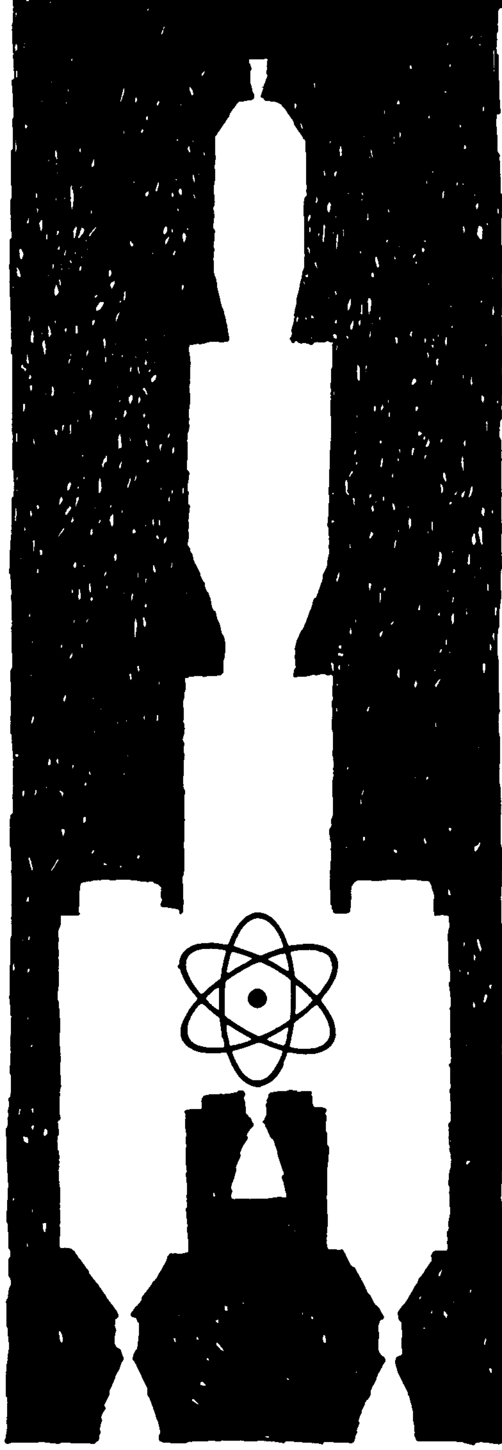
MISSION ORIENTED ADVANCED NUCLEAR SYSTEM PARAMETERS STUDY

DECEMBER 1966
PHASE IV FINAL REPORT



VOLUME I

01977-6005-R000

SUMMARY TECHNICAL REPORT

FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

BY

TRW SYSTE
AN OPERATING GROUP OF TRW

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MISSION ORIENTED STUDY

OF

ADVANCED NUCLEAR SYSTEM PARAMETERS

Phase IV. Final Report

Volume I

Summary Technical Report

for

George C. Marshall Space Flight Center

National Aeronautics and Space Administration

by

TRW Systems

Redondo Beach, California

Volume I	Summary Technical Report
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FOREWORD

This volume, which is one of a set of three volumes, summarizes the study tasks, analyses, and results that were accomplished under Contract NAS8-5371, Mission Oriented Study of Advanced Nuclear System Parameters, for George C. Marshall Space Flight Center, Huntsville, Alabama. This work was performed during the period from May 1965 to December 1966 and covers Phase IV of the subject contract.

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

Volume I	Summary Technical Report
Volume II	Detailed Technical Report
Volume III	Research and Technology Implications Report

Volumes I and II include, respectively, a summary and 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. Volume III delineates those areas of research and technology in which further efforts would be desirable based on the results of the study.

The principal contributors to this study were Messrs. A. R. Chovit, R. D. Fiscus, and L. D. Simmons. In addition, Dr. C. D. Kylstra, in a consulting capacity, provided technical support on computer program revisions.

Also the assistance given by the following persons is gratefully acknowledged: Dr. R. K. Plebuch and Messrs. W. H. Bayless, G. W. Cannon, H. W. Hawthorne, G. Rosler, and R. L. Sohn, TRW Systems; Mr. C. D. McKereghan, Lockheed Missile and Space Division; Mr. P. G. Johnson, SNPO-W; and R. J. Harris, W. Y. Jordan, and D. R. Saxton, MSFC.

ABSTRACT

A summary of the study approach and basic guidelines and assumptions which were used in a series of analyses of manned Mars stopover missions is given. Analyses were performed for five separate study tasks, viz, (1) analyses and comparisons of swingby, opposition, and conjunction class missions, (2) a detailed parametric analysis of the conjunction class mission, (3) an investigation of the effects of providing launch windows at Earth and Mars for various missions, (4) an evaluation of the vehicle's abort capabilities for various missions, and (5) an analysis to determine the effects of Earth launch azimuth constraints for various missions and launch opportunities. A summary is presented of the results obtained for each of these study tasks with an evalution of and recommendations based on the results.

CONTENTS

ILLUSTRATIONS

TABLES

I. INTRODUCTION

This final report summarizes the mission, trajectory, and vehicle analyses conducted during Phase IV 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 summaries of the basic 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.

STUDY OBJECTIVES AND TASKS

The basic objectives of this study were to expand the mission evaluations performed in the earlier study phases to include trade-offs, mission mode comparisons, and sensitivity investigations of the Venus swingby mode for manned Mars stopover missions; to perform vehicle and engine sizing computations for evaluating launch and abort operations and constraints; and to revise and modify existing computer programs to incorporate additional mission concepts and parameters that would render the programs more effective. To this end, five separate analysis tasks were established.

The five study tasks were 1) Swingby Mission Analysis, 2) Conjunction Class Mission Analysis, 3) Launch Window Analysis, 4) Mission Abort Analysis, and 5) Launch Azimuth Constraint Analysis. A detailed description of each task is included at the beginning of each task section in this report. (Due to a discrepancy in a computer program utilized in Task 4, the Launch Window Analysis, many of the final results obtained for this task were invalid. The computer program has been corrected and the launch window analysis is in the process of being revised. The results of the revised analysis will be presented in a supplemental report at a later date.)

REVIEW OF PREVIOUS STUDY PHASES

Phase IV 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 parametric data and analysis techniques developed during Phase III. Therefore, a brief review of Phases I, II, and III is given here in order to present the study continuity and background applicable to Phase IV.

The first major task of the previous phases was to develop the SWingby Optimization Program (SWOP), 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. The development of this computer program required detailed analyses of interplanetary trajectories, nuclear engines, and the spacecraft in order to determine the required scaling laws, data, and correlations which would relate all of the pertinent variables.

The SWOP program was then utilized to analyze opposition class and flyby missions for various trajectory types, launch opportunities, vehicle configurations, and performance parameters. The results of this analysis permitted the determination of 1) the best compromise engine thrust level for these missions in the 1975 to 1990 time period, 2) the vehicle and stage weight sensitivity to variations in performance, vehicle, and mission parameters, and 3) the mission, vehicle, and engine requirements for these future manned interplanetary missions.

Finally, all of the parametric data that were generated in the course of this study were compiled in an extensive parametric data book, to support future analyses of interplanetary missions, vehicles, and propulsion systems. (The detailed technical report of the mission and vehicle analysis portion of Phases I, II, and III is given in Reference 1. Reference 2 is the parametric data book mentioned above.)

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.

The SWOP program uses a unique employment of analytic and mathematical techniques, specified curve fit routines, and precomputational 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 perihelion distance (solar flare shielding), 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 fixed velocity parameters. These include the launch or arrival dates at Earth or the target planet; the individual leg or total trip times; the velocity increments and the perihelion distance; the periapsis distance; 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 flexible constraint option was very useful for the launch window and vehicle sizing analyses.

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. 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. These percentage values ranged from 0 to 2.3 percent for the various mission propulsive phases.

The impulsive velocities used by the program 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; for the braking maneuver at Mars, the vehicle is decelerated into a 600 km circular orbit.

Running time for the SWOP program is typically two seconds per case.

REPORT ORGANIZATION

Each of the following four sections of the report summarizes the analyses and results for one of the study tasks. Each section essentially is complete within itself although for those tasks in which specific task data or guidelines are identical to those of a previously described task, repetition has been avoided by referencing back to the section where the data was first presented. A final section presents a summary of only the more salient results for each task.

II. SWINGBY MISSION ANALYSIS

TASK DESCRIPTION

The SWingby Optimization Program (SWOP) was utilized to determine the initial weight requirements in Earth orbit for manned Mars stopover missions employing the Venus swingby mission profile. The necessary free flight trajectory data for the gravity turn swingby legs were supplied to TRW Systems by NASA. The mission analyses included both gravity turns and powered turns at Venus for mission opportunities from 1980 to 1986. Both outbound and inbound Venus swingby trajectories were analyzed together with both long and short trajectories for the direct leg of the round trip mission.

These investigations included variations in the vehicle propulsive and deceleration systems both at Mars and at Earth, in nuclear engine performance parameters, and in vehicle structural scaling laws.

Opposition class round trip missions to Mars were reanalyzed for those vehicle weight and performance parameters which were not investigated in Phase III. The results of these mission evaluations were incorporated with existing data from Phase III, the swingby mission results, and the conjunction class mission results (Section III) to illustrate the effect on initial vehicle weight of the variations in launch opportunities, mission and trajectory types, performance parameters, and vehicle systems and scaling laws.

ASSUMPTIONS AND CONSTRAINTS

A set of assumptions and constraints were postulated for this task in order to circumscribe the mission types and modes, the vehicle system weights and performance parameters, the mission and vehicle operational criteria, and the scope of analysis.

Missions

The basic set of missions analyzed and compared in this task consisted of various types of the manned Mars stopover mission. These variations included the Venus swingby mode with both gravity and powered turns at Venus, the opposition class mission, and the conjunction class mission.

Mission Description. A typical opposition class stopover mission is characterized by four major acceleration and deceleration phases that occur during the mission. These are the leave Earth injection phase, braking at Mars, the leave Mars injection phase, and retro braking at Earth with a subsequent aerodynamic entry and surface landing. While in circular orbit at Mars, the spacecraft jettisons a Mars surface lander and at the end of the Mars stopover period recovers an ascent module. Additional vehicle weight requirements are considered for life support expendables, propellant boiloff, and attitude control. If an aerodynamic braking mode is employed at the target planet (Mars), a propulsive velocity change is used for circularizing or adjusting the resulting orbit. The Earth braking propulsive retro can be eliminated by option and an all aerodynamic Earth braking mode employed.

A swingby mission is essentially the same as the opposition class Mars stopover mission except the trajectory is constrained to pass in the vicinity of the planet Venus either during the outbound or inbound leg. The vehicle, therefore, performs a hyperbolic turn about Venus, the degree of turn governed by the choice of the periapsis radius. For the swingby mission, a third mid-course correction propulsion maneuver is assumed.

For the Venus powered turn swingby, a desired departure hyperbola is attained by initiating a propulsive impulse in the vicinity of Venus. A discrete approach periapsis radius exists that minimizes the propulsive impulse required to attain the desired outgoing asymptote. If this periapsis radius is less than the specified minimum, the approach distance is constrained to the minimum and the corresponding (nonminimum) propulsive impulse is computed.

The conjunction class stopover mission is essentially similar to the opposition class mission except a stopover time at Mars is selected so that the return trip to Earth occurs during the next Earth-Mars opposition following the opposition that occurs during the outbound leg. The spacecraft, therefore, dwells at Mars during the Earth-Mars conjunction which occurs between the two oppositions.

Trajectory Types - Two types of trajectories were considered for the direct leg of the swingby missions, types I or B and types II or A. 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° and less than 360°; the II or A designates a trajectory leg where $0^{\circ} < \theta < 180^{\circ}$.

Three types of trajectories were considered for the swingby leg of a swingby mission, types 1, 3, and 5. A detailed discussion of swingby trajectory characteristics is presented later in this section and in Ref 3.

Only the IIB round trip trajectory was considered for the opposition class mission comparisons. It was previously shown in Phase III (Ref 1) that the IIB trajectory generally produces the minimum initial vehicle weight for all opportunities.

A IA conjunction class mission trajectory was selected for comparing the conjunction class mission with the opposition and swingby class missions in this task. The IA conjunction class trajectory yields a lower weight vehicle than the other three possible trajectories (types IB, IIA, and IIB). (See Section III.)

Mission Matrix. Table II-1 presents the matrix of opportunity years and mission and trajectory types analyzed for this task in order to provide comparisons among opposition, swingby, and conjunction class missions.

Table II-1 Comparative Mission Matrix

Mission Type	Year	Trajectory Type Outbound	Trajectory Type Inbound
Swingby (Gravity Turn)	1980	I and II	1
		3	A and B
	1982	I and II	3
		1	A and B
	1984	I and II	5
		5	A and B
	1986	I and II	1
		3	A and B

Mission Type	Year	Trajectory Type Outbound	Inbound
Swingby (Powered Turn)	1980	I	1
	1982	I	3
Opposition Class	1980, 1982, 1984 & 1986	II	B
Conjunction Class	1983	I	A

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.

Propulsion System Weight Scaling Laws - Two inherently different types of vehicle configurations were used for this study task, a tanking mode and a connecting mode.

The tanking mode tends to make full use of the Earth launch vehicle payload volume capacity by orbiting empty or partially filled modules. The modules are then filled via propellant transfer from tanker vehicles or from an orbital propellant storage facility. The maximum capacity of each module or tank was set by the limitations of the Earth launch vehicle.

In the connecting mode, the modules are orbited fully loaded with propellant, hence, their propellant capacity is limited by the Earth launch vehicle payload weight capacity. The use of the connecting mode gives rise to specific vehicle configurations as tanks are added to each stage as the propellant requirements increase. For the leave Earth stage, a cluster of three propulsion modules is first assumed, 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 2A is still insufficient, two additional propellant modules are clustered to the single propellant module. The resultant three 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.

The configurations for the arrive Mars and leave Mars stages are similar to the leave Earth stage except single propulsion modules or propellant modules are used at each level or tier. A schematic depiction of the connecting mode configuration is shown in Figure II-1.

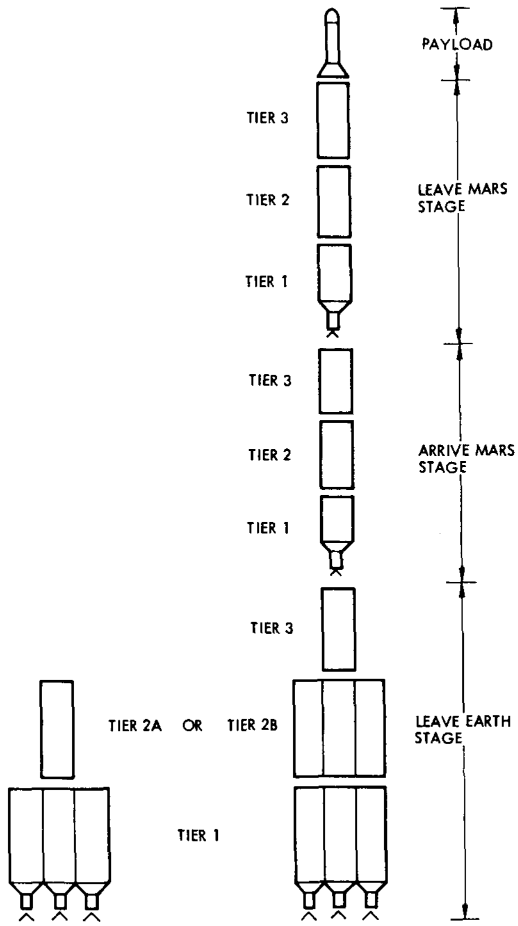

Figure II-1. Connecting Mode Vehicle Configuration

An additional fixed weight is assigned to each stage of the connecting mode configuration to account for the interstage structure and additional docking and assembly weight requirements.

Each of the two modes, the tanking mode and the connecting mode, has its own set of structural scaling laws. Two sets of scaling laws were investigated for the tanking mode. These sets of scaling laws were taken from Phase III and are designated mass fraction case No. 2 and No. 3 (Ref 1). Vehicle configurations employing either chemical or nuclear propulsion systems for main stages were analyzed for mass fraction case No. 2. Mass fraction case No. 3 was used only for the vehicle configurations employing chemical main stage propulsion systems.

The scaling laws for the connecting mode configuration were based on preliminary results from the LMSC Modular Nuclear Vehicle Study, Phase II. Subsequent LMSC design analyses have shown that these mass fraction values can be considerably improved. The connecting mode configuration was used only for vehicle configurations employing nuclear propulsion stages. Table II-2 gives the average mass fraction obtained from the scaling laws used for the main propulsion stages. Additional scaling laws were used to define the weights of the secondary propulsion systems.

Table II-2. Structural Scaling Laws

Mission Phase	Average Mass Fraction		
	MF No. 2	MF No. 3	Connecting Mode
Earth Depart			
*Nuclear	.84	.80**	.78
Cryogenic	.90	.86	
Planet Braking			
*Nuclear	.83	.79**	.78
Cryogenic	.87	.82	
Planet Depart			
*Nuclear	.83	.79**	.78
Cryogenic	.87	.82	
Storable	.91	.87	.87

*Engine Weight not Included

**Mass fraction case No. 3 for nuclear engines used only for Conjunction Class Mission Analysis (See Section III).

For the conjunction class mission, an additional weight was added to the planet depart stage to account for the increased micrometeoroid protection required due to the longer planet stopover period. This weight was added to the tank weight for all sets of scaling laws, i.e., mass fraction case No. 2 and No. 3 and connecting mode.

A single nuclear engine weight of 38,000 pounds was used and each engine was assumed to have 230,000 pounds thrust. The engine weight and thrust for clusters of two or more nuclear engines were taken as direct multiples of these values.

<u>Payload and Expendable Weights</u> - The payloads and expendable weights assigned to the various missions were selected jointly by MSFC and TRW. They represent reasonable values obtained from the many interplanetary mission studies performed by NASA, TRW, and industry in the past years.

The Earth recovered payload is the module weight including the crew that lands on the Earth's surface after aerodynamic braking has been accomplished. 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 retro-braking at Earth or aerodynamic braking if a retro is not employed.

The solar flare shield is not included in the mission module weight. The shield weight is computed as a function of the assumed solar activity and perihelion distance and is added to the mission module weight to determine the total weight to be jettisoned prior to Earth arrival. The yearly variation in solar activity was accounted for by developing three solar flare shield weight scaling laws, for a quiet, intermediate, and active sun. These laws were developed in Phase III (Ref 1). The variations in perihelion distances for the various missions and opportunities together with the applicable scaling laws produced solar flare shield weights that varied from approximately 16,000 to 25,000 lbs.

The Mars excursion module for the stopover mission is jettisoned from the spacecraft at the target planet, out of the circular parking orbit. It contains the planet surface landing system, experiment equipment, and the 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 the planetary orbit.

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

A list of the payload and expendable weight data used in this task are given in Table II-3. The weights for the conjunction class mission are approximately 50 percent greater than the weights for the opposition and swingby class missions to account for an increased crew size and crew and system requirements dictated by the long stay time at Mars.

Table II-3. Payload and Expendable Weights

Payload	Mission Mode: Opposition and Swingby	Mission Mode: Conjunction
Earth Recovered Module	10,000 lb	15,000 lb
Mission Module (not including Solar Flare Shield)	68,734 lb	100,000 lb
Mars Excursion Module	80,000 lb	135,000 lb
Orbit Return Weight	1,500 lb	3,100 lb
Life Support Expendables	50 lb/day	75 lb/day

Aerodynamic Braking Scaling Laws - The weight of the aerodynamic heat shield was expressed as a function of the entry velocity for the operational modes employing aerodynamic braking for the Earth entry module and for arriving at Mars. The analysis and derivation of the scaling laws were accomplished during Phase III and are described in Ref 1.

The scaling laws for aerodynamically braking the Earth recovered module for the two module weights used in this task produced gross vehicle weights before Earth braking that ranged from 13,000 to 20,000 lbs over the range of Earth arrival velocities encountered.

The weight scaling law for aerodynamic braking at Mars resulted in a heat shield weight including all jettisonable ablative material, structure, and insulation that averaged about 25 percent of the gross vehicle weight arriving at Mars.

Secondary Spacecraft Systems - Additional weight expenditures were allowed for secondary spacecraft systems including midcourse corrections, attitude control, and orbit adjustment for modes employing aerodynamic braking at Mars.

It was assumed that the midcourse corrections were performed with a 330 sec, liquid storable propellant system. Separate jettisonable stages were used for the outbound and inbound leg velocity corrections and for a third leg correction for swingby missions. A midcourse correction of 100 m/sec was used for each leg of the mission.

One percent of the vehicle weight was used for attitude control during each leg of the mission including the third swingby leg. The attitude control provisions during the planetary stopover period were computed on the basis of 0.2 percent of the vehicle weight in planetary orbit.

A separate propulsion system was used for all modes employing aerodynamic braking at Mars for circularizing and adjusting the orbit after braking at Mars. This jettisonable, liquid storable, propulsion stage has a specific impulse of 330 sec and is sized for a characteristic velocity of 130 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 cryogenic propellant storage analysis determines the optimum trade-off between the thickness or weight of insulation and the weight of vaporized propellant such that a minimum weight vehicle results. The derivation of the necessary equations used in this analysis 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 cryogenic propellant storage analysis for the connecting mode determines the weight of propellant vaporized during the various phases of the mission based on specified rates of propellant boiloff, i.e., a fixed insulation thickness is preselected to form the best compromise for all of the mission phases during which the propellant is vaporized. The weight of this insulation per tank is, therefore, a fixed quantity and is included in the structural laws.

Since the propellant tanks for this mode are not filled or topped off in Earth orbit, the quantity of propellant vaporized from the tanks during assembly and checkout prior to injection into the interplanetary orbit must be considered, and the scaling laws and tank capacities adjusted to account for the additional tankage required and the reduction in available tank capacity. The weights of propellant vaporized in Earth orbit were approximately 24,000 lbs for tier 1 of the leave Earth stage; 4,700 lbs for tier 2A of the leave Earth stage; 10,000 lbs for tier 2B of the leave Earth stage; and 2,500 lbs for tier 3 of the leave Earth stage and for each of the tiers of the arrive and leave Mars stages.

The propellant boiloff rates used for computing the propellant vaporized during the outbound leg are 24 lbs per day per tank plus 2.388×10^{-3} lbs per day per ft^2 of tank surface area; during the stopover period at Mars the rates are 58.32 lbs per day per tank plus 4.674×10^{-3} lbs per day per ft^2 of tank surface area.

<u>Vehicle Mode Matrix</u>

Each of the mission cases represented in the Comparative Mission Matrix, Table II-1, pg II-3 was analyzed for a variety of vehicle modes, propulsion types, engine performance parameters, and stage scaling laws. Table II-4 shows the combination of sets of scaling laws and propulsion and aerodynamic braking systems analyzed for each of the opposition, conjunction, and gravity turn swingby class missions of the Comparative Mission Matrix.

Table II-4 Vehicle Mode Matrix

	Earth Depart, Mars Arrival, Mars Depart			
Earth Arrival	NNN	NAS	CCC	CAS
A	MF #2 Connecting Mode	MF #2 Connecting Mode	MF #2 MF #3	MF #2 MF #3
S(P)	MF #2 Connecting Mode		MF #2 MF #3	

N - Nuclear Propulsion (800 and 850 sec)
C - Chemical Cryogenic Propulsion, H_2/O_2 (440 sec)
S - Liquid Storable Propulsion (330 sec)
A - Aerodynamic braking
S(P) - Liquid storable propulsion (330 sec) to parabolic entry velocity followed by aerodynamic braking

The use of one, two, three, and four clustered nuclear engines was investigated for the depart Earth stage for mass fraction case No. 2 in order to determine the optimum engine configuration, i.e., the number of engines which produces the optimum thrust-to-weight ratio for the depart Earth stage.

The powered turn swingby mode was analyzed primarily to obtain a comparison with the gravity turn swingby. For this purpose only the NNNA connecting mode configuration was investigated. A specific impulse of 850 sec was used for the nuclear engines and a chemical cryogenic propulsion system with a specific inpulse of 440 sec was assumed for providing the propulsive kick during the Venus swingby.

GRAVITY TURN SWINGBY MISSIONS

The major objectives for this task were 1) to determine the initial vehicle weight requirements for Mars stopover missions employing gravity turn swingbys at Venus for the years 1980 through 1986, and 2) to compare these results with the analogous results for opposition and conjunction class missions for the same time period.

Generation and Processing of Trajectory Data

The generation of the gravity turn swingby trajectory data was accomplished at the NASA/Office of Manned Space Flight (Washington) and transmitted to TRW.

Possible Venus swingby missions were found by matching one-way, Mars-Venus and Venus-Earth trajectories at Venus. In order for Venus to perturb the vehicle sufficiently to alter its heliocentric trajectory, it is necessary for the vehicle to pass well within the sphere of influence of Venus. While the vehicle is within the sphere of influence of Venus, it is assumed to be on a free-flight conic section (hyperbolic) trajectory about Venus and there is no change of energy of the vehicle with respect to Venus. Therefore, conservation of energy requires that the magnitude of the vehicle's velocity at infinity (V_∞) leaving Venus must equal its arrival velocity at infinity. It is possible, then, to match Mars-Venus trajectories and Venus-Earth trajectories at Venus (for a given date at Venus) to form the Mars-Venus-Earth gravity turn swingby trajectories simply by matching the arrival and departure V_∞ magnitudes. The magnitudes and directions of the arrival V_∞ and the departure V_∞ together define a unique hyperbolic trajectory about Venus, i.e., the planet passage distance (periapsis) is uniquely determined.

Figure II-2 illustrates the V_∞ matching procedure for a given Venus encounter date of an inbound swingby. Inbound gravity turn swingbys are possible for a given Venus encounter date wherever the arrival and departure V_∞'s match. For a given Venus encounter date there are actually sixteen possible matching combinations yielding sixteen combinations of first leg time and second leg time. If Venus is assumed to be a dimensionless point mass and if we ignore what happens to the velocities leaving Mars and arriving at Earth, all of these sixteen combinations are theoretically possible. However, practical restrictions eliminate some of these possible matching combinations. Since Venus is an approximate sphere of finite size with an atmosphere, it is necessary to eliminate any combinations yielding a planet passage distance less than some minimum value (1.05 Venus radii was assumed to allow some margin above the atmosphere). Also combinations yielding impractically high speeds leaving Mars or arriving at Earth and those which do not permit a feasible outbound trip can be eliminated. After these practical restrictions are applied, typically two to four combinations remain.

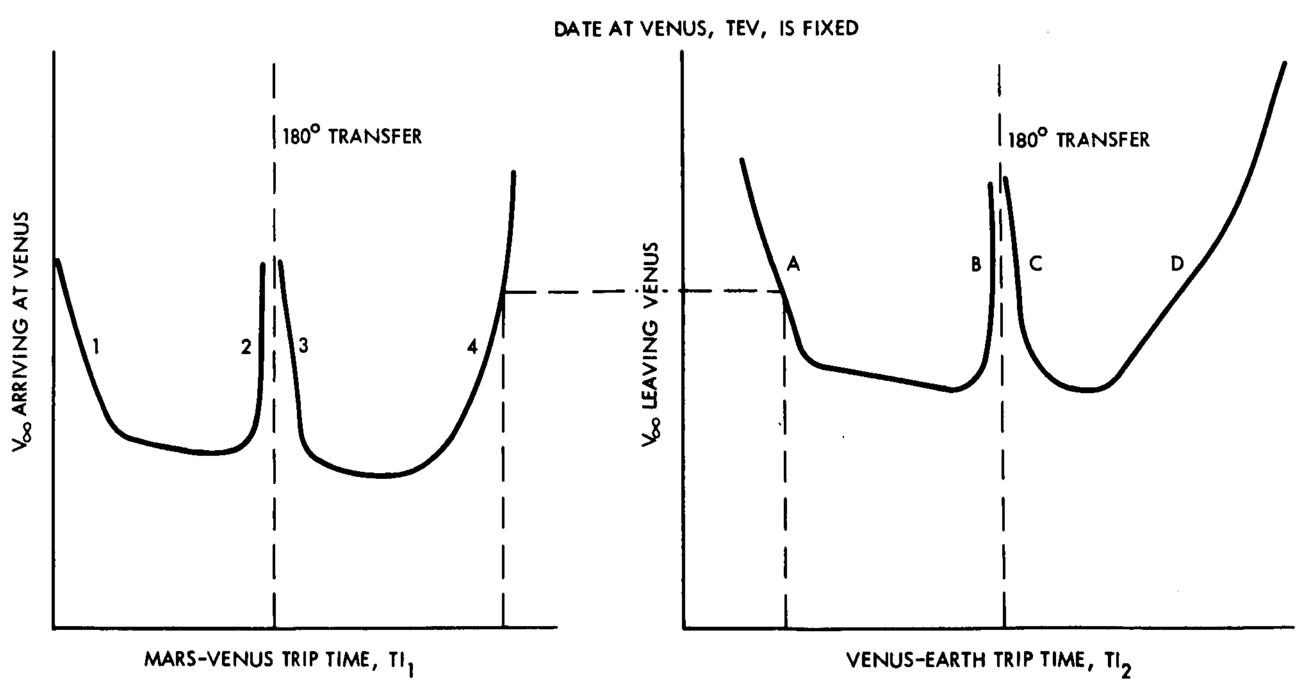

Figure II-2. Typical Possible Gravity Turn Swingby Trajectories

However, generally all quantities of interest for pairs of these combinations are continuous so each pair of combinations can be taken together as one set of trajectory data.

Since SWOP mates the outbound and inbound trajectories at Mars, it is convenient in SWOP to use the Mars date as one of the independent variables defining the round trip trajectory. It was necessary, therefore, to convert the NASA-furnished data to data having the Mars departure date as the independent parameter. This required a subtraction and a reordering of the data to obtain sets of constant Mars departure date. In additon, the first leg time is unsatisfactory as an independent parameter since for each combination of Mars departure date and first leg time, the second leg time (as well as all of the other data) is double valued. Therefore, an interpolation was required to obtain fixed increments of the second leg time so that it could be used as the second independent parameter.

The procedure for preparing the outbound unpowered swingby data for use in the SWOP program was the same as outlined above with one additional step. Since the data was generated with fixed increments of Venus encounter date and Earth departure date and since SWOP requires fixed increments of Mars arrival date, an additional interpolation was required to obtain fixed increments of Mars arrival date.

Figures II-3 and II-4 illustrate typical sets of trajectory data as they have been finally prepared for mission analysis using SWOP. In the nomenclature of Ross and Gillespie (Reference 3), these represent a type 1 swingby and a type 3 swingby, respectively. They differ primarily with respect to the effect of the planet passage distance constraint. In the type 1 swingby (Figure II-3), the minimum planet passage distance is encountered before the minimum ΔV's at Mars and Earth are reached; whereas in the type 3 swingby (Figure II-4), practical swingby trajectories containing the lowest ΔV's at Mars and Earth are feasible because they lie outside the region restricted due to the planet passage distance constraint. A third practical swingby type, (type 5) also occurs. It differs appreciably from the types 1 and 3 since it results from a much different alignment of the three planets than that yielding the types 1 and 3. A more comprehensive discussion of the swingby types, the differences between them, and their merits is available in Reference 3.

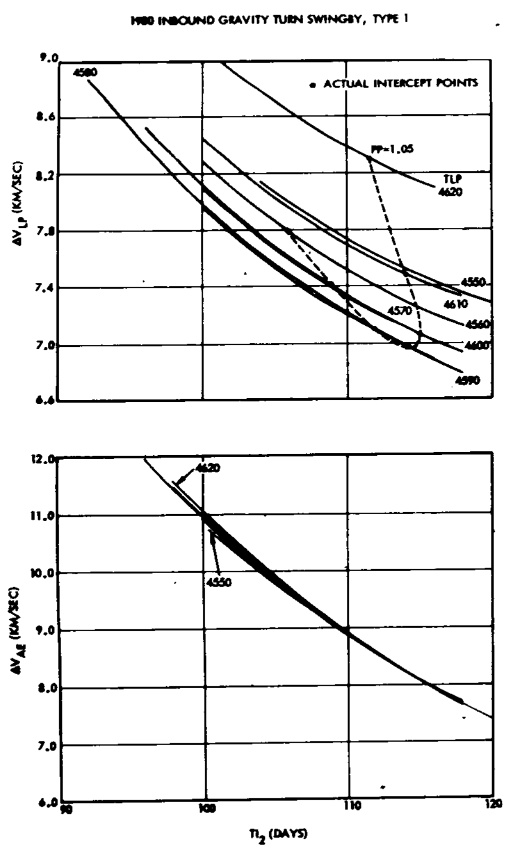

Figure II-3, Velocity Data for 1980 Inbound Gravity Turn Swingby Trajectory

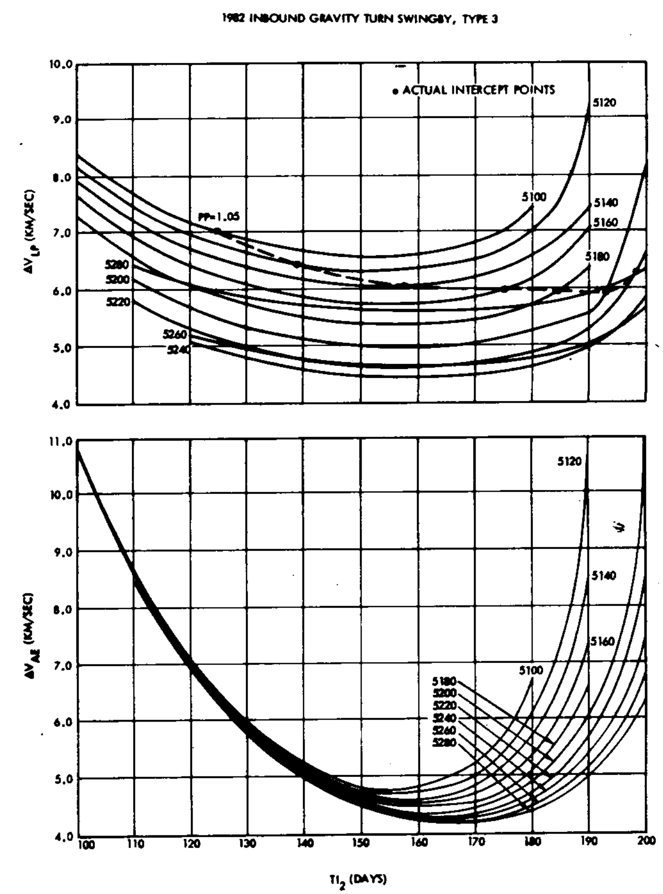

Figure II-4. Velocity Data for 1982 Inbound Gravity Turn Swingby Trajectory

Results and Discussion

Gravity Turn Swingby Evaluation - The gravity turn swingby missions are evaluated on the basis of several levels of trajectory and vehicle characteristics. First, the different types of swingby trajectories are compared; second, the use of the short or long direct leg with a swingby leg is evaluated; third, a comparison of the results for the various mission years is made; and finally, the results for the different vehicle modes are discussed.

Figures II-5 and II-6 for the NNNA and NASA modes indicate that the type 3 gravity turn swingby trajectories lead to lower weight vehicles than the type 1 trajectories for the years 1980, 1982, and 1986. The superiority of the type 3's over the type 1's is consistent regardless of the direct leg type or vehicle mode. In 1984, the inbound type 5 yields a lower weight vehicle than the outbound type 5 for the vehicle modes employing propulsive braking at Mars (Figure II-5). For vehicle modes employing aerodynamic braking at Mars, the outbound type 5 swingby trajectories with a long (type B) direct inbound leg show a slightly lower vehicle weight.

The type 3 trajectories for the years 1980 and 1982 yield a lower weight vehicle by employing a long trajectory for the direct leg. That is, a 1980 outbound type 3 swingby leg with a type B inbound direct leg (3-B) and a 1982 inbound type 3 swingby leg with a type I outbound direct leg (I-3) are the best swingby trajectories for those two years. The use of the long instead of the short direct leg tends to increase the total trip time by 128 and 50 days for 1980 and 1982, respectively, but reduces the gross vehicle weight by approximately 15 percent.

The long direct leg also yields the minimum weight vehicles when coupled with the type 5 trajectories for 1984. For those vehicles employing propulsive braking at Mars, the longer inbound swingby or type I-5 round trip trajectory results in a vehicle weight approximately five percent less than that given by the shorter II-5 trajectory. As previously shown for vehicles utilizing aerodynamic braking in 1984, the longer outbound swingby mission, type 5-B is best; the vehicle weight is about five percent lower than for the next best swingby (II-5).

In 1986 the shorter type 3 swingby mission (3-A) yields slightly lower weight vehicles than the longer type 3-B. This lower weight and the shorter trip time (49 days less), therefore, make the type 3-A trajectory the preferred swingby mission for this year.

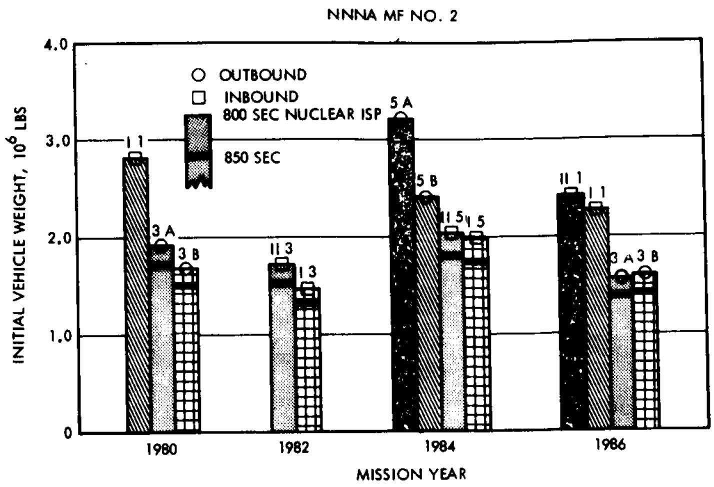

Figure II-5. Swingby Trajectory Type Comparison, NNNA Mode

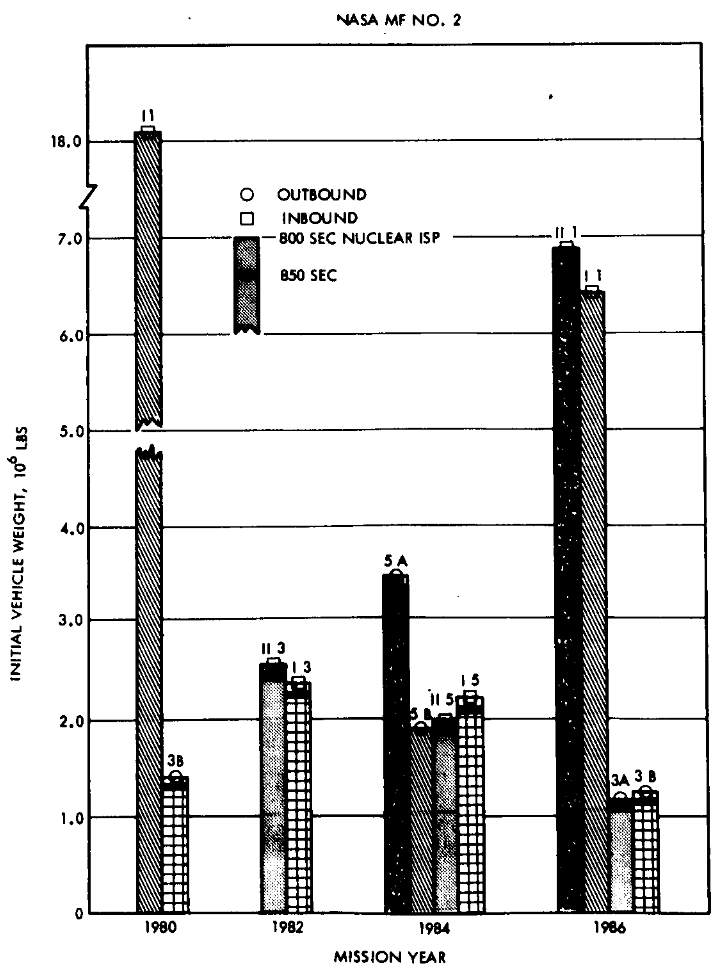

Figure II-6. Swingby Trajectory Type Comparison, NASA Mode

The results for the NNNS(P), CCCA, CCCS(P), and CASA vehicle modes are qualitatively similar to the analogous NNNA and NASA modes.

A typical comparison of swingby missions on the basis of the mission year or opportunity is shown in Figure II-7 for the NNNA, NASA, CCCA, and CASA modes. Plotted on this graph are the vehicle weights for the best (minimum weight) trajectories as previously discussed. The vehicle weights for the NNNA mode increase in the following order; 1982 (minimum), 1986, 1980, and 1984 (maximum). For the NASA mode, the weight is a minimum in 1986 and increases in 1980, 1984, and 1982 (maximum). The results for the other analogous nuclear and chemical propulsive modes have vehicle weights that vary through the years in the same order except that for the CCCA and CCCS(P) modes, 1986 is the minimum weight year followed by 1982 which has only a slightly greater weight.

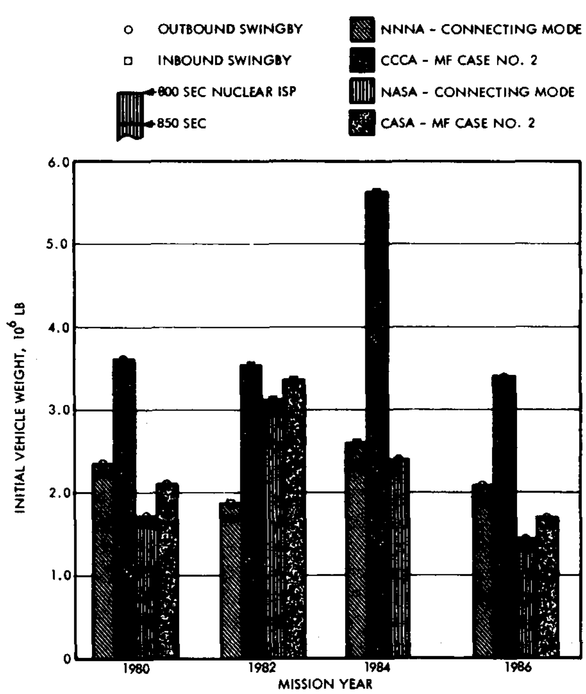

Figure II-7. Launch Opportunity Comparison for Gravity Turn Swingby Missions

The weight differences on the basis of the yearly variations are quite significant. For example, the vehicle weight for the maximum year is from 30 to 40 percent greater than the minimum year for the NNNA and NNNS(P) modes; and from 65 to 80 percent for the CCCA and CCCS(P) modes. The increase from minimum to maximum for the modes utilizing aerodynamic braking at Mars is 88 to 120 percent.

Although the characteristics of swingby trajectories repeat on a synodic cycle of approximately 6.4 years, it does not necessarily follow that favorable (or unfavorable) round trip missions will occur at like intervals because the direct leg is periodic over a fifteen-year cycle. Therefore, any conclusions reached by comparing the results for the years 1980 through 1986 may not necessarily apply for preceding or succeeding six-year periods.

The results on Figure II-7 show that the modes employing chemical cryogenic propulsion stages require vehicles whose weights range from 15 to 110 percent greater than the nuclear vehicles. In addition, a decrease in the nuclear engine specific impulse from 850 to 800 sec increases the vehicle weight from 5 to 20 percent.

The arrival velocities at Earth for the best swingby trajectories in any year are generally only slightly greater than parabolic velocity. Therefore, there is only a slight difference in weight between the modes employing all aerodynamic braking at Earth and those employing a retrostage to decelerate the vehicle to parabolic entry velocity.

Mission Modes Comparison - A comparison of the three mission modes or types is presented by first comparing the conjunction and opposition class missions, then the conjunction and swingby class missions, and finally the opposition and swingby class missions. Figure II-8 presents a comparison of the initial vehicle weight for the NNNA mode for all three classes of missions, i.e., the swingby, opposition, and conjunction class missions. It should be remembered that the payloads for the conjunction class mission have been increased by 50 percent over the other two types of missions to account for the approximately double total trip time and extremely long Mars dwell time.

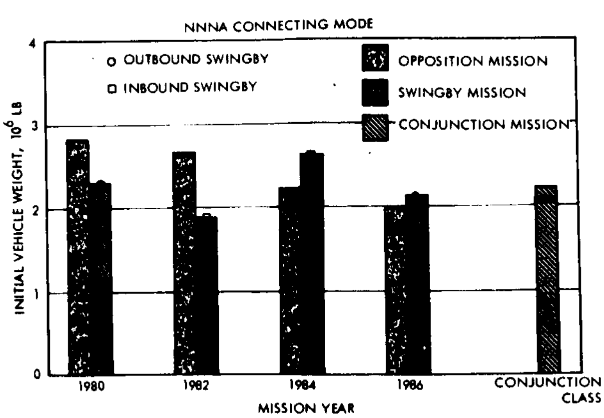

Figure II-8. Mission Mode Comparison, NNNA Vehicle Configurations

A comparison of the conjunction and opposition class missions shows that the vehicle weight for the conjunction class mission is essentially equal to that for the opposition class mission in 1984 and an average of 16 percent greater than the 1986 opposition vehicle; but the 1980 and 1982 opposition class missions have greater vehicle weights than the conjunction class mission; 25 percent greater in 1980 and 12 percent greater in 1982.

A comparison of the conjunction and swingby class missions shows that the swingby mission yields a lower weight vehicle for the years 1982 and 1986; 17 percent less in 1982 and approximately 5 percent less in 1986. In 1980, the swingby mission vehicle weight is essentially equal to the conjunction class weight and in 1984, approximately 16 percent greater.

A comparison of the opposition and swingby class missions shows that the swingbys give lower vehicle weights in 1980 and 1982, 20 percent less in 1980 and 30 percent less in 1982. In 1984 and 1986, the swingby missions yield greater vehicle weights than the opposition class missions; 18 percent greater in 1984 and approximately 10 percent greater in 1986.

Similar comparisons of mission classes for the CCCA vehicle mode revealed that the conjunction class mission yields a vehicle weight lower than either the opposition or swingby class mission for all years. In the comparison of the opposition or swingby class missions, the swingby mission has a lower weight vehicle in all years except 1984.

An extension of the comparisons to vehicles employing aerodynamic braking at Mars is shown in Figure II-9 for the NASA mode. A comparison of the conjunction and opposition class missions for this mode shows that the vehicle weights for all years of the opposition class missions are greater than for the conjunction class mission. The opposition class vehicle weights are 112 percent greater in 1980, 70 percent in 1982, 48 percent in 1984, and 40 percent in 1986.

A comparison of the conjunction and swingby class missions shows that the swingby missions of 1980 and 1986 yield lower weight vehicles; 9 percent less in 1980 and 23 percent less in 1986. In 1982 and 1984, the swingby mission vehicles are greater in weight; 65 percent greater in 1982 and 48 percent in 1984.

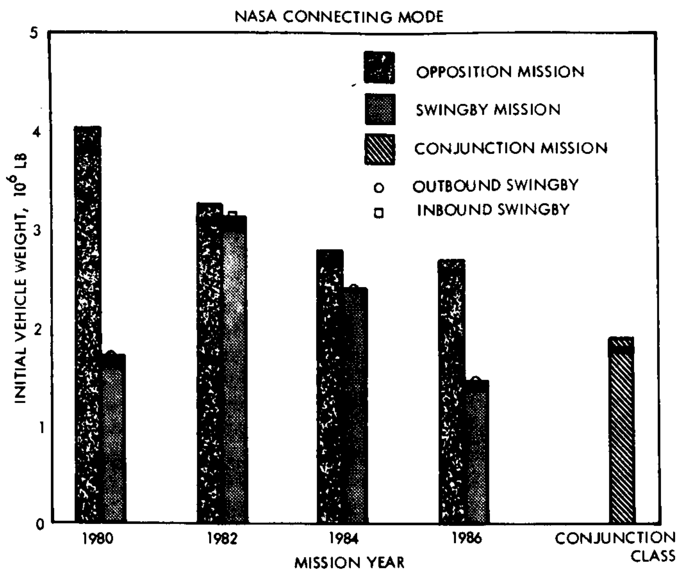

Figure II-9. Mission Mode Comparison, NASA Vehicle Configuration

A comparison of the opposition and swingby class missions shows that the swingby missions yield lower vehicle weights in all years. The vehicle weights for the swingby class missions are 57 percent less in 1980, 3 percent in 1982, 14 percent in 1984, and 45 percent in 1986.

Similar comparisons of mission modes for the CASA vehicle mode revealed that qualitatively the results are identical to those of the NASA mode.

POWERED TURN SWINGBY MISSIONS

Analysis Approach

A number of investigators have found that using a powered turn at Mars can provide significant reductions in the total velocity requirements for the nonstop Mars flyby mission. Hollister (Ref 4) suggested that a powered turn might also offer an improvement to the unpowered or gravity turn Venus swingby missions. Hollister found that in some cases the best powered swingbys did require a few hundred feet per second less total ΔV than did the best unpowered swingbys. However, a comparison on the basis of total ΔV is not sufficient to determine whether the addition of the powered turn at Venus (which requires an additional vehicle stage) will improve the mission in terms of reduced vehicle weight. It is necessary to actually size vehicles for the powered swingby mission to determine whether it offers any advantages over the unpowered swingbys and other mission modes.

A comparison was made of powered with unpowered swingbys for the 1980 (type 1) inbound swingby and the 1982 (type 3) inbound swingby.

The graph of the velocity data for the 1980 inbound gravity turn swingby (type 1) was shown previously on Figure II-3, p II-14. Since the use of a gravity turn swingby is constrained to operate above a minimum Venus passage distance, the possible gravity turn trajectories must lie in the region to the left of the PP = 1.05 constraint line shown on Figure II-3. The depart Mars and arrive Earth velocities are relatively high in this region since this region is considerably removed from the region of minimum velocities which occurs off the graph to the right. By adding an additional propulsive capability to the vehicle for performing a powered turn about Venus, the angle turned at the planet can be favorably increased while the vehicle can still operate on or above the minimum passage distance constraint. This is equivalent to moving below the PP = 1.05 radii line on the graph into the region of minimum velocities (ΔV_{LP} and ΔV_{AE}), which would have been impossible for unpowered swingbys.

It would, therefore, appear that the use of a powered swingby mode for the type 1 trajectories might have an advantage over the gravity turn (type 1) mode by permitting lower depart Mars and arrive Earth velocities (or depart Earth, arrive Mars velocities for an outbound type 1 Venus swingby).

The velocity data shown on Figure II-4, p II-14 is typical for the type 3 swingby trajectories. The region of possible gravity turn swingby trajectories for the type 3 swingby trajectories lies below the PP = 1.05 constraint line shown on Figure II-4. Therefore, the region of possible gravity swingbys includes the minima of the ΔV_{LP} and ΔV_{AE} curves. It would, therefore, appear that for 1982 inbound swingbys or type 3 trajectories, no improvement is possible over the optimum gravity swingby by employing a powered turn at Venus.

Results and Discussion

Figure II-10 presents a comparison of the sum of the mission characteristic velocities for both the gravity turn and powered turn swingby missions for the 1980 inbound swingby, a type 1 trajectory. The velocity summations are for the optimum (minimum weight) trajectories for each Mars arrival date, and the curve for the powered swingby includes the additional ΔV required for the powered turn at Venus.

As was inferred from the previous discussion of the type 1 swingby velocity data, the powered swingbys do indeed yield a lower total ΔV than do the unpowered swingbys.

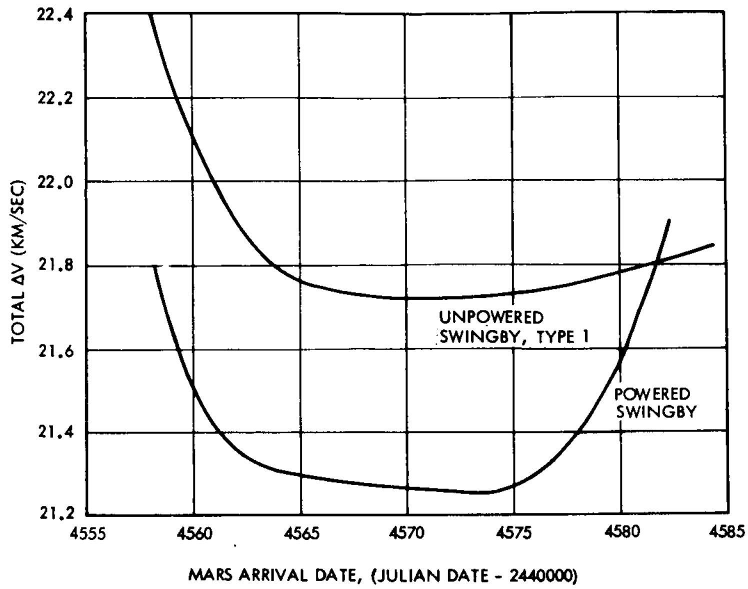

Figure II-10. 1980 Inbound Swingby Velocity Comparison

Notwithstanding the fact that the total ΔV is lower for the type 1 powered turn swingby than for the gravity turn swingby, the vehicle weight for the powered turn swingby mission is approximately five percent greater than for the gravity turn swingby mission (3.18 and 3.02 million pounds, respectively for the NNNA vehicle mode). In order to perform the powered swingby mission, the vehicle must carry an additional stage for the maneuver at Venus. Since this additional fixed weight must be carried through three previous propulsion maneuvers, a lower vehicle weight is not obtained.

A comparison of the sum of the characteristic velocities and vehicle weights for the 1982 inbound (type 3) swingby missions shows that as was expected, the velocity summation for the powered turn swingbys is greater (by approximately four percent) than that for the gravity turn swingby; the vehicle weight is correspondingly greater by approximately five percent (1.78 and 1.70 million pounds, respectively).

The results of the powered swingby mission analysis indicate that the powered swingby mission offers no advantage over the unpowered swingby mission for the type 3 swingby trajectories because there is no reduction in total mission characteristic velocity or initial vehicle weight. (The type 5 swingby missions have trajectory characteristics similar to those of the type 3 missions and would therefore also not obtain any benefit from the use of a powered turn at Venus.)

For the type 1 inbound swingby, use of the powered turn at Venus reduces the total mission characteristic velocity by about 4 percent, but no reduction in vehicle weight results. The major effect obtained from the use of a powered turn with the type 1 inbound trajectory is a reduction in the Earth arrival velocity. Since this maneuver is performed by a low weight aerodynamic braking system that is relatively insensitive to arrival velocity, only a small weight reduction is obtained for this stage. On the other hand, for an outbound type 1 swingby the major velocity reductions obtained from a powered Venus flyby apply to maneuvers performed by propulsion systems. However, for the 1982 outbound type 1 gravity swingby (the only outbound type 1 swingby occurring in the 1980 to 1986 period) the total mission characteristic velocity is approximately twice that for the 1982 inbound (type 3) gravity swingby. Therefore, the use of a powered Venus swingby for this launch opportunity would have to reduce the total velocity of the outbound swingby by 50 percent to make it competitive with the inbound gravity swingby. Since a reduction of this magnitude is not apparently feasible, it is concluded that the best gravity swingby (outbound or inbound with short or long connecting leg) would be superior to the best powered swingby for any given year in the 1980 to 1986 time period.

III. CONJUNCTION CLASS MISSION ANALYSIS

TASK DESCRIPTION

The SWingby Optimization Program (SWOP) was utilized to determine the initial weight requirements in Earth orbit for conjunction class, manned Mars stopover missions. These investigations included parametric variations of the vehicle and propulsive modes, structural scaling laws, and payload weights in order to illustrate their effect on initial vehicle weight.

ASSUMPTIONS AND CONSTRAINTS

The set of assumptions and constraints that was postulated for this task in order to circumscribe the vehicle's performance, operation, and system weights is essentially the same as that of the Swingby Mission Analysis with only a few exceptions or additions. Therefore, the basic set of assumptions, constraints, and definitions are set forth in detail in Section II and only those peculiar to this task are given below.

Mission and Trajectory Description

The conjunction class mission is designed to take advantage of the lowest possible energy requirements for both the outbound trip and the return trip. The opposition class mission is characterized by short stopover periods at Mars and, therefore, cannot take advantage of minimum energy trips. The minimum energy return trip occurs before rather than after the nearest minimum energy outbound trip. Therefore, the opposition class trips are a compromise combination of outbound and inbound energy requirements close to the time of the Earth-Mars opposition.

It is possible to utilize the minimum energy trips, but to do so requires long stopover periods at Mars. These missions using the minimum energy trips are referred to as conjunction class because the Earth-Mars conjunction occurs during the stopover period and the mission is approximately symmetrical about the conjunction. The energy requirements for the conjunction class missions vary much less from year to year than do the requirements for other mission types. Therefore, one year, 1983, was taken as typical and vehicles were analyzed for only that year.

For both the outbound and inbound legs of the round trip trajectory, two types of trajectories were considered, viz, the short and the long one-way transfers. Therefore, four types of round trip missions were investigated; type IA (long outbound leg, short inbound leg), type IB (long, long), type IIA (short, short), and type IIB (short, long).

Vehicle Configuration and 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 the same as those for the Swingby Mission Analysis except for the qualifications and exceptions noted below.

Propulsion System Weight Scaling Laws - Only the tanking mode configuration was employed in the mission analyses conducted for this task. The sets of scaling laws used were the mass fraction cases Nos. 2 and 3. The average mass fractions for these scaling laws were previously given in Section II in Table II-2 on page II-6.

An additional weight was added to the planet depart stage to account for increased micrometeoroid protection requirements due to the longer planet stopover period. This weight is added to the tank weight for all sets of scaling laws, i.e., mass fraction cases Nos. 2 and 3, and is jettisoned prior to Mars departure. This weight varied with the assumed payload and is given in Table III-1 which follows later in this section.

A single nuclear engine weight of 38,000 pounds was used and each engine was assumed to have 230,000 pounds of thrust. The engine weight and thrust for clusters of two or more nuclear engines were taken as direct multiples of these values. A single nuclear engine was assumed for the arrive and depart Mars stages and the optimum number of engines was determined and used for the depart Earth stage, i.e., the number of engines that yield the minimum initial vehicle weight.

Payload and Expendable Weights - The longer stopover time for the conjunction class mission requires that the mission module, life support and Mars payload weights be increased. Four payload sets were postulated as shown in Table III-1. The first set is identical to that used for the missions in the other study tasks which use a 20-day stopover time. The other three sets represent increased weights to account for added experiments and crew. Payload set 3 was used in the comparisons of the conjunction class mission to the other mission types in the other tasks.

Table III-1 Conjunction Mission Payloads

Payload Set	Crew	Earth Return Module (lb)	Mission Module (lb)*	Mars Excursion Module (lb)	Orbit Return Weight(lb)	Life Support Expendables (lb/day)	Additional Micrometroroid Protection (lb)
1	8	10,000	68,734	80,000	1500	50	
2	8	11,500	75,000	109,000	2500	50	$27{,}500 + 27\ T_{SO}$
3	12	15,000	100,000	135,000	3100	75	$38{,}000 + 40\ T_{SO}$
4	20	27,000	150,000	178,000	7500	120	$57{,}000 + 60\ T_{SO}$

*Does not include solar flare shield

The solar flare radiation shield weight used in this task was based on an assumed active year for solar flare activity. A perihelion distance of 1.0 AU was used.

Aerodynamic Braking Scaling Laws - The weight scaling laws for aerodynamically braking the spacecraft at Mars or Earth were identical to those described in Section II and given in detail in Ref 1.

Vehicle Mode Matrix

The conjunction class missions for 1983 were analyzed for a variety of vehicle modes, trajectory types, propulsion systems, engine performance parameters, and stage scaling laws. Table III-2 shows the matrix of cases investigated.

Table III-2 Vehicle Mode Matrix

Year	Trajectory Types	Vehicle Modes	Scaling Laws	Payloads
1983	IA Long, Short	NNNA	MF No. 2	Sets 1, 2, 3 and 4
	IB Long, Long	NNNS(P)	MF No. 3	
	IIA - Short, Short	NASA		
	IIB - Short, Long	CCCA		
		CCCS(P)		
		CASA		

N - Nuclear Propulsion (800 sec)
C - Chemical Cryogenic Propulsion, $H_2/0_2$ (440 sec)
S - Liquid Storable Propulsion (330 sec)
A - Aerodynamic braking
S(P) - Liquid storable propulsion (330 sec) to parabolic entry velocity followed by aerodynamic braking.

RESULTS AND DISCUSSION

Trajectory Type Comparison

A comparison of the four combinations of trajectory types shows that the type IA gives the lowest vehicle weight as shown in Figure III-1 for the NNNA mode, mass fraction case No. 2, and payload set 3. The type IIA trajectory, on the other hand, gives the shortest trip but with the greatest vehicle weight. Therefore, the type IA was selected as the best overall compromise since it requires 11 percent less vehicle weight than type IIA with only 4 percent longer total trip time. The type IA trajectory was used in the comparisons of the conjunction class mission to the other mission types in the other four study tasks.

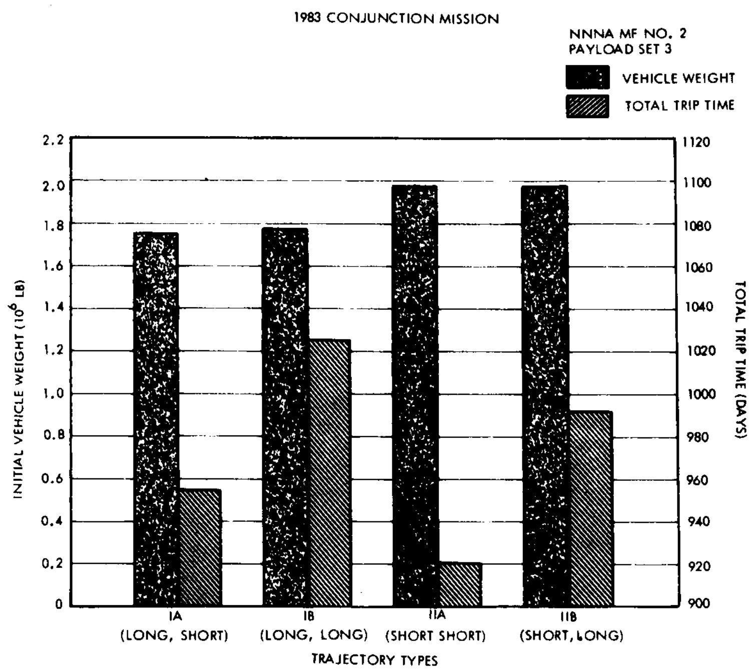

Figure III-1. Conjunction Mission Trajectory Type Comparison

A comparison of trajectory types for the other vehicle modes and payload weights results in identical comparative conclusions.

Vehicle Mode Comparisons

Figure III-2 shows the increased vehicle weight that results from increases in payload weights for the various vehicle modes based on the mass fraction case No. 2 structural scaling laws. Analogous results were obtained for mass fraction case No. 3. The vehicle weights corresponding to payload set 1 (which is the same as that used for all other mission types) are extremely low. A comparison among mission modes in the other study tasks based on this set would show the conjunction class mission requiring considerably lower vehicle weights than the other modes for all years and vehicles. However, payload sets 2, 3, and 4 are probably more realistic for comparisons involving the conjunction class mission.

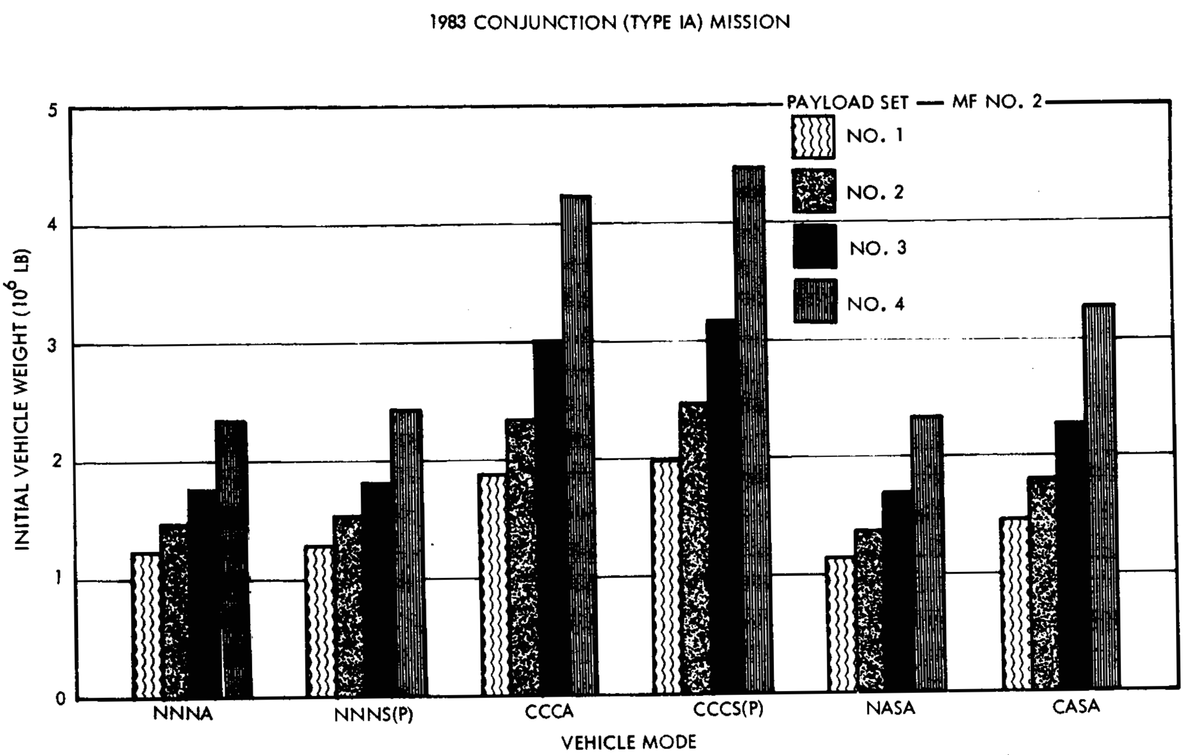

Figure III-2. Conjunction Mission Mass Fraction No. 2 Payload Comparison

IV. MISSION ABORT ANALYSIS

TASK DESCRIPTION

An investigation was made of opposition, swingby, and conjunction class missions to determine the abort capability of the vehicle from various points along the outbound trajectory using the available vehicle propulsive systems. Various combinations of the vehicle propulsive systems were considered for providing the abort velocity increment and the Earth deceleration requirements. Velocity contour maps were constructed indicating the vehicle abort capabilities, Earth entry velocities, and Earth rescue requirements.

ASSUMPTIONS AND CONSTRAINTS

The four missions and vehicles used for the abort analysis are shown in Table IV-1.

Table IV-1. Abort Analysis Vehicles

Mission	Year	Vehicle Mode
Opposition, IIB	1982	NNNS(15)
Conjunction, IA	1983	NNNS(P)
Inbound Swingby, I3	1982	NNNS(P)
Outbound Swingby, 3A	1986	NNNS(P)

The connecting mode vehicle configuration was used for all vehicle weight calculations.

ANALYSIS APPROACH

For convenience of discussion we can divide the manned Mars mission into five general phases: launch, outbound leg, near planet and surface maneuvers, inbound leg, and capture and landing. Abort during launch from Earth will probably be provided by a launch escape system to separate the manned capsule from the launch vehicle and a landing system for the capsule or its occupants.

Once the vehicle has reached the necessary energy to escape the Earth it can be considered to be on its outbound trip, and abort analysis for the beginning of the outbound trip will then be applicable. During the outbound trip, the vehicle is carrying the propulsive capability to be used for the Mars arrival and departure and Earth arrival maneuvers. Therefore, during the outbound trip, the vehicle has propulsive capability which could possibly be adequate for a return to Earth if it was necessary to abort the mission.

The operations near Mars and on its surface consist of a complicated series of maneuvers for which certain failures could terminate the mission, e.g., propulsion failures on the surface of the planet or during escape from the planet. The only possible abort mode for such failures would be redundant propulsion systems which would greatly increase the total vehicle weight to the point of being unreasonable. However, in most cases, it would be possible to return to Earth if the failure occurred in the Mars parking orbit and did not involve the depart Mars propulsion system.

During the return trip to Earth, the only available propulsive capability is the Earth arrival stage. The only possible use which could be made of the Earth arrival stage would be to provide a faster return trip or to provide additional midcourse correction capability. In either case, the vehicle would be left without the necessary means for its arrival maneuver at Earth. Therefore, any abort attempt during the inbound trip which employs the Earth arrival propulsion would be essentially impractical unless a rescue mode was available at Earth.

The capture and landing maneuvers at Earth involve using the last of the propulsive capability of the vehicle. Therefore, without redundant propulsive or aerodynamic braking capability there is very little the crew can do in case of failure during these maneuvers. However, at this point in the mission, it may be feasible to consider rescue of the crew by an Earth-based vehicle.

It is clear from the above discussion that only during the outbound trip will the crew have clear alternatives for aborting the mission if the need arises. It was necessary then to analyze the requirements for abort from the outbound trajectory so that these alternatives could be compared. The approach which was taken was to compute the impulsive abort and Earth arrival velocity requirements for aborting along the nominal outbound trajectory. The results were plotted as contours of constant ΔV on a grid of return trip time versus date of abort. Lines were also included on the contour maps to indicate the regions where the vehicle would not pass through perihelion on the abort trajectory or to indicate how near the vehicle must approach the sun when it does pass through perihelion.

Then six different combinations of the vehicle propulsive systems for abort and arriving at Earth were assumed and the ΔV capability for each of these combinations was computed as a function of mission date. Envelopes showing the region of possible abort for each combination were overlaid on the contour maps. The final result shows when abort will be possible for a given mission and failure mode, and the time required for the return trip. Typical examples of opposition class, conjunction class, and inbound and outbound Venus swingby missions were selected and abort analyses completed for each.

RESULTS AND DISCUSSION

The contour maps with their associated vehicle abort capability overlays for the four missions analyzed are given in Figures IV-1, IV-2, IV-3 and IV-4. With the vehicle abort capability curves overlaid on the contour maps it is immediately apparent when abort is possible and when it is impossible, which of the possible abort trips gives the quickest return to Earth, which will require the least amount of fuel, which will give the lowest arrival velocities at Earth, and which will give the greatest solar passage distance. In some instances, such as when a failure or malfunction is discovered late, it may be impossible or undesirable to follow one of these "optimum" trips. For such instances the map shows all trips that are still possible and a choice can be made.

Figure IV-1 illustrates the abort capabilities for the 1982 opposition class mission. Successful abort is possible during approximately the first half of the outbound leg for all assumed vehicle abort capabilities. (The regions of possible aborts lie to the left, within the areas that are partially enclosed by the individual capability curves.) The abort capability is extended over nearly the entire outbound trip for two of the cases which employ both the arrive Earth retro and aerodynamic braking capability for decelerating at Earth.

The abort curves for the conjunction class mission shown on Figure IV-2 indicate that at best an abort is possible only during the first third of the outbound trajectory for those cases in which both the arrive Earth propulsive retro and aerodynamic braking capability is employed for decelerating the vehicle at Earth. For those cases in which the arrive Earth propulsive retro is known to be inoperable or has been utilized for the abort ΔV, no successful abort is possible since the vehicle is left without the necessary means for performing its arrival maneuver at

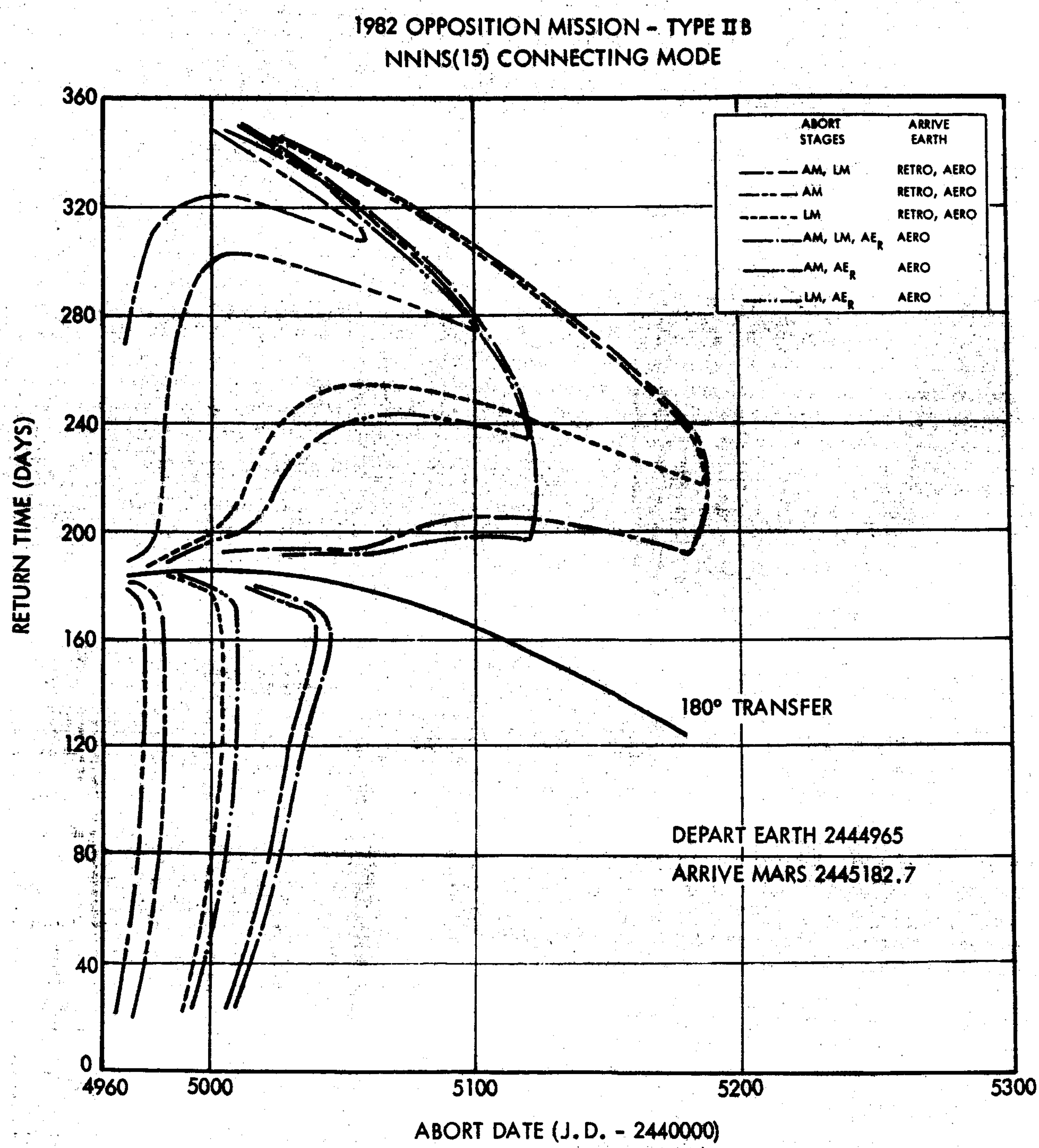

Figure IV-1a. 1982 Opposition Mission Vehicle Abort Capability

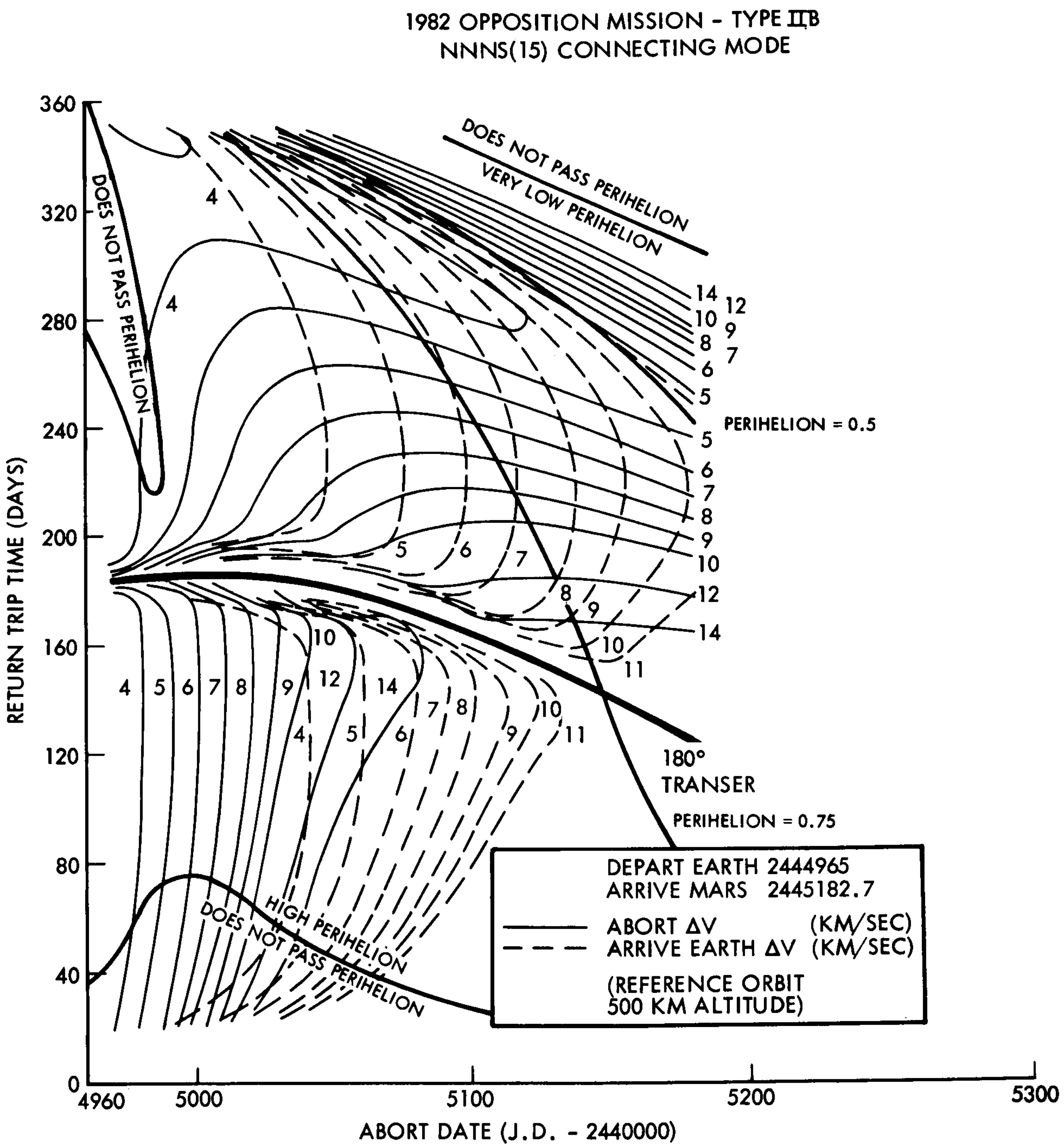

Figure IV-1b. 1982 Opposition Mission Abort Velocity Contour Map

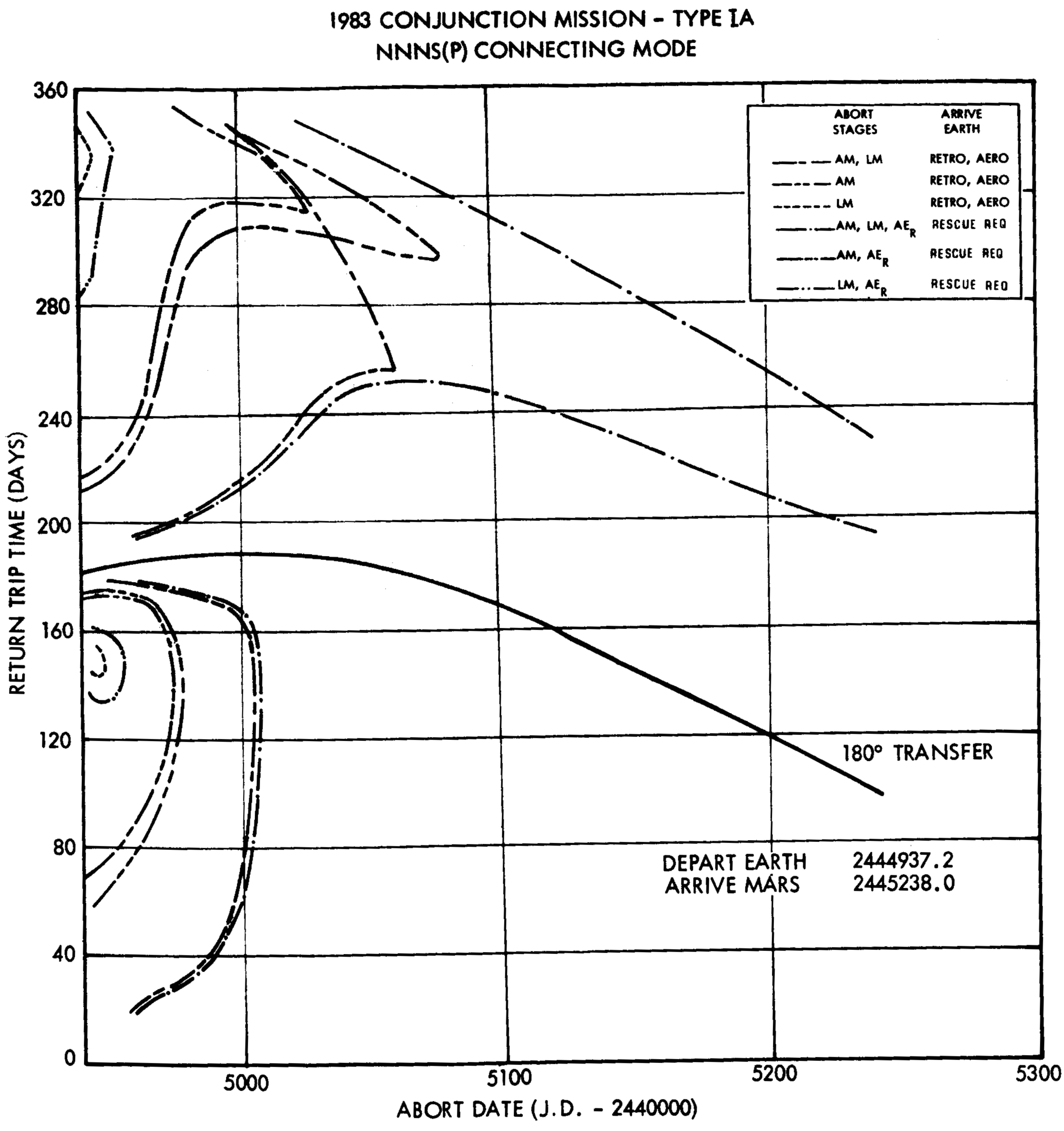

Figure IV-2a. 1983 Conjunction Mission Vehicle Abort Capability

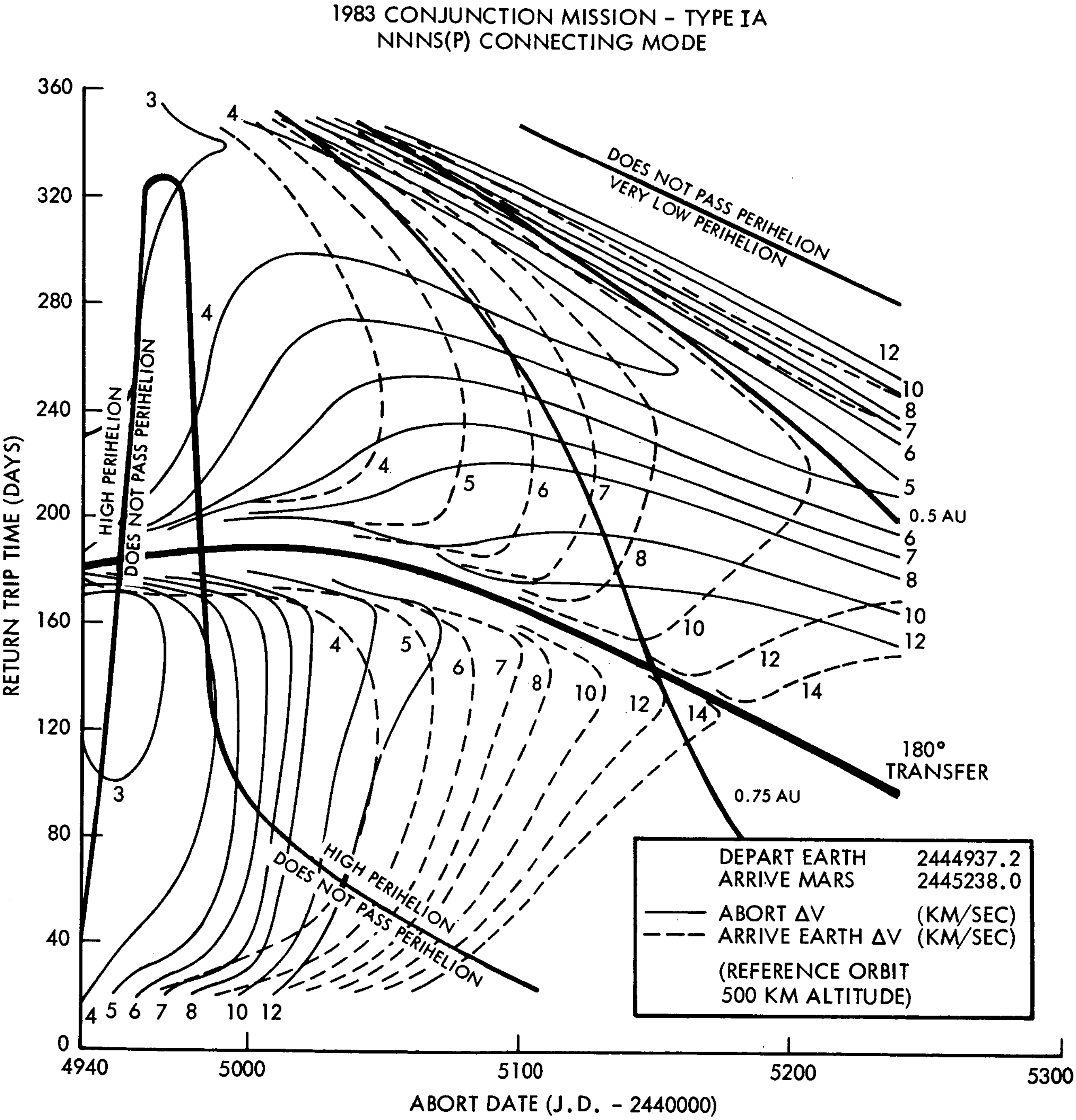

Figure IV-2b. 1983 Conjunction Mission Abort Velocity Contour Map

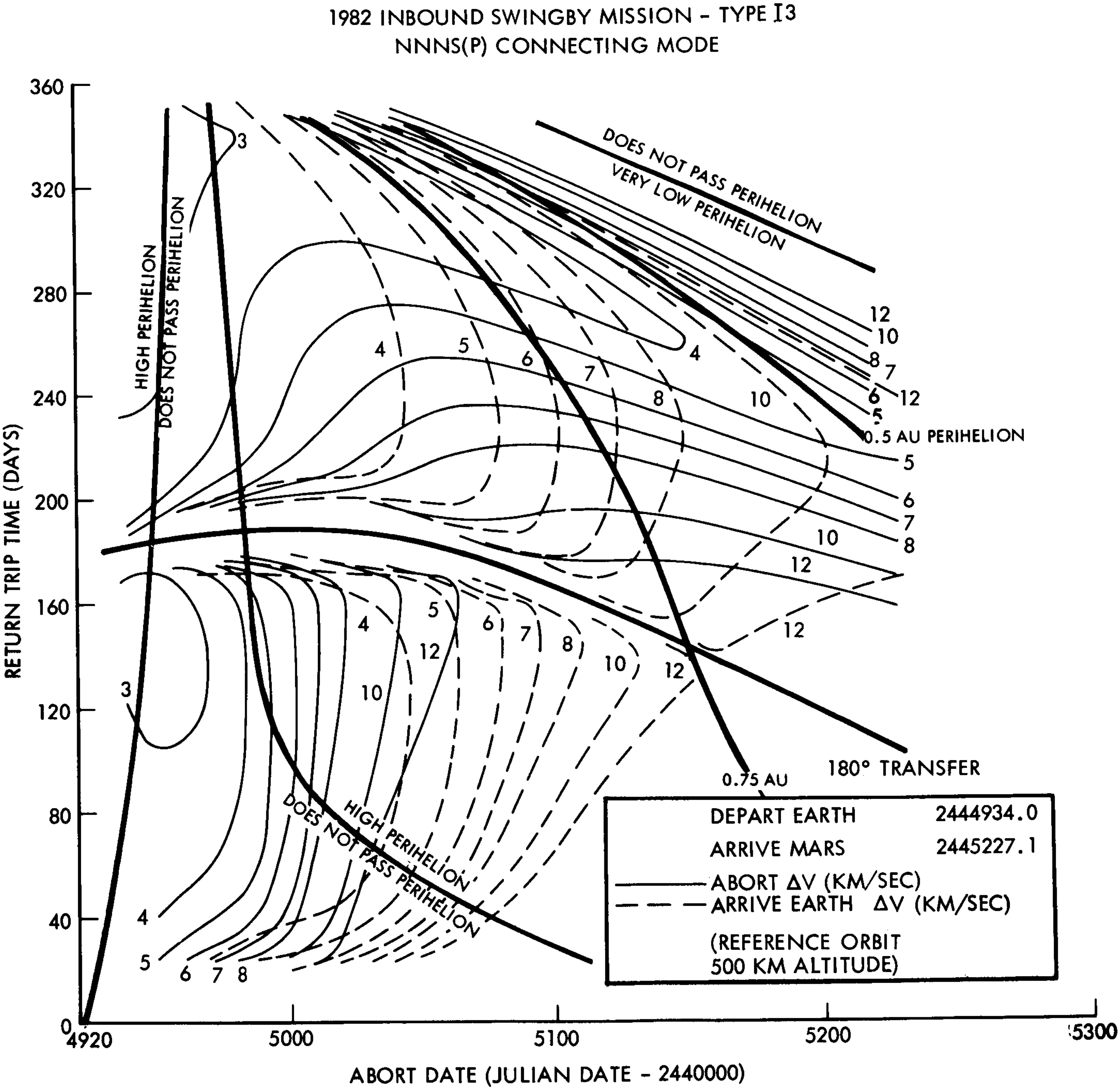

Figure IV-3b. 1982 Inbound Swingby Mission Abort Velocity Contour Map

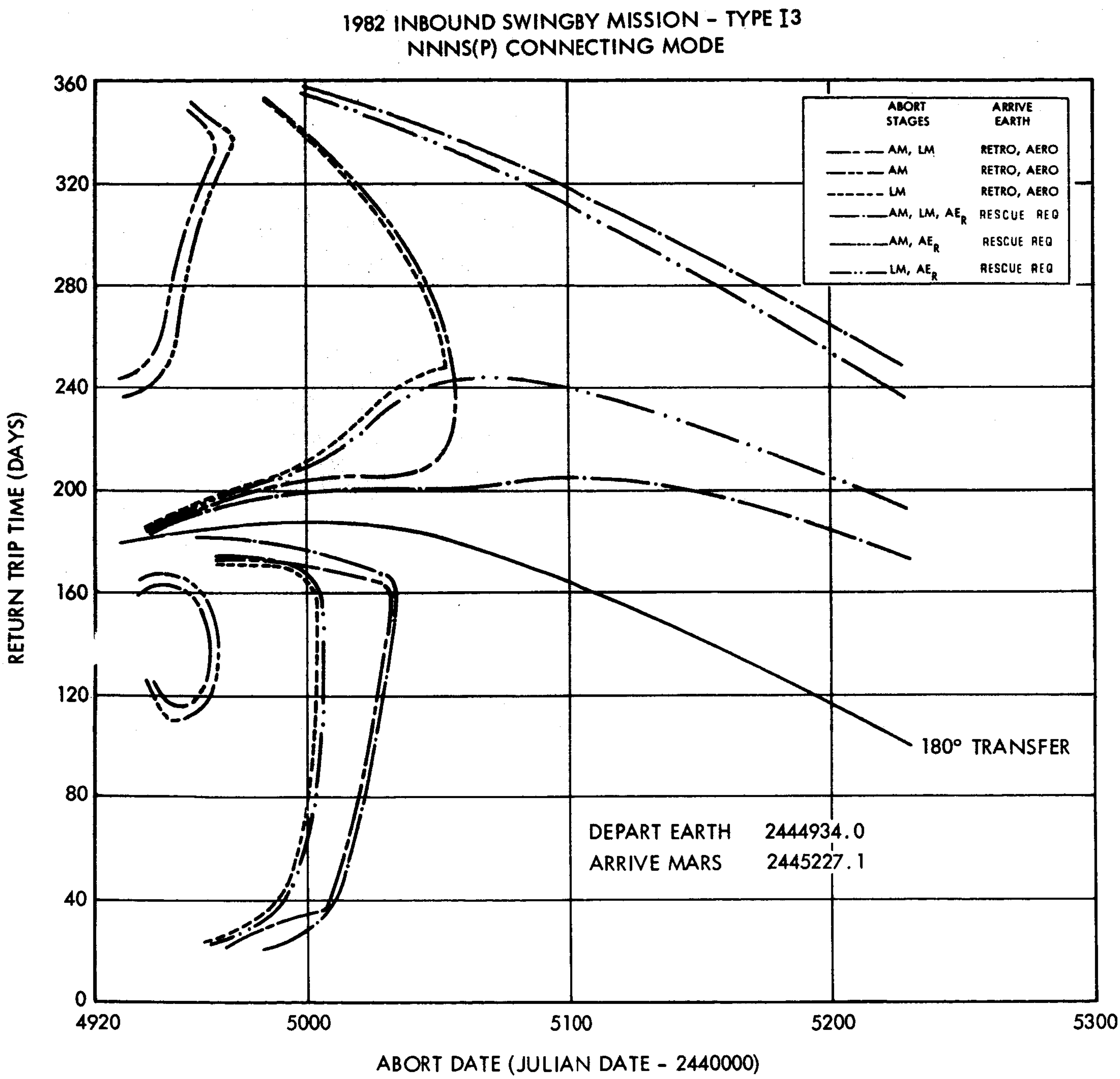

Figure IV-3a. 1982 Inbound Swingby Mission Vehicle Abort Capability

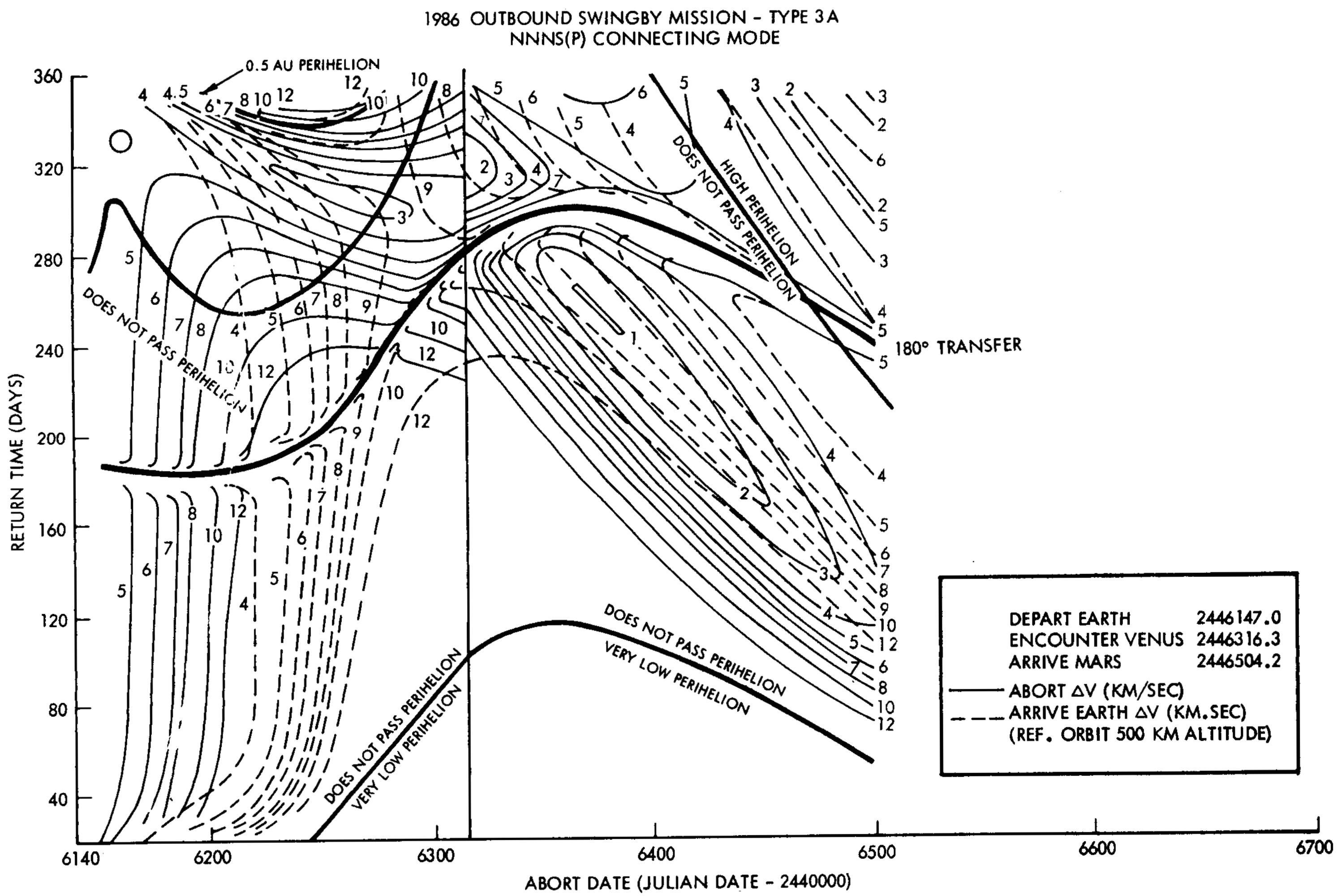

Figure IV-4b, 1986 Outbound Swingby Mission Abort Velocity Contour Map

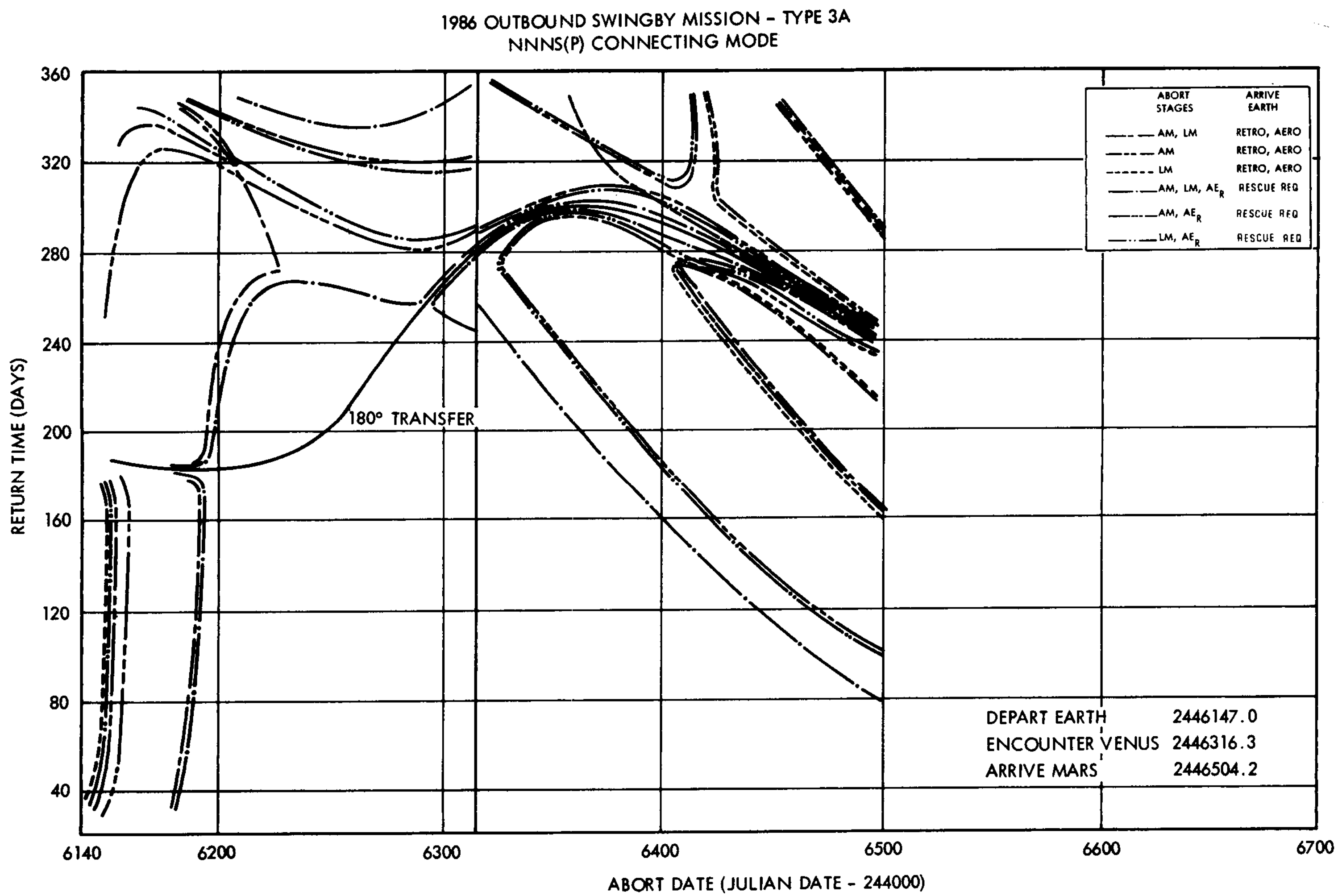

Figure IV-4a. 1986 Outbound Swingby Mission Vehicle Abort Capability

Earth. This condition exists because the vehicle will arrive at Earth at a relative speed greater than parabolic velocity. Since for this vehicle, it has been assumed that its aerodynamic braking capability extends only to parabolic entry velocity, a successful abort would require either a rescue mode by an Earth-based vehicle or a redundant propulsive retro. Therefore, although abort regions are shown on the graphs for three such cases, it must be noted that rescue at Earth or added propulsive or aerodynamic braking capability must be provided to the vehicle.

The results of the abort analysis for the inbound swingby are given in Figure IV-3. These results are essentially similar to those of the preceding conjunction class mission. That is, a successful abort at best can be accomplished only during the first third of the outbound leg if rescue at Earth or an increased arrive Earth braking capability is not available to the spacecraft.

The contour map of aborts from an outbound Venus swingby shown in Figure IV-4, is actually a composite of two contour maps; the map of aborts from the Earth-to-Venus transfer trajectory, and the map of aborts from the Venus-to-Mars transfer trajectory. As for the previous figures, the regions of possible aborts, on each side of the ridge line (180^{o} transfer line), lie within the area partially enclosed by the capability curves.

Without resorting to an Earth rescue mode or an increased arrive Earth braking capability, aborts for the outbound swingby as shown in Figure IV-4, are possible at best during approximately the first half of the Earth-Venus leg and during the last 75 percent of the Venus-Mars leg.

A vehicle Earth braking capability consisting of a retro maneuver to parabolic entry velocity followed by aerodynamic entry is a reasonable assumption for the conjunction class and swingby missions considering their nominal mission arrival velocities. However, as these abort analysis results indicate, the abort capability of the vehicle is severely limited if the retro stage is not available at Earth arrival. Furthermore, it becomes apparent that by increasing the aerobraking capability for all of the missions analyzed, a greater abort flexibility is achieved and the regions in which aborts are possible are increased. The same effect is obtained if the arrive Earth retro stage is sized to be greater than that required for the nominal mission. It should be noted that the effects are additive if both the retro and aerodynamic braking capabilities are increased.

V. LAUNCH AZIMUTH CONSTRAINT ANALYSIS

TASK DESCRIPTION

An analysis was conducted to determine the effects on Mars stopover mission launches due to the constraints imposed on allowable launch azimuths by range safety restrictions and the physical limits on the departure declination achievable for launches from the ETR. The regions in which the interplanetary departure declinations exceed the allowable limits were superimposed on energy contour maps together with points representing the optimum trips for several types of missions, interplanetary trajectories, and vehicle configurations. Mission opportunities from 1975 to 1990 were investigated. Opposition class, conjunction class, and outbound and inbound swingby missions were considered.

For those missions and opportunities for which the optimum (minimum weight) trajectories require Earth departure declinations that exceed the allowable limits, weight penalties were determined for various methods of circumventing the launch azimuth limitations.

ASSUMPTIONS AND CONSTRAINTS

Mission Matrix

The matrix of mission types and launch opportunities that are analyzed in this task is shown in Table V-1.

Table V-1. Launch Azimuth Constraint Mission Matrix

Mission Type	Launch Opportunity
Opposition	1975, 1978, 1980, 1982, 1984 1986, 1988, 1990
Conjunction	1983
Outbound Swingby	1980, 1986
Inbound Swingby	1982, 1984

Vehicle Configuration and System Weights

The scaling laws and system weights used in this task for the propellant tanks and propulsion systems are those given in Section II for the connecting mode. In addition, the scaling laws and system weights used for defining the mission payloads, secondary spacecraft systems, and operational modes are those also given in Section II.

Vehicle Mode

The vehicle mode used for all missions analyzed in this task is the NNNA connecting mode configuration. The specific impulse of the nuclear propulsion system was taken at 850 sec.

Launch Azimuth and Declination 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.

Therefore, 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. Figure V-1 illustrates the limiting declination as a function of launch azimuth for ETR. The azimuth limits based on the allowable firing sectors are also shown. From these constraint envelopes, the maximum achievable declination can be determined. For the nominal azimuth constraints, the maximum achievable declination is 52.4° (at 44° launch azimuth). For the reduced azimuth range which misses Europe, the maximum achievable declination is 36.6° (at 114° launch azimuth).

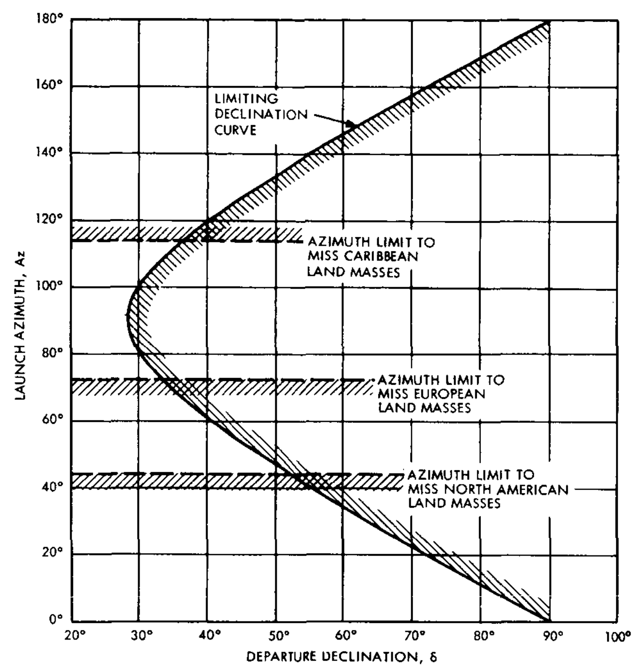

Figure V-1. Launch Azimuth and Declination Limits for ETR

ANALYSIS APPROACH

For each of the optimum (minimum weight) interplanetary missions considered in this task it was necessary to determine if the necessary departure declination could be achieved with nominal launches out of ETR. For all of the launch opportunities considered, Earth to Mars and Earth to Venus trajectory data were used to construct basic contour maps showing the contours of hyperbolic excess speed leaving Earth and arriving at Mars or Venus. The regions where the Earth departure declinations exceed the limits of 36.6° and 52.4° were superimposed on the contour maps. Points were plotted on these maps representing all of the optimum missions. From these graphs it was easily ascertained which missions require Earth departure declinations that exceed the achievable limits.

Next, three alternative modes of carrying out the mission were evaluated for those missions exceeding the declination limits. This evaluation was made by determining the weight penalty associated with each of the three mission alternatives. The three alternative modes were:

- o Make a plane change during the parking orbit escape maneuver to reach the necessary declination for the "optimum" trip.
- o Use a non-optimum outbound trip for which the departure declination does not exceed the achievable limit.
- o Use the opposite type outbound trajectory (which in all cases required declinations less than the achievable limit).

RESULTS AND DISCUSSIONS

Typical results of the launch azimuth constraint analysis, i.e., the energy contour plots with superimposed regions of Earth departure declinations exceeding 36.6^{o} and 52.4^{o}, are given in Figure V-2 for 1982 opportunity missions together with the points that represent the trips leaving Earth for the various mission types. Table V-2 contains the detailed trajectory data for these missions including the weight penalties associated with the three alternative modes used for circumventing the declination limits for those optimum missions that exceed the limits.

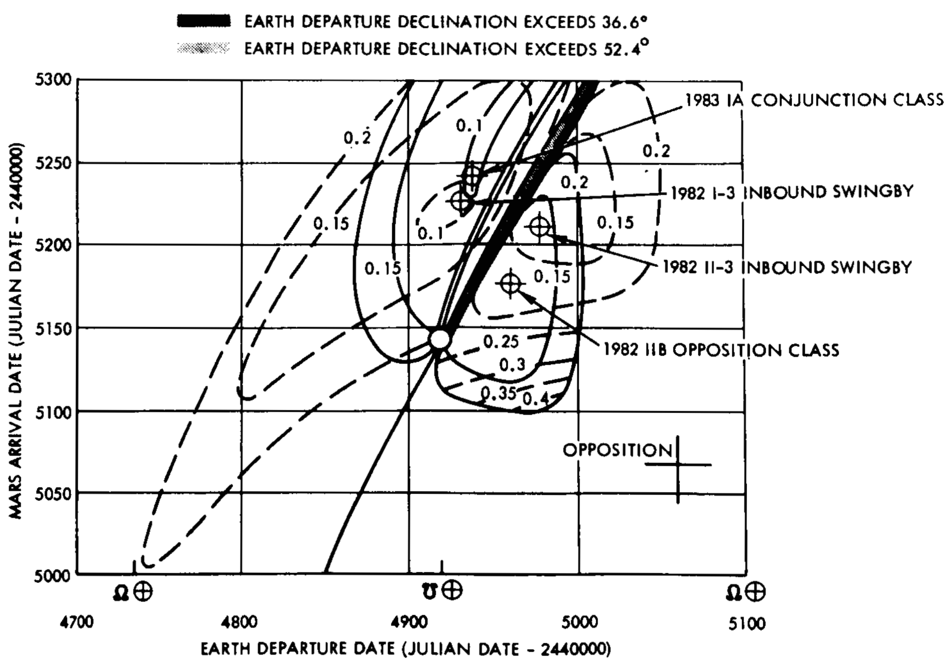

Figure V-2. 1982 Earth Departure Declination Limits

Table V-2. Effects of Earth Departure Declination Limits

Alternative Modes for Missions Exceeding Declination Limit of 36.6° (4)

Planet	Mission Class	Year and Type	Earth Departure Date (J.D. - 2440000)	Mars Arrival Date (J.D. - 2440000)	Venus Arrival Date (J.D. - 2440000)	Earth Departure Declination (Degrees)	Initial Weight for Optimum Mission (10^6 lbs.)	Opposite Type Trajectory: Vehicle Weight for Opposite Type Trajectory (10^6 lbs)	Opposite Type Trajectory: Weight Penalty for Opposite Type Trajectory (%) (1)	Plane Change Out of Parking Orbit: ΔV for Plane Change if out of Azimuth Range (km/sec) (2)	Plane Change Out of Parking Orbit: Vehicle Weight with Plane Change to Achieve Declination (10^6 lbs.)	Plane Change Out of Parking Orbit: Weight Penalty Due to Plane Change (%)	Non optimum Outbound Trajectory: Earth Departure Date for Non optimum Trajectory (J.D.-2440000) (3)	Non optimum Outbound Trajectory: Vehicle Weight for Non optimum Trajectory (10^6 lbs)	Non optimum Outbound Trajectory: Weight Penalty Due to Non optimum Trajectory (%)
Mars	Opposition	1975 IIB IB	2667.6 2594.8	2864.1 2857.1		45.7	2.558	2.688	5.1	0.524	2.854	11.6	2684.0	2.655	3.8
Mars	Opposition	1978 IIB IB	3433.9 3375.1	3648.4 3655.2		48.0	2.725	2.923	7.3	0.791	3.271	20.0	3454.8	2.926	7.4
Mars	Opposition	1980 IIB	4192.5	4425.8		33.4									
Mars	Opposition	1982 IIB	4963.7	5180.6		-6.5									
Mars	Opposition	1984 IIB	5747.9	5943.5		-35.0									
Mars	Opposition	1986 IIB IB	6540.5 6454.8	6719.5 6706.0		-51.2	1.724	2.064	19.7	1.484	2.408	39.7	6571.0	1.979	14.8
Mars	Opposition	1988 IIB	7337.3	7516.7		8.1									
Mars	Opposition	1990 IIB IB	8128.5 8050.0	8317.9 8299.2		40.7	2.532	2.697	6.5	0.126	2.598	2.6	8136.4	2.554	0.9
Mars	Conjunction	1983 IA	4937.5	5239.3		27.3									
Mars	Out. Sw.	1980 3A 3B	3838.3 3840.4		3998.4 4000.8	2.3 3.6									
Mars	In. Sw.	1982 13 113	4933.7 4981.1	5226.5 5221.3		26.4 -7.5									
Mars	In. Sw.	1984 115 15	5735.0 5656.7	5916.9 5912.2		-37.2 0.2	2.288	2.344	2.4	.005	2.289	0.07	5735.8	2.290	0.09
Mars	Out. Sw.	1986 3A 3B	6148.2 6148.5		6317.2 6317.4	20.1 20.1									
Venus	Opposition	1980 IIB	4343.0		4452.3	-9.7									

Notes:
(1) Weight penalty based on increase in weight for type I trajectory over optimum type II trajectory.
(2) ΔV penalty for doing a plane change during the single burn escape maneuver; two possible azimuth ranges were considered; 72° to 114° giving a maximum achievable declination of 36.6° (at azimuth = 114°); 44° to 114° giving a maximum achievable declination of 52.4° (at azimuth = 44°); the declination does not exceed 52.4° for any of these missions.
(3) Mars arrival date is held fixed and the nearest depart Earth date outside the declination limits is found.
(4) Vehicle configuration for all missions; NNNA, 850 sec, connecting mode.

If the nominal azimuth range of 44° to 114° is allowed for the manned Mars missions from 1975 to 1990, no declination constraint problems will be encountered. However, if range safety restrictions require using the launch azimuth range of 72° to 114°, then five of the optimum missions analyzed will require adjustments to compensate for the declinations constraints. The results obtained by using the alternative launching modes for these five missions are shown on Table V-2.

Of the three alternative modes considered, the use of a nonoptimum outbound trajectory, in general, required the lowest vehicle weight increase to compensate for the declination constraints. The opposite type outbound trajectories, i.e., a type I in lieu of a type II, gave the next lowest weight increase for all except those missions only slightly out of the declination limits. The plane change maneuver during departure generally required greater weight increases than the other two methods of compensating for the declination constraints.

The mission most affected by the declination constraints is the 1986 opposition class mission, shown in Figure V-3. The optimum IIB mission requires a very low vehicle weight, but the departure declination associated with the optimum trip is -51.2°. The weight penalties to compensate for the declination constraint if the nonoptimum or the opposite type outbound trajectories are used are 14.8 percent and 19.7 percent, respectively. However, even with these severe penalties, the 1986 opposition class mission requires lower vehicle weight than most of the other missions.

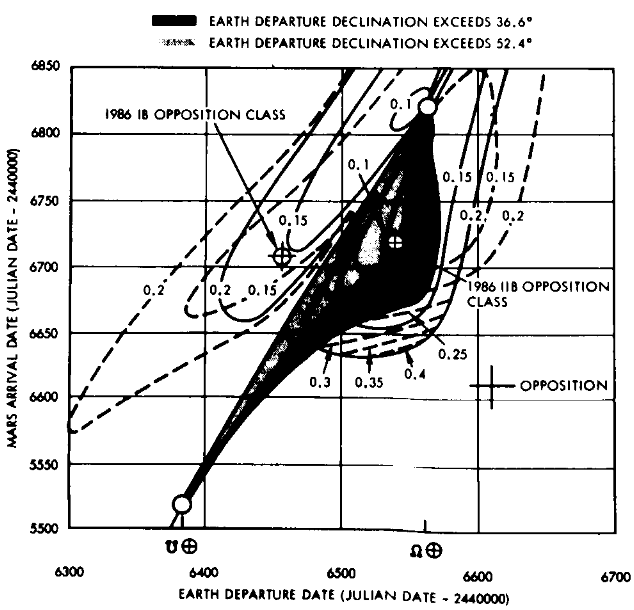

Figure V-3. 1986 Earth Departure Declination Limits

The other four optimum missions that exceeded the declination limits, viz, the 1975, 1978, and 1990 opposition class and the 1984 swingby missions, incur weight penalties that are less than four percent when the nonoptimum outbound trip is employed.

This analysis did not consider any launch window declination effects for departing Earth. It is likely that the nonoptimum trips used to avoid the region of declination constraints would be undesirable when launch window requirements are taken into account. The launch window effects would place a constraint on the start of the launch window as well requiring the nominal date or center of the window to shift to a later depart date. It appears likely that if declination constraints and launch window requirements were considered simultaneously, the opposite type outbound trajectory would yield the lowest vehicle weight.

VI. SUMMARY

Due to the diverse and detailed nature of each of the tasks in this study, an overall summary of the results in depth is neither warranted nor possible in this section without rendering a repetition of the discussion included at the end of each section. Therefore, this summary is limited to a recapitulation if only the more salient results for each task.

SWINGBY MISSION ANALYSIS

The results of the swingby mission analysis task showed that for manned stopover missions, the type 3 swingby leg coupled with the long direct leg (type I or B) yielded the lowest weight vehicle in the years 1980 and 1982. However, in 1986, the type 3 swingby with the short, type A direct leg (3A round trip trajectory) leads to both a lower weight vehicle and shorter trip time. In 1984, an inbound type 5 swingby is best for the all propulsive modes (NNN or CCC modes) while the outbound type 5 swingby is preferable for the Mars aerodynamic braking modes (NAS or CAS modes); the long direct leg is best for both of the latter mission types. The vehicle weights for swingby missions for the NNNA mode increase in the following order: 1982 (minimum), 1986, 1980 and 1984 (maximum). For the NASA mode, the weight is a minimum in 1986, and increases in 1980, 1984, and 1982 (maximum). A comparison of the three different mission types shows that for the NNN mode, the opposition class mission yields the minimum vehicle weight in 1984 and 1986: the swingby mission is minimum in 1982; and a conjunction class mission yields the minimum vehicle weight in 1980. The powered swingby analysis revealed that the powered swingby produces no weight savings over the unpowered swingby for either the type 1 or type 3 trajectories.

CONJUNCTION CLASS MISSION ANALYSIS

The conjunction mission analysis showed that the type IA trajectory (long outbound, short inbound) yielded the minimum weight vehicle. The vehicle weight was eleven percent less than for the shortest trip time trajectory (type IIA) but had a four percent increase in total trip time, 920 to 956 days.

LAUNCH WINDOW ANALYSIS

As mentioned previously, the launch window analysis is in the process of being revised and the results of this task will be presented in a supplemental report at a later date.

MISSION ABORT ANALYSIS

It was shown for all missions analyzed that aborts were generally possible during the first third to first half of the outbound leg duration. These possible abort regions can be extended to cover essentially the entire outbound leg durations by providing the vehicle with sufficient retro and aerodynamic braking capability. The combined retro and aerodynamic braking capability would have to be increased to permit braking at Earth for Earth arrival velocities from 15 to 18 km per sec (approximately 50,000 to 60,000 ft per sec).

LAUNCH AZIMUTH CONSTRAINT ANALYSIS

It was shown in the launch azimuth constraint analysis that for all mission types and launch opportunities considered, launches are possible if the nominal allowable firing sector (departure declination limit of 52.4°) can be used. If the departure declination limit of 36.6° is imposed, the optimum 1975, 1978, 1986, and 1990 opposition class missions, and the 1984 inbound swingby mission are not possible. By resorting to the long (type I) direct leg for these missions, the launches are possible but the vehicle weights are increased by 2 to 7 percent for all missions except the 1986 opposition class; the vehicle weight for this latter mission is increased by approximately 20 percent.

VII. REFERENCES

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