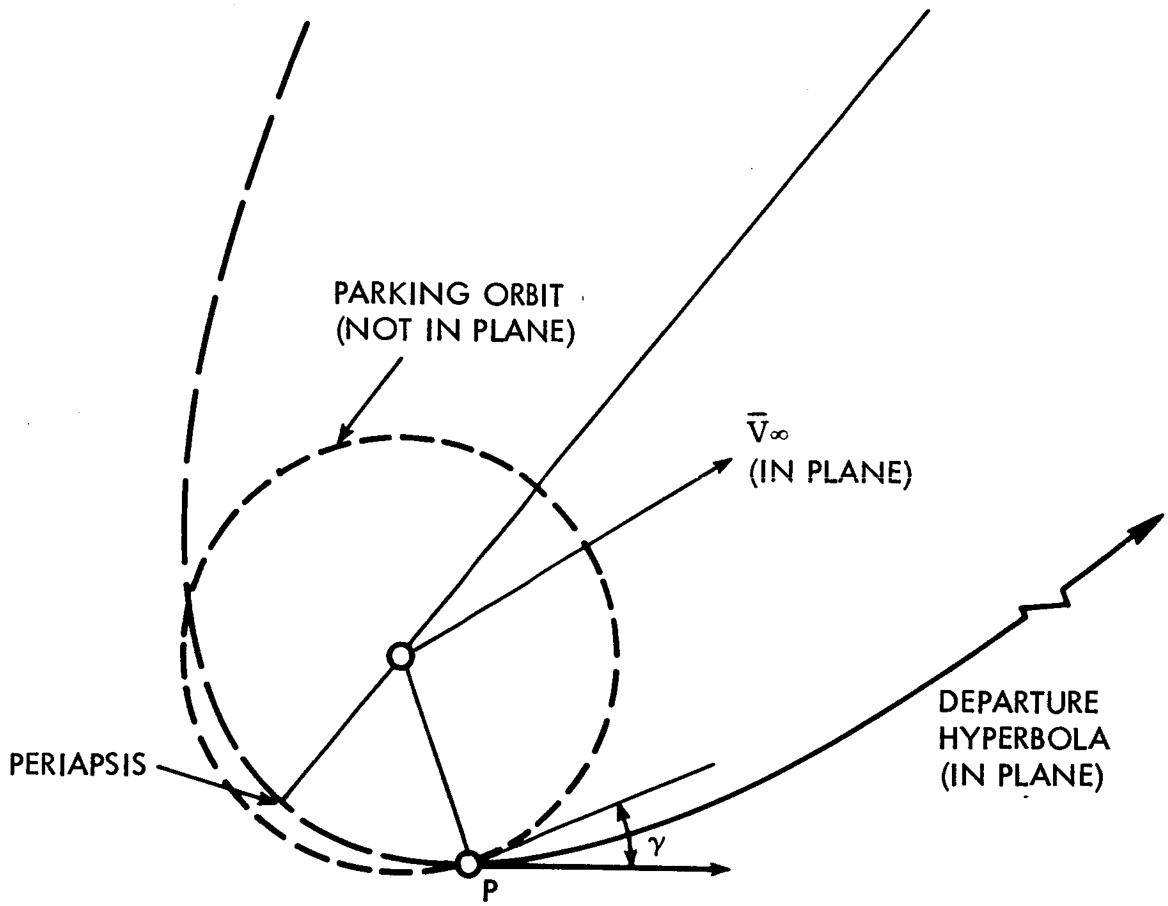


Figure IV-10. Parking Orbit Characteristics the right ascension, α, and the declination, δ, of $\overline{V}_\infty$ remain fixed over the launch window (actually the "optimum" right ascension and declination vary with date, and this was taken into account in the final analysis). It is seen that as time progresses past the first day of the window and as the parking orbit precesses ($\overline{V}_\infty$ advances away from the reader), $\overline{V}_\infty$ will be below the orbit plane for a period of time and at some later time will again lie in the plane of the orbit. If the inclination of the parking orbit is chosen properly, thereby governing the rate of precession, V_∞ can be made to lie in the orbit plane (thus requiring no plane change) on the specified first and last days of the window. For this condition, the plane change requirement curve shown in Figure IV-9 is obtained.

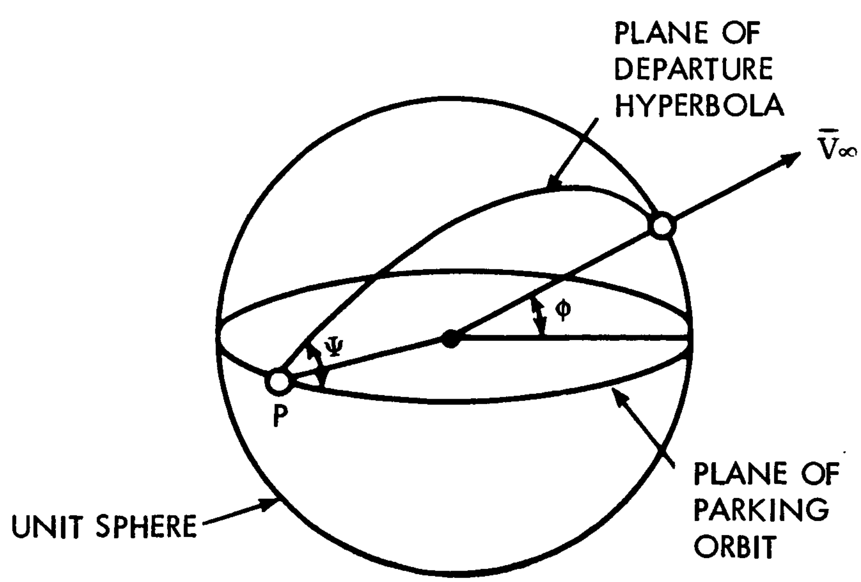
The above technique for selection of the proper optimum orbit plane represents a considerable oversimplification of the technique used in the actual launch window analyses performed. There are several trajectory constraints and conditions that significantly influence the choice of an optimum parking orbit. First, a selected orbit plane which results in minimum plane change ΔV requirements for a given range of depart Earth days may not be acceptable due to the launch azimuth constraints. That is, the inclination of the parking orbit must lie between approximately 28.4 deg and 36.6 deg. Secondly, the right ascension and declination of the departure asymptote generally vary monotonically across a range of Earth depart dates. Since the magnitude of the plane change ΔV, for a given depart date, is a function of the angle between the $\overline{V}_\infty$ vector and the orbit plane, the optimum parking orbit (inclination and longitude of the nodal points) is a function of the initial launch date (which specifies the range of right ascensions and declinations) for the specified 20-day Earth launch window. Finally, the impulsive velocity (without any plane change ΔV) required for achieving the desired Earth departure asymptotic conditions also varies across a range of Earth depart dates. The variation in this velocity is generally small and therefore usually has a relatively smaller effect on the optimization than does the influence of the variations in the ΔV's required for the plane changes. (It should be noted that the range of required arrive Mars, depart Mars, and arrive Earth velocities also varies with depart Earth date, and hence can influence the selection of the optimum initial depart Earth date for the specified launch window. Generally, the variations in these latter velocities with depart Earth date are also small and therefore, have a second order effect on the optimization.)

Due to the above trajectory conditions and constraints, the overall launch window optimization or parking orbit selection results from a tradeoff among the effects of: 1) the inclination of the parking orbit, which determines the time rate of precession of the line of nodes and affects the magnitude of the plane change ΔV for any given depart date and hyperbolic asymptote; 2) the orientation of the parking orbit, i.e., the longitude of the nodal points, which determines the relative angle between the $\overline{V}_\infty$ vector and the orbit plane; and 3) the initial depart date which determines the right ascension, declination, and magnitude of the departure asymptote (as well as the later mission impulsive velocities) for each day in the launch window. The effects of these parameters are measured in terms of their effect on initial vehicle weight, although in the actual analysis it was generally sufficient to measure their effects on vehicle weight by analyzing the resulting total Earth departure ΔV's. Finally, as mentioned previously, the launch azimuth constraints limit the range of acceptable orbit inclinations. The actual tradeoff effects of these parameters are shown in the subsequent discussion of the launch window analysis results.

The computer program used first computes the parking orbit inclination which will place $\overline{V}_\infty$ in the plane of the orbit on the first and last days of a selected 20-day Earth window. At this point in the analysis the 20-day window is selected so that the optimum launch date (minimum weight vehicle) lies at or near the center of the window. The program uses as input the specified launch window duration and the values of α, δ, and V_∞ magnitude for each date in the window. The method of Deerwester (Ref. 9) is then used to compute the minimum plane change requirements to transfer optimally from the parking orbit to the departure hyperbola for each day in the window. Figure IV-11 illustrates the



VIEW OF PLANE OF DEPARTURE HYPERBOLA



PROJECTION OF TRAJECTORY PLANES ON UNIT SPHERE

Figure IV-11. Departure Geometry from Parking Orbit departure geometry and the tradeoffs which give rise to the need for the optimization. On any day during the launch window there will be an angle ϕ between the $\overline{V}_\infty$ vector and the orbit plane (for purposes of the analysis it makes no difference whether $\overline{V}_\infty$ is above or below the plane). (The angle ϕ is equal to zero on the first and last days of the launch window.) In order to transfer to the departure hyperbola, the vehicle's velocity magnitude must be increased from its circular parking orbit velocity to the appropriate velocity on the departure hyperbola and the velocity direction must be turned through an angle so that the vehicle's velocity has the proper direction on the hyperbola. There is a point P on the given parking orbit which minimizes the total angle turned through and which, therefore, will also minimize the total impulsive velocity change required to transfer from the parking orbit to the departure hyperbola. Since the angle turned through consists of two components — the plane change angle, ψ, and the flight path angle, γ, an optimization is required to determine the location of point P on the parking orbit such that the associated values of ψ and γ yield the minimum angle turned through.

The minimum angles turned through are determined for each day in the launch window and from these are computed the total impulsive departure ΔV's for a launch from the parking orbit whose plane contains the $\overline{V}_\infty$'s on the first and last days of the 20-day Earth window. Next, a series of similar computer runs are made for a matrix of cases in which both the initial depart date of the window as well as the two dates on which the $\overline{V}_\infty$ lies in the plane of the orbit are incrementally varied. These two parameter variations, respectively, have the effect of varying the range of departure asymptote right ascensions, declinations, and V_∞ magnitudes; and the resulting orbit inclination.

The one combination of initial depart date and orbit inclination within this matrix that yields the set of total impulsive departure ΔV's which would result in the lowest vehicle weight would then be considered the optimum. However, in this analysis it was sufficient to plot the total impulsive ΔV's as a function of depart Earth date and select as optimum the set that contained the minimum/maximum ΔV within the 20-day window. This set of total ΔV's was then used in resizing the vehicle for each day of the window at Earth. A similar procedure was employed for determining the launch window conditions at Mars.

Strictly speaking, the initial computer run described in this procedure could have been included within the matrix of cases that was subsequently analyzed. However, the results of the initial run were useful in defining the range of the parameter variations in the matrix, i. e., the direction in and extent to which the initial depart date should be varied as well as the probable effects the launch azimuth constraints would have on the final selection of the optimum orbit plane.

<u>Vehicle Sizing Analysis</u>. The results obtained for a typical mission after the plane change ΔV's are included and the vehicle is resized are shown in Figure IV-12. Plotted on this graph is the ratio of propellant required for each vehicle stage (after the plane change ΔV's are added) to the propellant required for the "optimum" mission (without any plane change ΔV), as a function of the depart Earth date in the 20-day launch window.

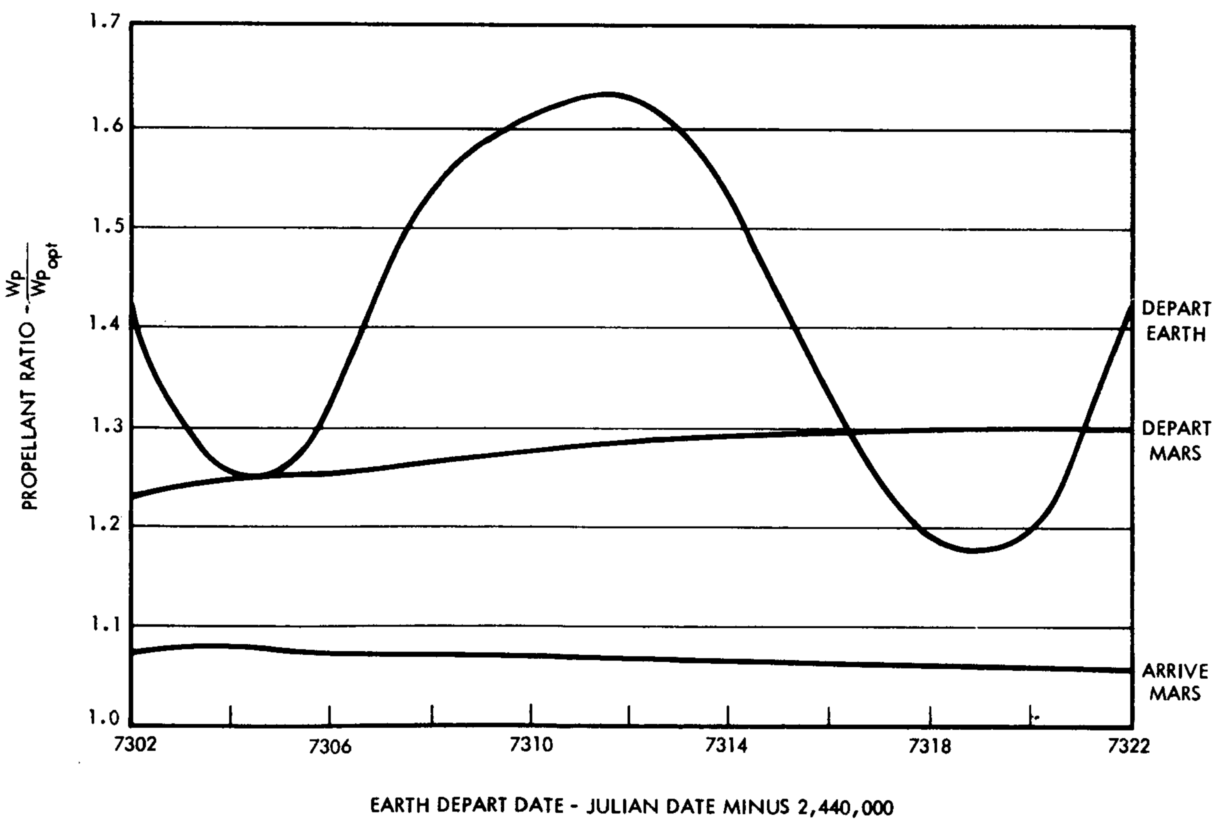


Figure IV-12. Typical Stage Propellant Requirements

The next step in the vehicle sizing procedure is to note in Figure IV-12 that the leave Mars stage must be sized for the maximum amount of propellant required during the launch window. Consequently, the tank structural weight of this stage is fixed at the value corresponding to the maximum required propellant capacity for that stage and the vehicle is again resized. The result of this computation increases the propellant requirements for all stages and a new set of propellant curves is obtained across the launch window.

A similar step is then made for the arrive Mars stage, fixing its tank weight to correspond to its maximum propellant requirement. A final step is required to fix the leave Earth stage tank weight to match its maximum propellant capacity.

Throughout this iterative procedure, consideration must be given to the fact that the propellant carried in each stage must reflect the subsequent mission propellant requirements, That is, once the vehicle is injected on its interplanetary trajectory on a particular date, only that amount of propellant need be retained in the remaining stages that could possibly be required for that committed mission. Also, until an injection out of the planetary parking orbit is completely accomplished, sufficient propellant must be maintained in the injection and other remaining stages to permit a launch on a subsequent date.

Finally, since the propellant requirements of the various stages increase in each of the above steps, the engine firing times also increase with each progressive vehicle sizing step. Since a fixed engine clustering arrangement is selected initially, the resulting engine firing times must be continually compared with the 2700 sec. constraint. Should the firing time for a given stage exceed this limit at some point in the iterative procedure, a complete iteration of the analysis would be required, including the determination of new parking orbit parameters and plane change requirements.

Results and Discussion

The vehicle weight results obtained from the launch window analysis are given in Table IV-5 for the three missions analyzed. Also given in

Table IV-5. Launch Window Analysis Results

Mission	Optimum			20-Day Launch Windows		
	Vehicle Weight (10^6 lb)	Configuration	Number Modules	Vehicle Weight (10^6 lb)	Configuration	Number Modules
Opposition Class 1984 IB	1.830	3-1-1	6	2.913	4*-1*-1*	11
Outbound Venus Swingby 1986 3A	1.936	3*-1*-1	8	2.453	4*-2-1	10
Inbound Venus Swingby 1988 II3	1.545	2*-1-1	7	2.066	4-1-1	7

*Tier 2A required

this table are the vehicle weights for the optimum or minimum weight missions when no launch windows are considered. In addition to vehicle weights, the data in the table indicate the number of engines required per stage to observe the 2700 sec firing time limit and the number of modules constituting the total spacecraft. These modules are the propulsion modules, the propellant modules, and the module containing the payloads and secondary systems. A brief discussion of these results follows for each of the launch opportunities analyzed.

Opposition Class Mission, 1984 IB. The launch window penalty incurred for the opposition class, 1984 IB mission was the largest of all three missions analyzed. The initial vehicle weight was increased by approximately 59 percent from the optimum (no launch window) mission for this opportunity and trajectory type.

The optimum (minimum weight) mission has an Earth depart Julian date of 2,445,691 and requires a vehicle weighing approximately 1.83 million pounds. The 2.91 million pound vehicle required for providing launch windows at Earth and Mars would be capable of departing Earth on any day from 2,445,689 through 2,445,709 and stopping over at Mars for any period from 30 to 50 days. In addition to the increased vehicle weight requirements, the launch window vehicle requires one additional engine for departing Earth and five additional propellant modules as indicated in Table IV-5. Although a 2700 sec firing time constraint was used in the launch window analyses, the firing time for the four-engine Earth depart stage in this case exceeded this constraint by 240 seconds. In order to meet the constraint, five engines would be required (5-1-1 configuration) and the vehicle weight would probably increase by 30,000 to 40,000 pounds.

Figure IV-13 presents the results of the tradeoff analysis among the parking orbit inclination, the orientation of the parking orbit line of nodes, and the initial Earth depart date. For a launch window centered around the optimum (no window) depart date, an orbit inclination of 31.79 deg is required and the resulting maximum total depart Earth velocity requirement reaches a relatively high value (5.64 km/sec).

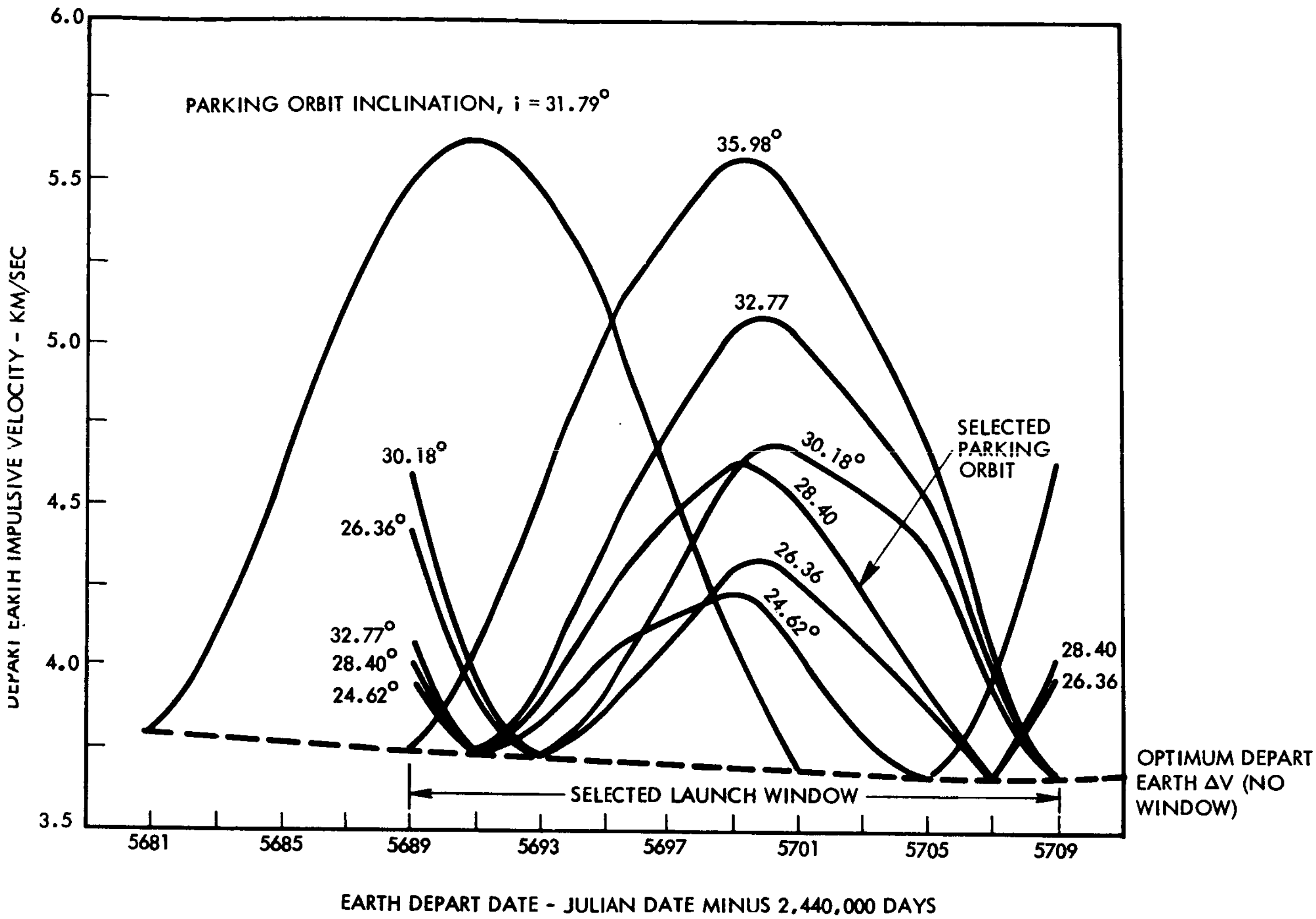


Figure IV-13. Earth Launch Window Velocity Requirements, Opposition Class Mission, 1984 IB

By shifting the initial date of the 20-day launch period towards a later date, reducing the orbit inclination to the minimum possible (28.4 deg.), and orienting the longitude of the line of nodes so that a small plane change is required on the first and last date of the launch window, the maximum total depart Earth velocity can be reduced to approximately 4.64 km/sec. The effect of shifting to later dates is to reduce the heliocentric depart Earth velocity requirements as well as to reduce the declinations of the hyperbolic asymptotes. The latter effect permits the use of a lower orbit inclination, which increases the time rate of precession for the orbit and in turn permits a more suitable orientation of the line of nodes. It might well be argued that the launch window depart dates might be further delayed, since both the magnitude and the declination of the hyperbolic velocity vectors are becoming smaller in that direction. However, an investigation of this possibility for this launch opportunity and trajectory type indicated that for Earth launch dates after approximately 2, 445, 710 both the arrive Mars and depart Mars velocities are increasing at such a rapid rate as to far outweigh any weight reduction obtained from the slightly lower total Earth velocities available in this region. These conditions occur because for this mission the range of dates falls close to the ridge area of the trajectory map in the region of high and rapidly increasing velocities. This is also the reason for the relatively high vehicle weight increase required for providing a launch window for this mission compared to the two other missions analyzed.

Although the above trajectory conditions pertain only to the 1984 IB opposition class mission, additional factors exist to explain the weight increases required for providing launch windows for this and the other two missions analyzed (27 percent for the 1986 3A swingby mission and 34 percent for the 1988 II3 swingby mission). First, it is clearly the case that the provision of launch windows will increase the vehicle weight to some extent. However, as the vehicle weight increases with each additional day in the launch window, a point is reached where the engine firing time for one or more stages exceeds the firing time constraint and an additional engine(s) is required. Also as the vehicle weight increases, a point is reached where the propellant required for one or more stages exceeds the specified capacity for a module and an additional module(s) must be provided. Either of these conditions tends to produce quantum or discrete jumps in the total vehicle weight due to a decreased stage mass fraction resulting from respectively, either the addition of a relatively heavy engine or the addition of a small capacity propellant module.

Second, in the vehicle sizing procedure the individual stages must be sized to provide sufficient propellant to accomplish the mission on any date within the launch window, and this has a large effect on the resulting vehicle weight. The variations in propellant requirements for the individual stages can be examined over the launch window dates to gain an insight into the relative effect this has on the provision of a launch window. Figure IV-14 presents the ratio of propellant required for each vehicle stage (after the plane change ΔV is added) to the propellant required for the

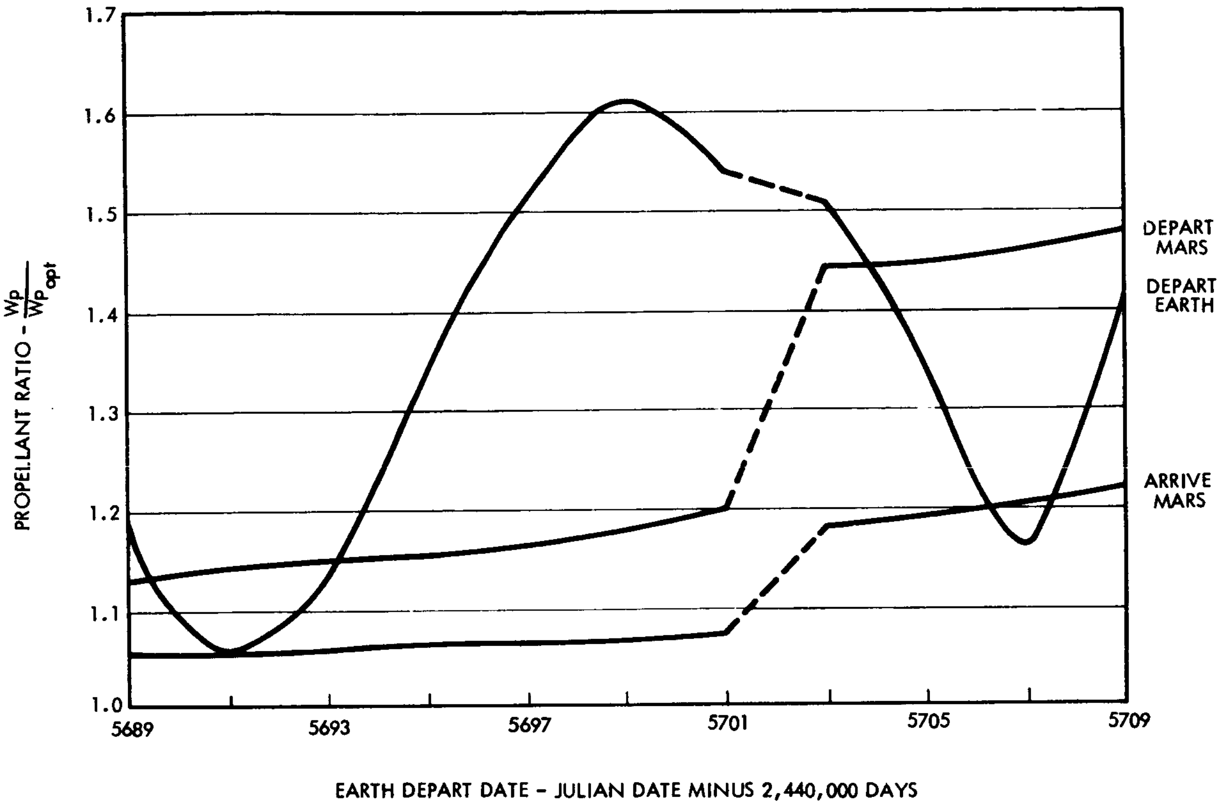


Figure IV-14. Stage Propellant Requirements, Opposition Class Mission, 1984 IB

optimum mission (without any plane change ΔV), as a function of the depart Earth date in the selected launch window.

It should be recalled that in the analysis procedure previously outlined, it was shown that for a launch on a particular date it is necessary to maintain sufficient propellant in the injection and other remaining stages to permit a launch on a subsequent date. With this fact in mind, it can be seen on Figure IV-14 that for the opposition class mission the depart Mars and arrive Mars stages must be sized for the propellant requirements occurring on the last day of the launch window. In addition, this amount of propellant must be carried through the leave Earth injection phase of the mission for all earlier launch dates. A similar condition is also true for the maximum propellant requirement of the depart Earth stage which occurs near the center of the window. It should also be noted that by fixing the weight of the depart Mars stage to correspond with its maximum required capacity occurring on the last day, the propellant requirements of all stages, for earlier launch dates, are increased when the vehicle is resized to account for this fixed stage weight.

The combined effects of these vehicle sizing characteristics with the interplanetary and plane change energy requirements shown previously in Figure IV-13 serve to explain the comparative launch window results obtained for the three types of mission classes.

Venus Swingby Mission, 1986 3A. The provision of 20-day launch windows at Earth and Mars for the 1986 3A Venus swingby mission resulted in the smallest vehicle weight increase of all three missions analyzed, approximately one-half million pounds or 27 percent of the optimum (no window) mission for this launch opportunity. However, two engines were required for the arrive Mars phase (the use of one engine would have violated the firing time constraint by about 100 to 200 seconds). Two additional propellant modules were required over the module requirements for the optimum mission. The optimum (no window) launch date occurs on Julian date 2, 446, 150 and the 20-day Earth launch window ranges from 2, 446, 140 to 2, 446, 160.

Figure IV-15 presents the results of the Earth parking orbit tradeoff analysis. A launch window centered around the optimum depart date and requiring no plane changes on the first and last days in the window required a parking orbit inclination of 32. 8 deg. and a maximum depart Earth velocity of 4. 87 km/sec. With a 28. 4 deg orbit inclination, the maximum velocity could be reduced to 4. 36 km/sec. In this case, a shift in the initial launch date to the region of lower declinations of the hyperbolic asymptotes (earlier, to the left on the graph) did not produce lower total velocity requirements. This was due to the fact that the optimum depart Earth velocities (no window) were increasing at a slightly greater rate than the plane change requirements were decreasing.

The variations in the stage propellant requirements across the launch window are given in Figure IV-16. It is noted that the arrive and depart Mars stage propellant requirements are practically flat across the launch window and the variation in the depart Earth propellant requirement is considerably smaller than for the 1984 IB opposition class mission and (as will be shown) for the 1988 II3 Venus swingby mission. Therefore

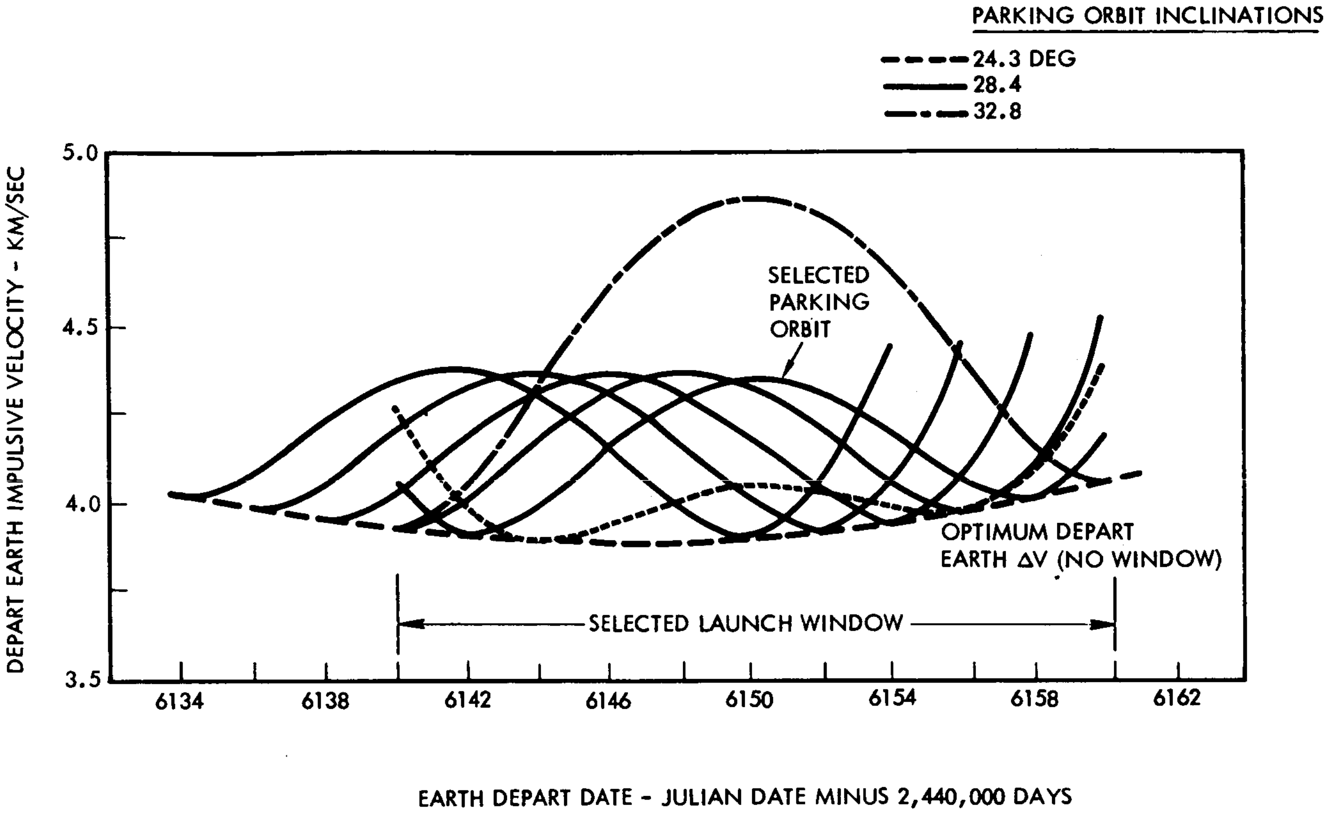


Figure IV-15. Earth Launch Window Velocity Requirements, Venus Swingby Mission, 1986 3A

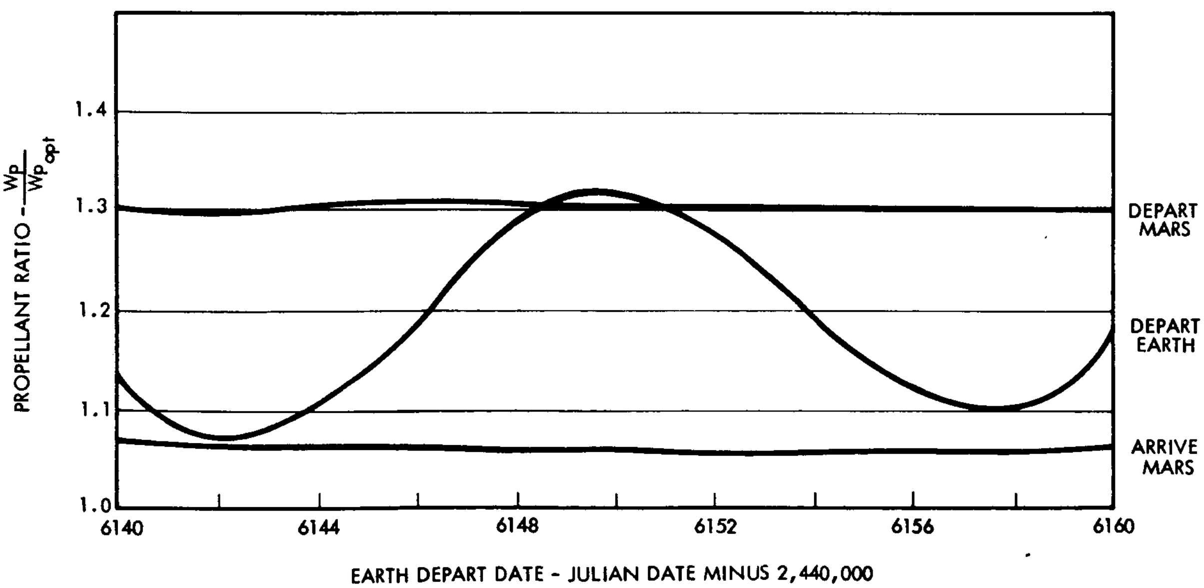


Figure IV-16. Stage Propellant Requirements, Venus Swingby Mission, 1986 3A the effects on the total vehicle weight of fixing the tank capacities and providing excess propellant loading are considerably less, accounting for the relatively small weight increase required for the launch window provision for this mission.

Venus Swingby Mission, 1988 II3. The initial vehicle weight for the 1988 II3, Venus swingby mission increased from approximately one and one-half million pounds to slightly over two million pounds (34 percent) when the Earth and Mars launch windows were provided. Two additional engines were required for the depart Earth stage but the same number of modules were employed. The optimum (no window) launch date occurs on Julian date 2, 447, 340 while the selected 20-day Earth launch window ranges from 2, 447, 302 through 2, 447, 322.

The Earth parking orbit tradeoff results for this mission are presented in Figure IV-17, which shows that the selected 20-day window occurs considerably earlier (to the left) than the optimum (no window) launch date (2, 447, 340). For this mission the declinations of the hyperbolic asymptotes in the region of the optimum launch date are quite small, less than 10 deg. Since the minimum available parking orbit inclination is 28.4 deg, the plane change velocity requirements for a 20-day window occurring in this region become prohibitively large. Therefore, in this case it was necessary to shift the initial launch date to an earlier launch since a shift to an earlier date resulted in higher hyperbolic asymptote declinations and therefore in smaller plane change requirements.

The variations in the stage propellant requirements across the selected launch window are given in Figure IV-18.

It should be pointed out that the quantitative results obtained in this launch window analysis task are strongly influenced by the assumptions and constraints listed at the beginning. Nevertheless, the results obtained in this study task indicate that the provision of launch windows can impose severe problems on future manned interplanetary mission design and operation. Therefore, it is strongly felt that additional analysis is required to fully explore all of the launch window implications uncovered in this study. Specifically, these analyses should include investigations of dual or multiple impluse and staging for the Earth injection stages coupled with elliptic orbits at Earth; the use of propellant transfer

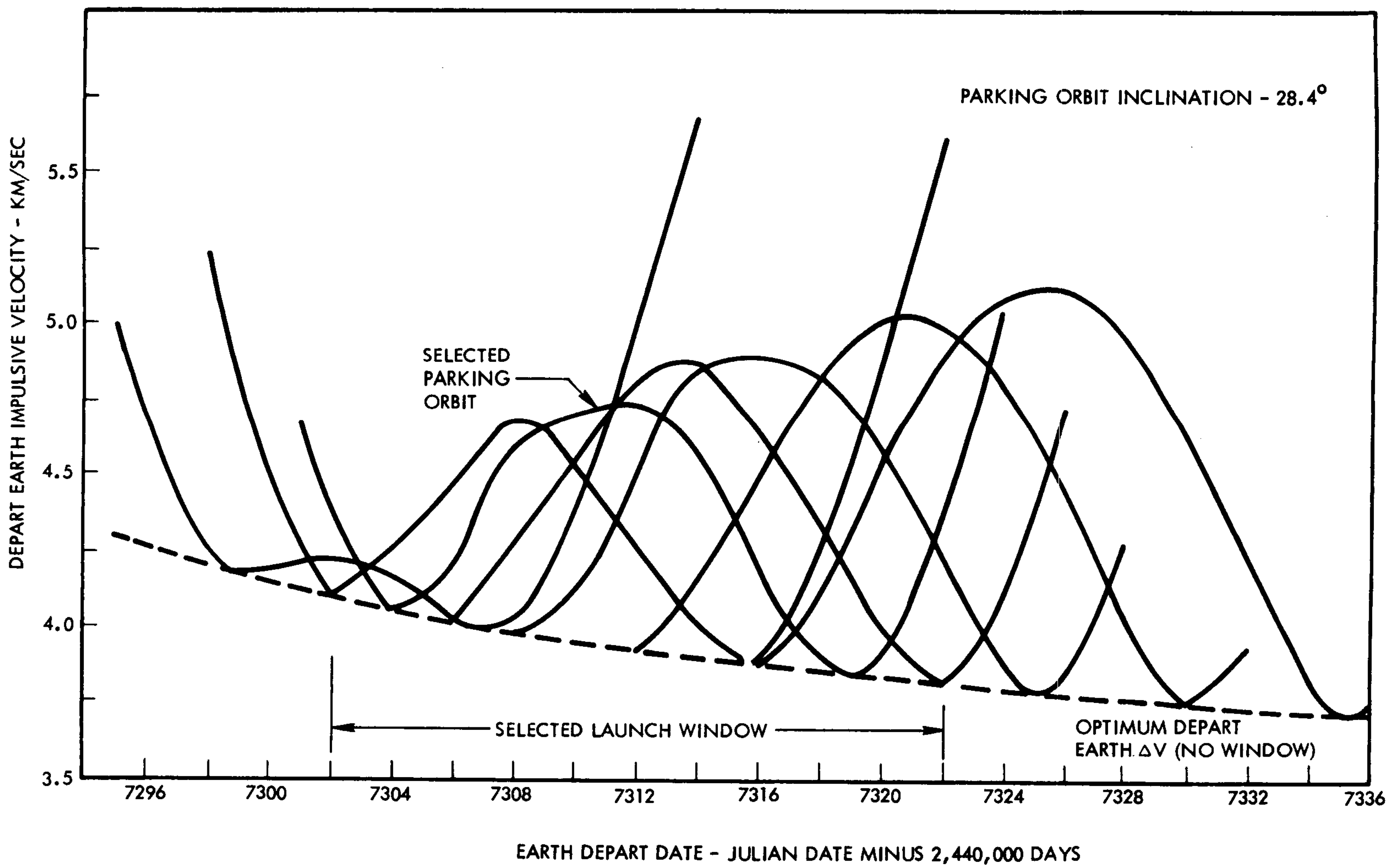


Figure IV-17. Earth Launch Window Velocity Requirements, Venus Swingby Mission, 1988 II3 between stages; the analysis of combined plane changes and "nonoptimum" interplanetary orbits; and a study of the launch window effects of Mars elliptic parking orbits. In addition, a separate investigation is required to bracket closely the range of Earth and Mars window durations that will be necessary from an operational standpoint.

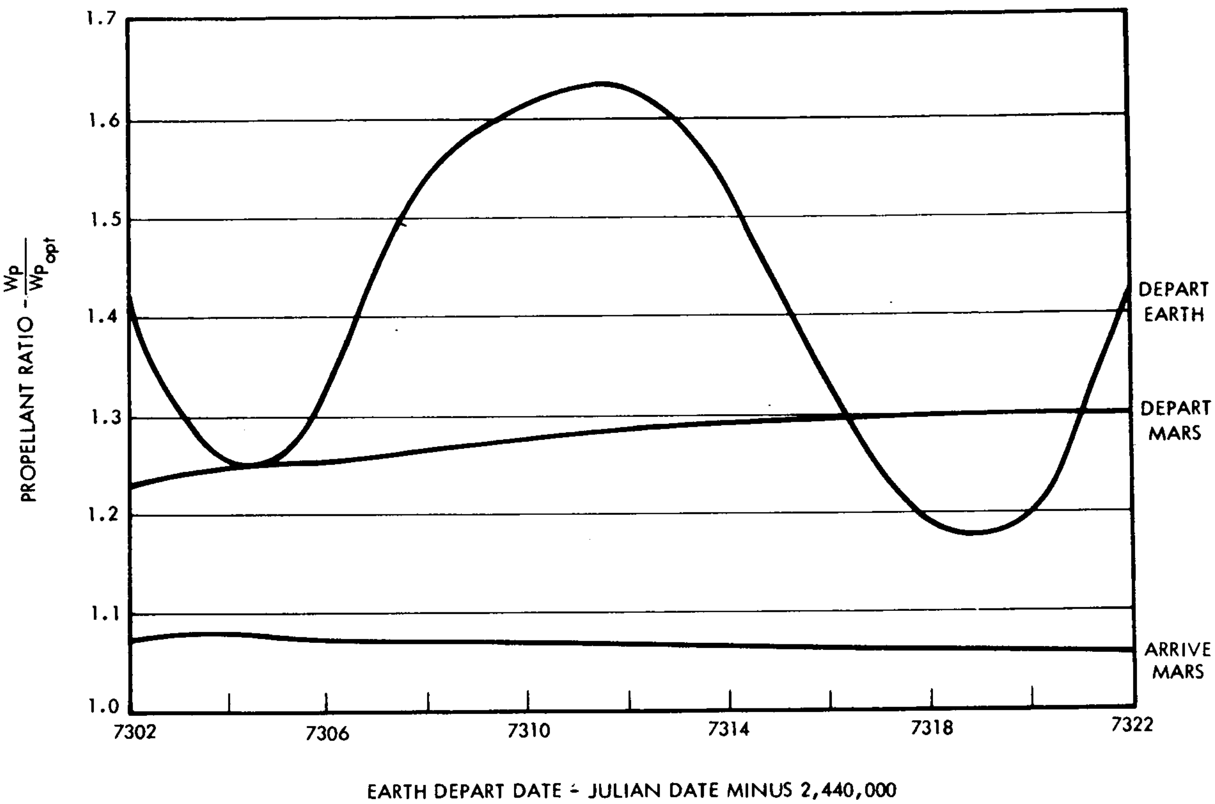


Figure IV-18. Stage Propellant Requirements, Venus Swingby Mission, 1988 II3

V. FUTURE RESEARCH AND ADVANCED TECHNOLOGY

From the results of this study several areas requiring future research and advanced technology development can be identified. First, additional analytical studies are necessary to investigate in greater detail and rigor various operational requirements and alternatives; and secondly, technology studies and developments should be initiated on a timely basis to assure availability of data and equipment necessary to the successful design and performance of the mission.

LAUNCH WINDOWS

Although the quantitative results obtained in this study were strongly influenced by the specific assumptions and constraints, it is clear that the provision of launch windows can, in general, impose severe problems on future manned interplanetary mission design and operational requirements. Therefore, additional analysis is required to fully investigate the various launch window problem areas, possible operational modes, and operational implications.

Specifically, these analyses should include investigations of dual or multiple impulse and staging for the Earth injection stages coupled with elliptic orbits at Earth; the use of propellant transfer between stages; the analysis of combined plane changes and "nonoptimum" interplanetary orbits; and especially a study of the launch window effects of Mars elliptic parking orbits. Additional analysis also should be performed to determine the overall vehicle and stage weight requirements for parametric variations of Earth launch holds. The results of this latter analysis will not only indicate the amount of additional weight required in Earth orbit but will also provide information as to what constitutes reasonable launch windows and the sensitivity of these launch windows to system and performance variations.

All of these analyses should be extended to include the entire range of launch opportunities for both Mars and Venus missions as well as alternative types of trajectory profiles and classes of missions. Finally, a separate investigation is required to bracket closely the range of Earth and Mars window durations that will be necessary from an operational standpoint.

ELLIPTIC PLANETARY PARKING ORBITS

Since the results of this study indicated that a large potential weight saving is available through the use of elliptic parking orbits at the target planet, efforts should be continued to develop a better understanding of elliptic versus circular parking orbits at the target planets. In addition to studies of the effects the use of elliptic orbits will have on launch window provisions, analyses should be conducted to: determine the requirements on lander payloads ejected from and returned to the planetary orbit; investigate the requirements of orbit adjustment systems including multi-impulse maneuvers and rotation of the line of apsides to reduce the overall arrive and depart energy requirements; investigate variations in orbital parameters such as the apsidal ratio and periapsis radius; and investigate a range of launch opportunities and alternative types of trajectory profiles and classes of missions. In addition, analyses should be made to determine both the system requirements and operational implications the use of elliptic parking orbits will impose on navigation and guidance systems and accuracies.

EARTH LAUNCH OPERATIONS AND SYSTEMS

In this study, as in most past studies, the interplanetary vehicles were configured on the basis of a nominal Saturn vehicle payload criterion. As the vehicle systems and configurations become more decisively defined, the use of nominal Saturn payloads will be inadequate and could lead to critical errors in the design and formulation of the overall interplanetary spacecraft.

The maximum useable payload that the Saturn vehicle can place in the Earth parking orbit has a profound impact on the resulting interplanetary spacecraft by in effect determining for any given mission the number of spacecraft modules that must be launched, rendezvoused, docked, and assembled. In turn, the number of modules required has a significant effect on the overall spacecraft weight, the stage jettison weights, the number of tanks and engines, the vehicle docking and assembly procedures and the associated weight penalties, the vaporized propellant, the launch scheduling, etc.

The specification of the Saturn payload capability should be based upon projections to the mission time period. In addition, the Earth parking orbit characteristics such as type (circular or elliptic), altitude, and inclination must also be considered. Since the analysis of Earth launch windows and launch azimuth constraints (declination restrictions) also involve these latter parking orbit parameters, the determination of the Saturn payload capability cannot be disassociated from the interplanetary mission analyses.

In summary, in order that the results of future analyses of manned interplanetary missions be completely valid, the areas of Saturn payload capability, launch window provisions, and launch azimuth constraints must be considered simultaneously.

VEHICLE CONFIGURATION AND OPERATION

All propellant tanks in this study were sized on the basis of continuous function scaling laws. A limitation was placed on the maximum capacity of any tank and if this were exceeded, an additional tank or tanks were added to the tank cluster and sized to contain the exact amount of propellant required. Although, for any given set of tank scaling laws, this method of sizing tanks will produce the minimum weight vehicle, in actuality it is doubtful that the many different size tanks which would be required could practically be designed, fabricated, and tested. Therefore, additional analysis should be performed for selected missions to determine the tradeoffs available and the vehicle weight penalties associated with the use of a series of propellant tanks of fixed but graduated sizes. The number of different sizes in the series should be varied parametrically.

In addition, several new concepts of vehicle staging have been proposed recently and should form the basis of additional analysis. One of these concepts is called the nonintegral burn (Reference 10). In this concept, a standard stage size is used and the total number of stages employed is simply matched to the total propulsion requirements of the mission. Restart of a stage is permitted to perform all or part of successive propulsive maneuvers of a mission. A variation of this concept (Reference 11) employs standard or common modules but propellant is transferred during the engine thrusting from a forward donor tank to the thrusting stage. Both of these concepts strive to reduce the required total number of Earth launches, make maximum utilization of the Saturn payload capability, and reduce the overall cost of the mission.

AERODYNAMIC EARTH BRAKING

Another area in which additional research has been required for some time is that of aerodynamic braking at Earth. The arrival velocities at Earth for the opposition class missions can be as high as 20 km per sec. Therefore, the determination of the maximum aerodynamic braking capability for the mission time period is critical. For spacecraft arriving at the greater velocities, a retro stage must be employed with its attendant increase in vehicle weight over an all aerodynamic braking stage. The determination of the Earth aerodynamic braking capability requires an intensified research and development program to advance the state-of-the-art past the currently planned Apollo technology.

ENGINE FIRING TIME

Finally, the constraint of a maximum engine firing time imposes the necessity of increasing the number of required engines which nearly always increases the initial vehicle weight. Also the use of many engines in a stage cluster tends to compromise the vehicle configuration in terms of staging, assembly and docking, tank module requirements, propellant transfer, and cost. The effects of an engine firing time limitation can be particularly severe if lower thrust engines are contemplated and in cases in which engine aftercooling and subsequent restart is used. Both theoretical and experimental efforts should be conducted in fuel element technology and reactor and engine design and testing to attempt to achieve reliable firing times of one or two hours.

VI. CONCLUSIONS

Although the Phase VI study was segmented into two parts on the basis of nuclear engine thrust level, results can be abstracted from both parts and combined into separate categories from which several significant conclusions can be drawn.

TRAJECTORY TYPES AND LAUNCH OPPORTUNITIES

For the launch opportunities existing during the period from 1980 to 1993, the criteria of minimum initial vehicle weight and launch azimuth constraints led to the selection of certain trajectory types for each of the launch opportunities if opposition class missions were employed. In the years 1980, 1984, 1986, 1990, and 1993, a type IIB trajectory requires Earth departure declinations which are outside of the allowable range dictated by the launch azimuth constraints. For these opportunities a type IB trajectory requires vehicles weighing slightly more than those for a type IIB trajectory, but the required declinations satisfy the launch azimuth constraints. In the two remaining years (1982 and 1988) the type IIB trajectory both meets the launch azimuth constraints and yields minimum weights and trip times. (It should be noted that the 1984 IIB missions require declinations ranging from -36 to -37 deg, which border just on the acceptable boundary limit.)

Of the Venus swingby missions analyzed in the 1980 to 1993 time period, only the 1984 II5 and the 1990 5A and 5B violate the launch azimuth constraints. In all but two launch opportunities, the minimum weight vehicle is obtained when the swingby trajectory (type 3 or 5) is combined with the long direct leg (type I or B); in the years 1986 and 1988 the use of the short direct leg (type A and II, respectively) results in a mission with both a lower weight vehicle and a shorter total trip time.

The minimum vehicle weights for the Venus orbital capture missions were obtained for the type IIB inferior conjunction trajectory profile for all launch opportunities from 1980 to 1985. None of these trajectories violated the launch azimuth constraints.

An overall selection of mission and trajectory types can be made for the 1980 to 1993 time period on the basis of the criteria of minimum vehicle weight and launch azimuth constraints. In all years except 1984 and 1990 the Venus swingby missions require a lower vehicle weight than the opposition class missions and also meet the launch azimuth constraints. In 1984 the IB opposition class mission requires a lower weight vehicle than the swingby mission; in 1990 the Venus swingby missions violate the launch azimuth constraints and therefore, the 1990 IB opposition class mission becomes the necessary alternative with its slightly lower weight requirement and acceptable Earth departure declination.

VEHICLE MODE COMPARISONS

The results obtained in comparing the vehicle weight requirements for the various vehicle modes analyzed are typical of those obtained in earlier analyses in which similar mission and vehicle parameters were assumed. The assumption of a maximum of 15 km/sec Earth aerodynmic braking capability increases the initial vehicle weight requirements by a factor of 1. 5 to 2 over the weight required for unlimited aerodynamic braking for the unfavorable launch opportunities in the synodic cycle, and the use of the nuclear engine results in vehicles weighing approximately 50 percent of those utilizing chemical cryogenic stages.

INITIAL VEHICLE WEIGHTS

For the Mars lander missions employing a NNNA, 3-1-1 vehicle configuration and 200, 000-pound thrust nuclear engines, the vehicle weight requirements range from 1. 53 to 2. 51 million pounds for the 1980 to 1993 launch opportunities. This weight range is for the selected mission and trajectory types for each opportunity that yield minimum vehicle weights and acceptable Earth departure declinations. The minimum vehicle weight mission occurs in 1988 (Venus swingby, II3); the maximum in 1990 (opposition class, IB).

The minimum NNNA vehicle weights required for performing orbital capture missions to Mars or Venus during each respectively available launch opportunity range from 1. 44 to 1. 65 million pounds. A comparison of the minimum weight results for the Mars orbital capture missions with the results for the Mars lander missions shows that the lander missions require approximately 170, 000 to 350, 000 pounds more vehicle weight than the orbital capture missions.

PLANETARY STOPOVER TIMES

A variation in planetary stopover time from 20 to 40 days for the opposition class orbital capture missions in the 1980 to 1984 time period produces a six to nine percent increase in vehicle weight for each 10-day increase in stopover time. An identical variation was obtained for the minimum weight Venus swingby trajectory in 1984; in 1980 and 1982 the minimum weight swingby trajectories required less than one percent, increase or decrease, in vehicle weight for each 10-day increase in stopover time.

The variation of initial vehicle weight with varying Venusian stopover time for the 1980 and 1982 Venus orbital capture missions was an approximate two to four percent increase with each-10 day increase in stopover time; in 1983 and 1985 there was no significant variation in initial vehicle weight for Venusian stopover periods from 20 to 40 days.

THRUST LEVELS

An increase in the nuclear engine thrust level from 75, 000 to 100, 000 pounds reduces the initial vehicle weight requirements by 3 to 10 percent depending on the launch opportunity and mission type, i. e., lander or orbital capture. A somewhat greater weight reduction, 6 to 13 percent, is obtained if the thrust level is increased from 100, 000 to 200, 000 pounds. By employing 200, 000-pound thrust engines in lieu of the 75, 000-pound thrust engines, the weight reductions range from 9 to 16 percent. At least one less engine and at most three fewer engines are required by the vehicle for each incremental increase in thrust level, i.e., from 75, 000 pounds to 100, 000 pounds or from 100, 000 pounds to 200, 000 pounds.

NUCLEAR ENGINE AFTERCOOLING

For the mission analyzed (1984 IIB, opposition class, orbital capture mission), a maximum reduction of only six percent was available by resorting to engine aftercooling at Mars; the vehicle weight for the $NN_{NJ}NA$ case, in fact, increased slightly over the nonaftercooled case. In addition the arrive Mars engine firing times increased substantially for the aftercooled cases, actually exceeding the 2700 sec limit for the circular orbit cases for both the NN_JNA and $NN_{NJ}NA$ vehicle modes.

ELLIPTIC ORBITS

A sizeable 28 to 37 percent decrease in initial vehicle weight is obtained if elliptic Mars parking orbits are employed instead of circular orbits for the 1984 opposition class, orbital capture missions. These vehicle weight reductions were obtained regardless of the vehicle mode or engine thrust levels employed and for both the aftercooled and nonaftercooled engine modes.

LAUNCH WINDOW

The provision of 20-day launch windows at Earth and Mars increases the vehicle weight requirements by 27 to 59 percent for the three lander missions investigated. The largest launch window penalty (59 percent) was incurred by the 1984 IB opposition class mission; the smallest (27 percent) by the 1986 3A Venus swingby mission. The vehicle weight for the 1988 II3 swingby mission increased by 34 percent when Earth and Mars launch windows were provided. The addition of two nuclear engines is required for all three mission vehicles in order to maintain the maximum engine firing times below a 2700 sec limit.

These launch window penalties are a function of the trajectory type and class of mission employed and the launch opportunity year, and therefore would be expected to vary for other opportunities by at least the range obtained for the three cases investigated in this study.

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APPENDIX A

CONNECTING MODE SCALING LAWS

Given in this appendix are the structural scaling laws used for computing the propulsion and propellant module weights for the propulsion systems employing 75,000- and 100,000-pound thrust nuclear engines. These scaling equations were used in the Phase B analyses. Tables A-1 through A-22 contain the module jettison weight equations and stage weights used for the various mission phases and engine clustering arrangements. Due to varying interstage structure requirements for the different engine clustering arrangements, each clustering configuration has a corresponding set of scaling equations.

Table A-1. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 2-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	12,472	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-2. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 2-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	20,462	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-3. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 3-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,859	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-4. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 3-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	23,850	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-5. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 3-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	31,841	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-6. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 4-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 120,880$	1,026,748
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	19,246	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-7. Connecting Mode Scaling Laws 75,000-Pound Thrust Nuclear Engine 4-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 120,880$	1,026,748
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	27,237	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-8. Connecting Mode Scaling Laws 75, 000-Pound Thrust Nuclear Engine 4-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 120,880$	1,026,748
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	35,228	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-9. Connecting Mode Scaling Laws 75, 000-Pound Thrust Nuclear Engine 5-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 151,100$	1,283,435
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	22,633	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-10. Connecting Mode Scaling Laws 75, 000-Pound Thrust Nuclear Engine 5-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 151,100$	1,283,435
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	30,624	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 60,440$	513,374
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-11. Connecting Mode Scaling Laws 75, 000-Pound Thrust Nuclear Engine 5-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 151,100$	1,283,435
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	38,615	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 90,660$	770,061
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 30,220$	256,687
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-12. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 2-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	12,472	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-13. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 2-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	20,462	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-14. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 3-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,859	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-15. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 3-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	23,850	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-16. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 3-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	31,841	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-17. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 4-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 112,880$	1,018,832
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	19,246	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-18. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 4-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 112,880$	1,018,832
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	27,237	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-19. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 4-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 112,880$	1,018,832
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 87,304$	1,098,344
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	35,228	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-20. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 5-1-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 141,100$	1,273,540
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	22,633	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	15,349	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-21. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 5-2-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 141,100$	1,273,540
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	30,624	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 56,440$	509,416
Tier 2A	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 2B	$W_j = 0.08085\ W_p + 43,652$	549,172
Tier 3	$W_j = 0.08085\ W_p + 43,652$	
Stage Constant	21,613	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	

Table A-22. Connecting Mode Scaling Laws 100,000-Pound Thrust Nuclear Engine 5-3-1 Vehicle Configuration

Mission Phase	Equation (lb)	Maximum Capacity (lb)
Earth Depart		
Tier 1	$W_j = 0.08085\ W_p + 141,100$	1,273,540
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	38,615	
Planet Braking		
Tier 1	$W_j = 0.08085\ W_p + 84,660$	764,124
Tier 2A	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 2B	$W_j = 0.08085\ W_p + 65,478$	823,758
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	27,877	
Planet Depart		
Tier 1	$W_j = 0.08085\ W_p + 28,220$	254,708
Tier 2	$W_j = 0.08085\ W_p + 21,826$	274,586
Tier 3	$W_j = 0.08085\ W_p + 21,826$	
Stage Constant	5259	