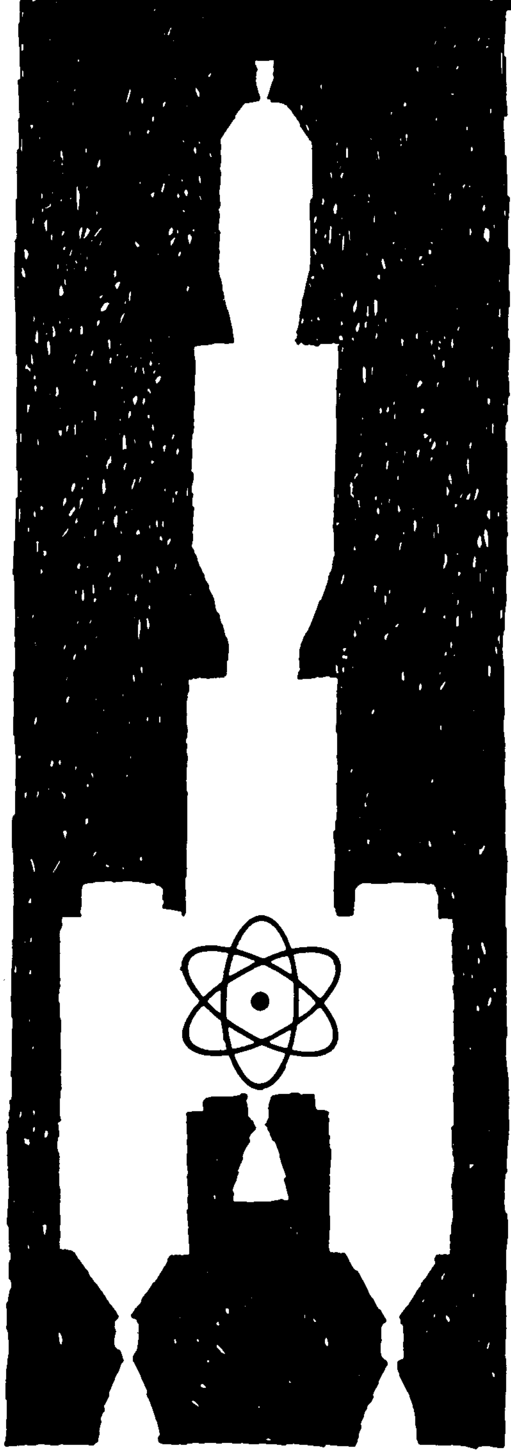
MISSION ORIENTED ADVANCED NUCLEAR SYSTEM PARAMETERS STUDY

DECEMBER 1966
PHASE IV FINAL REPORT



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RESEARCH AND TECHNOLOGY IMPLICATIONS REPORT

FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

BY

TRW SYSTEMS
AN OPERATING GROUP OF TRW INC

01977-6007-R000

MISSION ORIENTED STUDY

OF

ADVANCED NUCLEAR SYSTEM PARAMETERS

Phase IV. Final Report

Volume III

Research and Technology Implications Report

for

George C. Marshall Space Flight Center

National Aeronautics and Space Administration

by

TRW Systems

Redondo Beach, California

Volume III	Research and Technology Implications Report
Prepared by:	A. R. Chovit, Analytical Research Operations L. D. Simmons, Mission Design Department
Approved:	A. R. Chovit, Project Manager Mission Oriented Study of Advanced Nuclear System Parameters
	R. M. Page, Manager Analytical Research Operations

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 a summary and the details, respectively, 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 recomendations. 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 Divison; Mr. P. G. Johnson, SNPO-W; and R. J. Harris, W. Y. Jordan and D. R. Saxton, MSFC.

ABSTRACT

A discussion is presented of the areas of research and technology in which further efforts would be desirable based on the results of a study of manned Mars stopover missions in the decade following 1980. This study included comparison of opposition class, conjunction class, and Venus swingby missions; an analysis of launch window provisions; an analysis of mission aborts; and an investigation of launch azimuth constraints. The areas of research and technology that are discussed include the determination and estimates of system and performance assumptions, supplementary research of operational criteria, and the implications the study results have on future mission analyses.

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I. INTRODUCTION

The results obtained during Phase IV of the Mission Oriented Advanced Nuclear System Parameters Study suggested a number of areas in which further effort would be desirable to support future manned interplanetary mission and vehicle planning studies.

STUDY OBJECTIVES AND TASKS

The basic objectives of Phase IV of this study were to expand the mission evaluations performed in the earlier phases to include trade-offs, mission mode comparisons, and parameter 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. (Due to an error in a computer program used in the Launch Window Analysis, many of the final results obtained for this task were invalid. Therefore, the launch window analysis is in the process of being revised and the results will be presented in a supplemental report at a later date.) The results of the first two tasks together with available results from the previous study phases (Ref 1), permitted a detailed comparison of various mission aspects for the swingby, conjunction class, and opposition class missions. The last three tasks were concerned with investigations of three more or less independent mission operational requirements and constraints.

STATE-OF-THE-ART ESTIMATES

At the outset of the study it was necessary to establish for all of the tasks many technological capability and system performance assumptions. A number of these assumptions were approximate or tentative in nature. In some cases, they were based on extrapolations of current technology or on the results of related NASA and industry studies; in other cases significant parameters were varied over a range of values within which, it was assumed, the parameters describing the future state-of-the-art and system requirements would fall. Since the vehicle's performance and weight requirements were a function of those assumptions, it naturally followed that the mission performance characteristics and the mission comparisons would be influenced by the choice of these assumptions.

Primarily, the assumptions were based on estimates of future state-of-the-art and an extrapolation of the performance of existing systems into the 1980 time period. The validity and uncertainty of these estimates could possibly leave the interpretation of the study results open to question. In some instances, a state-of-the-art capability was assumed feasible for the 1980 time period which applied to only one specific trajectory mode or vehicle configuration. Obviously, if the assumed state-of-the-art capability were not attainable in that time period, any comparisons of that mode or configuration with others not requiring the assumed capability would be invalid. Particular examples of this include the duration of man's ability to exist and function in space; the navigational accuracies necessary and attainable for Venus swingby missions and Mars aerodynamic capture; and the maximum arrival velocities for which aerodynamic braking at Earth would be possible.

Secondly, other assumptions were made to define the performance and weight of vehicle systems for this time period. Some of these assumptions, such as midcourse correction parameters and attitude control and orbit adjustment requirements, affect the analysis results to a minor extent and even large errors in these variables would not invalidate the final results and conclusions. Other assumed performance parameters such as life support expendables could measurably bias the final conclusions only if they were grossly underestimated.

Finally, a number of performance and weight assumptions were made for which even relatively small variations could alter the mission comparisons and analysis results. These include the major propulsion systems' specific impulse and weight, payloads and mission modules, and weight provisions required for aerodynamically decelerating the spacecraft at both Mars and Earth arrival. In some instances, these variables were parameterized. In these cases specific results and comparisons based on single points in the parameter range are only as conclusive as the validity of the parameter value selected within the range.

In a study of this nature and at this time in the planning stage, it obviously is not possible to predict or extrapolate accurately into the future the many technological and system factors that can influence the study results. Nevertheless, the relative effect that these factors have on the study conclusions must be noted. Furthermore, subsequent research in depth should be initiated in order to attempt to verify or redefine the more influential assumptions including both the validity of the assumption in a given time period and the associated estimates of performance and weight parameters.

OPERATIONAL CRITERIA

A second set of guidelines established at the beginning of the study was concerned with operational criteria. These guidelines involved choices among alternative operational or system techniques that led to overall operational or analysis constraints. In some cases, two or more alternatives were established, e.g., both the tanking mode and connecting mode for defining parking orbit operations and spacecraft configurations; various failure modes for the abort analysis; and a range of yearly launch opportunities for which the missions were analyzed. In other cases, these criteria were limited to single point values or modes, e.g., the Saturn vehicle payload capability; no major system redundancy; and no auxiliary payloads to be jettisoned or deployed during the transfer trajectories or at Venus for the swingby modes.

A number of the criteria stated above are based on preliminary, associated investigations, others have been adopted to limit this study to a reasonable scope, while still other possible alternative criteria have been eliminated since their adoption would reflect a second phase refinement in the detail of the mission analysis. Nevertheless, revisions in these criteria or the adoption of alternative criteria can materially alter the results of a study of this nature as well as produce additional insight into many aspects of manned space flight. Accordingly, supplementary research on some of these criteria appears warranted in order to verify the operational or system techniques involved. Also additional mission analysis should be conducted for possible alternative criteria in order to assess their relative merit.

FUTURE MISSION ANALYSIS IMPLICATIONS

In general, past mission analysis studies of manned planetary flight have concentrated on investigations of specific problems within relatively narrow constraints.

Thus for the most part, the interactions that can occur when two or more problems are considered together were not integrated into the studies. As a result of the analyses performed in this study, it became evident that the conclusions obtained by investigating a seemingly independent problem area were dependent upon the results of analysis or assumptions made for other problem areas. For example, an analysis of the weight penalties associated with Earth launch windows cannot be separated from an analysis to determine the weight penalties imposed by launch declination limits due to launch azimuth constraints. Both of these analyses, in turn, have an interaction with the variation of the Saturn payload capability at different parking orbit inclinations and the launch opportunity and trajectory modes being considered. Obviously, the converse is likewise true; i.e., in order for comparisons of launch opportunities and trajectory modes to be completely valid, the effects of launch window provisions and launch azimuth constraints must be simultaneously considered.

As the results of mission analysis studies become more definitive in specifying future system and mission requirements, it becomes more important to note these interactions and their effects and to include them in future mission and planning studies.

The remainder of this volume discusses the areas of desirable future effort in terms of these three categories, i.e., State-of-the-Art Estimates, Operational Criteria, and Future Mission Analysis Implications, for each of the tasks of this study.

II. MISSION MODES COMPARISON

TASK DESCRIPTION

The initial weight requirements in Earth orbit were determined for manned Mars stopover missions employing the Venus swingbys, opposition class, and conjunction class modes. The mission analyses included mission opportunities from 1980 to 1986. These investigations included variations in the vehicle propulsive and deceleration systems both at Mars and at Earth, in nuclear engine performance parameters, vehicle structural scaling laws, and payload weights for the conjunction class missions.

The results of these analyses were compared to illustrate the effect on initial vehicle weight of variations in launch opportunities, mission and trajectory types, performance parameters, and vehicle systems and scaling laws.

STATE-OF-THE-ART ESTIMATES

The major comparison made in this study of the three basic mission trajectory modes, viz, opposition class, Venus swingby, and conjunction class missions, was on the basis of the initial vehicle weight required for each of the different modes. Due to the inherent nature of the conjunction mission trajectory profile, a long dwell time or stopover period (approximately 400 days) is required at Mars compared to that for either the opposition class or Venus swingby missions (20 days). Accordingly, the mission payloads and life support expendable weights allocated to the conjunction class mission were chosen arbitrarily to be approximately 50 percent greater than the weights for the other two types of missions to account for an increased crew size and crew and system requirements dictated by the longer dwell time at Mars. The payload and expendable weights used for the respective missions in the comparisons are given in Table II-1.

Table II-1. Payload and Expendable Weights

	Mission Mode	
Payload	Opposition and Swingby	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

Comparisons of the conjunction and opposition class missions for the NNNA* modes indicated that the vehicle weight for the typical 1983 conjunction class mission was less than the opposition class mission in two years, 1980 and 1982; approximately equal in 1984; and greater in 1986.

A comparison of the conjunction and swingby missions indicated that the vehicle weight for the conjunction class mission was less than the swingby mission in 1984; approximately equal in 1980; and greater in the two years, 1982 and 1986.

As part of another task in the study, an analysis of the conjunction class mission was made in which the payload and expendable weights were parameterized to determine their effect on initial vehicle weight. These weights are given in Table II-2.

Table II-2. 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

Payload set 3 corresponds to the conjunction class mission payload values used for the previous comparisons of mission modes comparisons and payload set 1 is equal to the payload values assumed for the opposition class and Venus swingby missions.

Now, if the mission mode comparison is extended to include all four sets of conjunction class payloads, the conclusions reached previously become qualified on the basis of conjunction class mission payload assumptions. This fact is illustrated in Figure II-1 which gives the initial vehicle weights for the various modes and opportunities for a NNNA vehicle mode, Mass Fraction Case No. 2 structural scaling laws, and a nuclear engine specific impulse of 800 seconds. (See Volume I or II for vehicle mode and structural scaling law definitions.) Payload sets 1 and 2 for the conjunction class mission yield vehicle weights less than both the opposition class and swingby missions for all years 1980 through 1986, while payload set 4 results in a greater vehicle weight than either of the other two mission types.

*Nuclear propulsion, depart Earth; Nuclear propulsion, arrive Mars; Nuclear propulsion, depart Mars; Aerodynamic braking, arrive Earth.

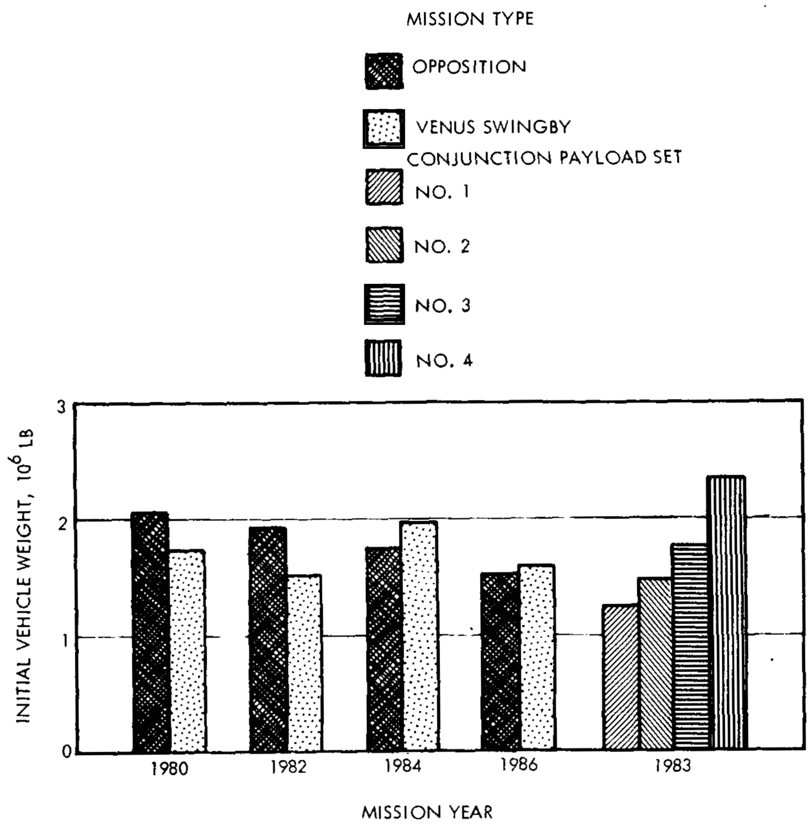

Figure II-1. Mission Mode Comparison, NNNA Vehicle Configuration

Therefore, it is clearly evident that the comparison results are a function of the payload state-of-the-art estimates, i.e., their applicability for each of the yearly launch opportunities, their applicability to this overall time period, and the absolute system and weight values assumed. The prediction and extrapolation of this state-of-the-art into this time period has been based on very preliminary system designs, unverified environmental and operational criteria, and speculative human factor capabilities. Therefore, it clearly follows that more definitive comparative conclusions can be reached only through more research to determine to a greater degree of accuracy man's ability to exist in space, his associated system requirements, his functional mission requirements, the mission experimental requirements, and both the interplanetary and planetary environments in which the spacecraft must operate.

It should also be noted that although greater payload weights were used for the conjunction class mission (payload set 3) in the mission mode comparisons, the payloads for the opposition class and swingby missions were assumed identical. This may have been an invalid payload assumption if similar reasoning as was used for the conjunction class mission is applied. The total trip times for the opposition class missions for these years vary from approximately 445 to 480 days, while the total trip times for the swingby missions range from approximately 560 to 670 days. This represents an average increase of approximately 30 percent in total trip time for the swingby mission over the opposition class mission. Since the total trip time for the conjunction class mission is approximately 50 percent greater than the swingby mission, it appears reasonable that the payload requirements for the swingby mission could be increased. Also it is likely that some sort of Venus probe would be carried to take advantage of the opportunity to further explore that planet. If such changes were made in the vehicle weight configurations, the conclusions of the mission mode comparisons would be altered. Again, additional research into the areas mentioned above would permit a resolution of these uncertainties.

Figure II-2 illustrates the same payload dependent variations for a similar mission mode comparison for a NASA vehicle configuration, Mass Fraction Case No. 2 structural scaling laws, and a nuclear engine specific impulse of 800 sec. As in the previous comparison, the basic mission mode comparisons in this study were made for the conjunction class payload set 3 but the results are considerably altered if the payload sets 1, 2, or 4 are assumed.

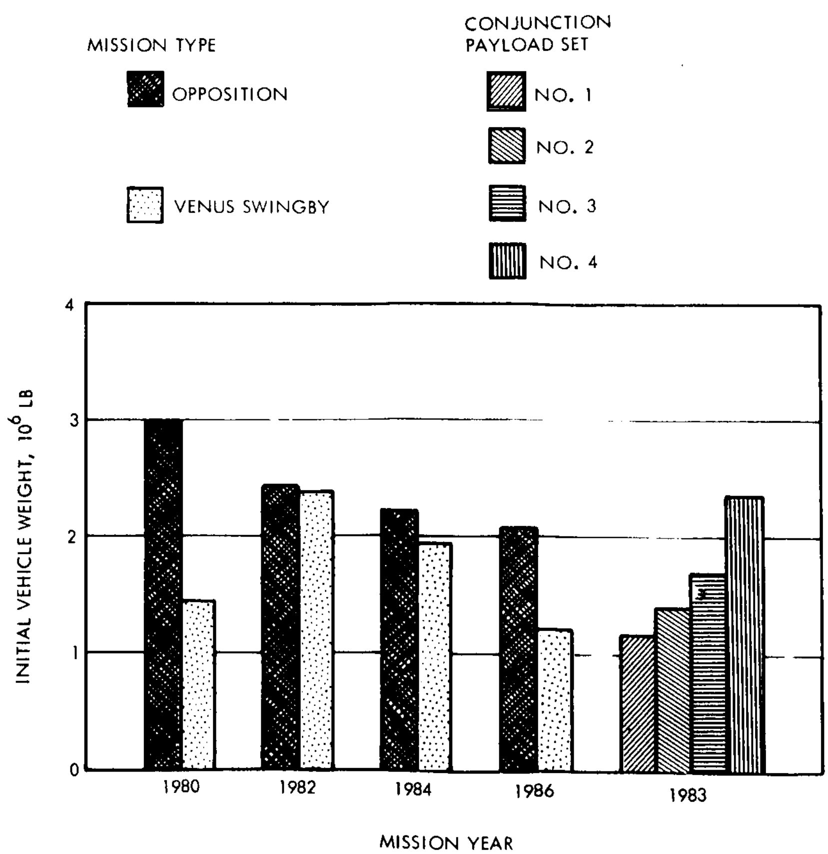

Figure II-2. Mission Mode Comparison, NASA Vehicle Configuration

Figure II-1 and II-2 can also be used to illustrate the difference in initial vehicle weight that results from the employment of propulsive or aerodynamic braking at Mars arrival, i.e., the NNNA and NASA* vehicle configurations. It should first be noted that the use of these two alternative vehicle modes results in different conclusions when the basic three mission modes are compared. This is so because the use of the NASA vehicle configuration affects the vehicle weight for each of the three mission modes in a different manner. For example, the use of the NASA mode instead of the NNNA mode reduces the vehicle weight for the conjunction class mission for all four payload sets (although by varying percentages). On the other hand, the NASA mode increases the vehicle weight for the opposition class mission for all years but decreases the vehicle weight for the swingby mission for all years except 1982 in which year the NASA mode substantially increases the weight.

Since the scaling law employed for computing the system weight for aerodynamically braking at Mars was based on projected technology and tentative Martian atmospheric models, the comparisons of the two vehicle modes as well as the comparisons of the mission modes are completely dependent on the accuracy and applicability to the time period of the scaling law employed. In addition, the use of higher performance chemical cryogenic propellant or nuclear stages in lieu of the liquid storable stage that was assumed for departing Mars in the NASA mode would have the effect of reducing the vehicle weights for the NASA mode. Again, the effect produced by using these alternative cryogenic systems would vary for the different mission modes and years. For those mission modes and years which require relatively large velocity increments for departing Mars, the weight reductions would be large; however, for those with long outbound leg times (outbound swingby missions) or long dwell times at Mars (conjunction class missions), the weight reductions would be lessened due to the propellant vaporized and increased insulation requirements. Another factor which will influence any comparisons is the assumed propulsion stage performance parameters, principally the specific impulse.

It is clear that basic reasearch is required into the technical areas involved above, i.e., aerodynamic braking at Mars, long term cryogenic propellant storage, and propulsion stage performance, if the conclusions reached in mission analysis comparisons such as these are to be completely valid.

*Nuclear propulsion, depart Earth; Aerodynamic braking, arrive Mars; liquid Storable propulsion, depart Mars; Aerodynamic braking, arrive Earth.

Another area in which additional research is required is that of aerodynamic braking at Earth. The arrival velocities at Earth for the conjunction class and swingby missions are only slightly greater than parabolic velocity, while those for the opposition class missions can be as high as as 20 km per sec. Therefore, the determination of the maximum aerodynamic braking capability for this time period is critical. For spacecraft arriving at greater velocities, a retro stage must be employed with its attendant increase in vehicle weight over an all aerodynamic braking stage. The enforced use of a retro stage due to a maximum limit on aero braking velocity would obviously tend to impose a greater weight upon the opposition class missions.

Another research area that concerns a system capability that may or may not be feasible in the 1980 time period is that of the stringent navigational requirements for Venus swingby missions and Mars aerodynamic capture. Since this requirement applies principally to these two modes of operation, its feasibility is critical when these modes are compared with others.

It must be noted that although many of the conclusions reached in studies such as this are dependent on tentative assumptions which are often based on limited data and information, every effort is made at the time to use the latest and most accurate data available. In fact, it is only through the results of these studies that it is possible to focus upon those technical areas requiring major research and further study and to assess the relative effect that the eventual state-of-the-art will have on the overall mission.

However, a continual iteration should be made on this process using the results of such research and study in further mission and system analyses. Such a continuing process will converge on the necessary system design criteria at the earliest possible date, i.e., when all technical data have been established to within the necessary tolerances for design.

OPERATIONAL CRITERIA

The primary operational modes that were adopted for this study and which have an influence in the mission modes comparison are those alternative modes which define the parking orbit operations and spacecraft configurations, viz, the tanking mode and connecting mode. The basic differences in these two modes as they affect the mission analysis are:

1) The tanking mode permits filling and topping off of the propellant tanks in the parking and assembly orbit whereas in the connecting mode the tanks are boosted into the parking orbit in a fully loaded condition and no topping off is permitted. Therefore, any propellant vaporization that occurs in the parking orbit for the tanking mode is replenished before injection into the interplanetary orbit while for the connecting mode, the propellant vaporized is not replaced. This has a dual effect on the connecting mode; that of reducing the maximum propellant capacity available for the mission as well as imposing an inert tank weight penalty (equal to the tank volume required for the vaporized propellant).

2) Since for the tanking mode, the tanks can be launched in an empty or partially filled condition their maximum size is limited by the maximum length payload the Saturn vehicle can launch. On the other hand, the tanks for connecting mode are boosted to the parking orbit in a fully loaded condition and, therefore, their size is limited by the Saturn vehicle's payload weight capability. This has the effect of limiting the maximum size of any given connecting mode propellant tank to a capacity equal to approximately 30 percent of that used for the tanking mode.

3) It was assumed for the connecting mode Earth departure stage that three nuclear engines would always be employed. On the other hand, the number of engines for the tanking mode Earth departure stage was selected so as to provide the optimum thrust for any given mission.

A second criterion which closely interfaces with both the tanking and connecting mode assumptions is the value of the maximum payload that the Saturn vehicle can place in the Earth parking orbit. The specification of this value should be based upon the Saturn state-of-the-art projected to this time period as well as the parking orbit inclination; the latter is dependent upon the declination of the required interplanetary hyperbolic asymptote which varies with each mission mode and opportunity.

The implications these two criteria have on the computed vehicle weights are complex and they involve 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. Therefore, it is difficult to assess how variations in these criteria will affect the mission modes comparisons as well as the comparisons between the alternative criteria themselves. The resolution of this problem lies in three directions. First, all mission analyses should be based upon the most up-to-date analyses and data of the Saturn launch and orbital operations available. Second, in the reporting and the assessment of results of manned interplanetary mission analyses, every effort should be made to continuously consider the uncertainties upon which these criteria are based as well as their probable effect on the results. Finally, research and detailed preliminary studies should be intensified to determine and specify more definitively the operations and system weights associated with the launch, rendezvous, assembly, and checkout requirements for manned interplanetary missions.

III. 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.

STATE-OF-THE-ART ESTIMATES

The approach taken in this task was to compute the impulsive abort and Earth arrival velocities for aborting a mission along its nominal outbound trajectory. The results were then plotted as contours of constant ΔV on a grid of return trip time versus date of abort. 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.

Two contour maps with their associated vehicle abort capability overlays for the opposition class and inbound swingby missions are given in Figures III-1 and III-2, respectively. (The conjunction class mission results produced a set of abort curves very similar in all aspects as those given for the inbound swingby mission on Figure III-2. The curves for the outbound swingby mission may be seen in either Volumes I or II of this series of reports.)

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

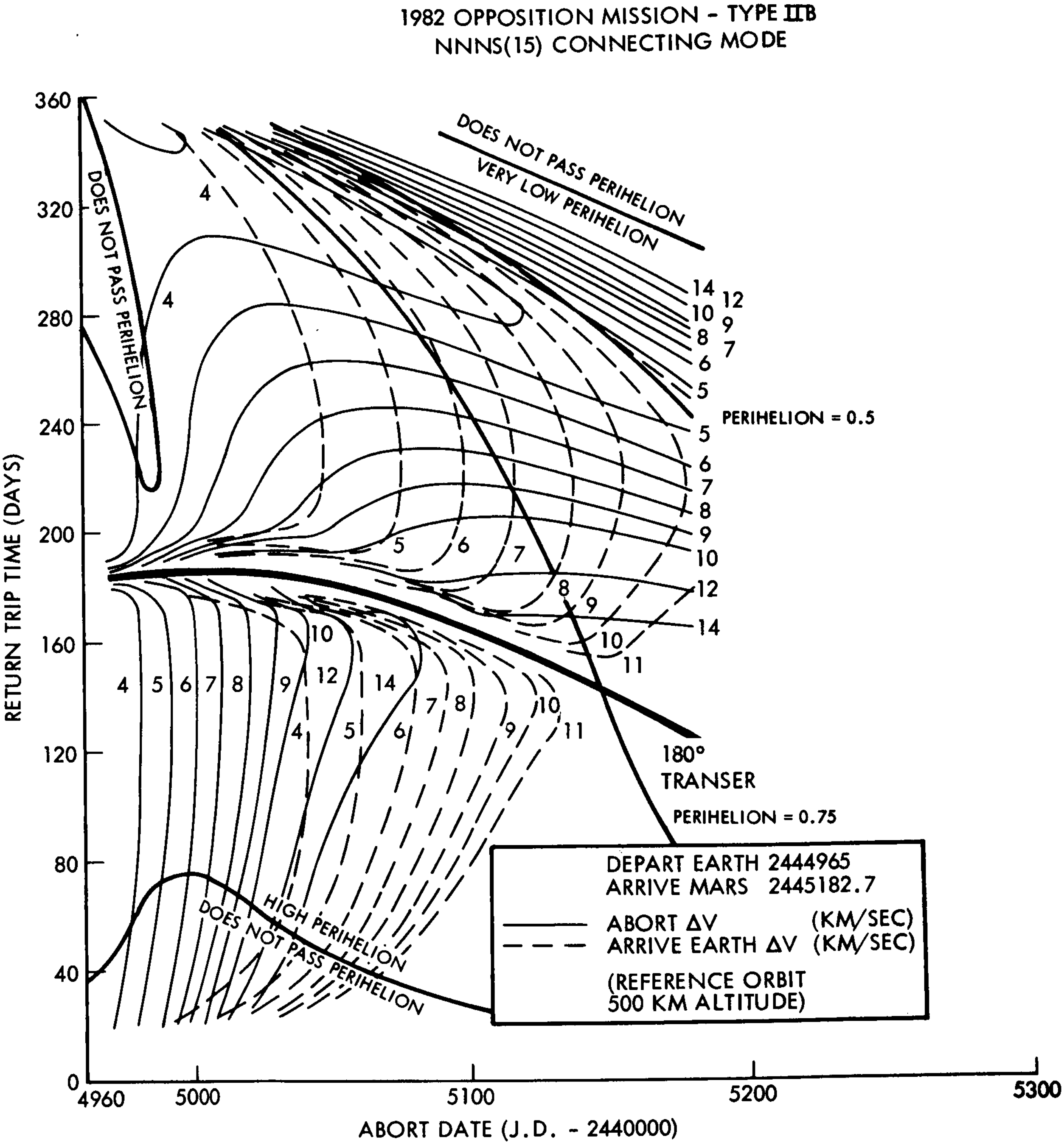

Figure III-1b. 1982 Opposition Mission Abort Velocity Contour Map

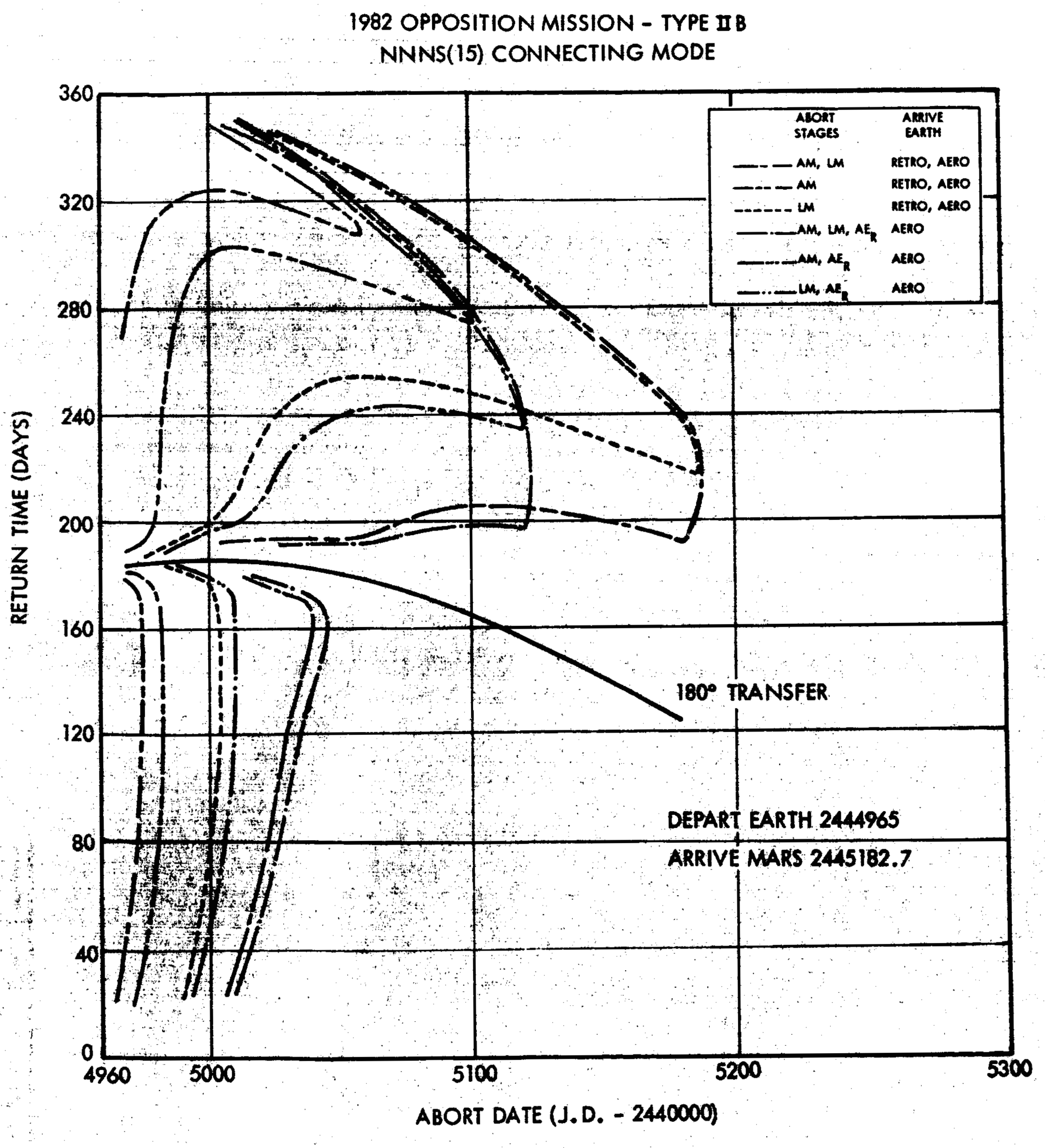

Figure III-1a. 1982 Opposition Mission Vehicle Abort Capability

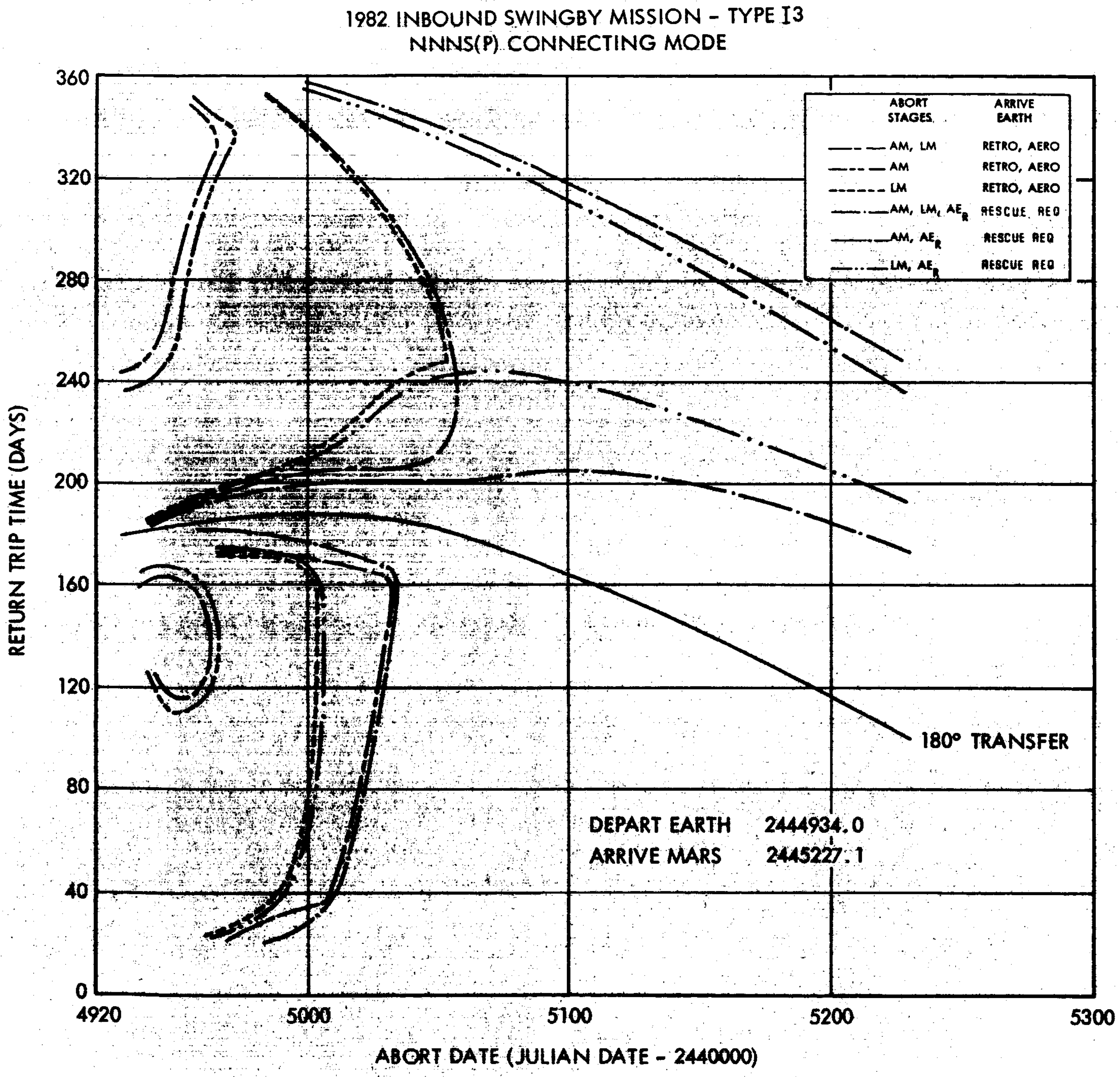

Figure III-2a. 1982 Inbound Swingby Mission Vehicle Abort Capability

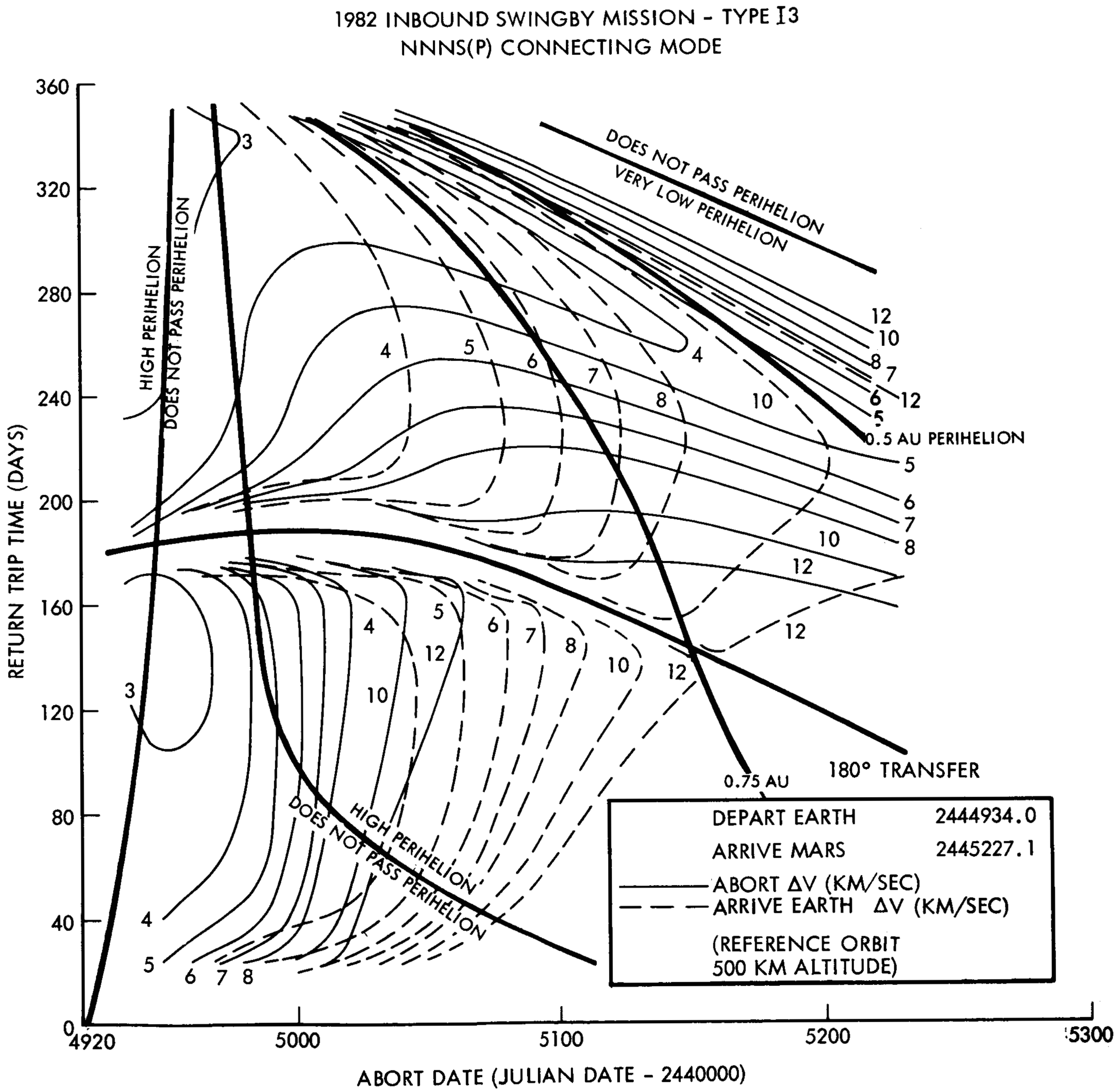

Figure III-2b. 1982 Inbound Swingby Mission Abort Velocity Contour Map malfunction is discovered late, it may be impossible or undesirable to follow one of these "optimum" trips. For such instances the map shows all trips that are still possible and a choice can be made.

Figure III-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 inbound swingby mission shown on Figure III-2 (as well as those for the conjunction class mission) 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 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 major system feasibility assumption affecting the results of this task was that of the aerodynamic braking capability at Earth. For example, 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 arrival velocities for the nominal mission. 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, although of course, this redundancy can appreciably increase the initial vehicle weight. It should be noted that the effects are additive if both the retro and aerodynamic braking capabilities are increased.

In general, the possible abort regions for all of the missions analyzed could be extended to cover essentially the entire outbound leg durations by increasing the vehicle's retro and aerodynamic braking capability to permit braking at Earth for arrival velocities from 15 to 18 km per sec (approximately 50,000 to 60,000 ft per sec). In order to ascertain if this abort capability is reasonably possible, additional research and development effort is required to determine accurately the maximum Earth aerodynamic braking capability for this time period as well as the weights associated with the aerobraking system.

This abort analysis was conducted only for an all nuclear propelled vehicle. If aerodynamic braking at Mars were to be assumed, the abort capability of the vehicle would be severely limited since no arrive Mars propulsive stage would be available for a possible abort maneuver during the outbound leg of the mission. The feasibility of this latter mode should be ascertained as soon as possible to permit an accurate assessment of the vehicle's abort capability as well as the other mission analysis aspects dependent on this information that has been discussed previously.

OPERATIONAL CRITERIA

In conducting this abort analysis task it was necessary to assume certain failure mode criteria. Since there are almost a limitless number of possible vehicle failures that could lead to an abort decision, only the more obvious were selected. These choices produced the six abort capabilities used, i.e., accomplishing the abort maneuver with both the arrive and depart Mars stages, the arrive Mars stage only, or the depart Mars stage only; the arrive Earth retro was assumed to be operable for either the abort maneuver or for Earth retro. Additional abort capabilities are certainly possible such as partial utilization of one or more stages as would be the case in the event of propellant leaks; jettisoning or transferring the propellant of the depart Mars stage in the event that stage is known to be inoperable; jettisoning the mission module in the case of critical abort conditions; the use of alternative abort missions where possible such as Mars flybys or Venus swingbys; and the provision of redundant engines or stages to improve the abort capability as well as the overall mission success probability. Therefore, the study and specification of definitive failure modes and failure probabilities with their attendant consequence upon the subsequent mission operations would provide the information necessary to perform more detailed and meaningful abort analyses. In turn, the abort analyses would reveal the abort capability sensitivities for the specified failure modes.

FUTURE MISSION ANALYSIS IMPLICATIONS

In past mission analyses, the vehicle abort capabilities were generally completely ignored or treated as an independent exercise (as was the case in this phase of this study). Since practically all spacecraft systems have an eventual effect on the vehicle's abort capability (either through its failure implications or through its use for the abort functions or both) any analysis which can eventually lead to their final specification should consider all aspects of aborting the mission.

Therefore, as the planning stages for manned interplanetary missions become more definitive in terms of guiding and specifying the mission and vehicle requirements and the system research and development efforts, the analysis of mission aborts must be closely integrated into the overall analyses and comparisons of the spacecraft systems, the trajectory types, and mission operations.

IV. 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.

OPERATIONAL CRITERIA

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° imposed by two launch azimuth constraint models 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).

Typical results of the launch azimuth constraint analysis, i.e., the energy contour plots with superimposed regions of Earth departure declinations exceeding 36.6° and 52.4°, are given in Figure IV-1 for the 1982 opportunity missions together with the points that represent the trips leaving Earth for the various mission types.

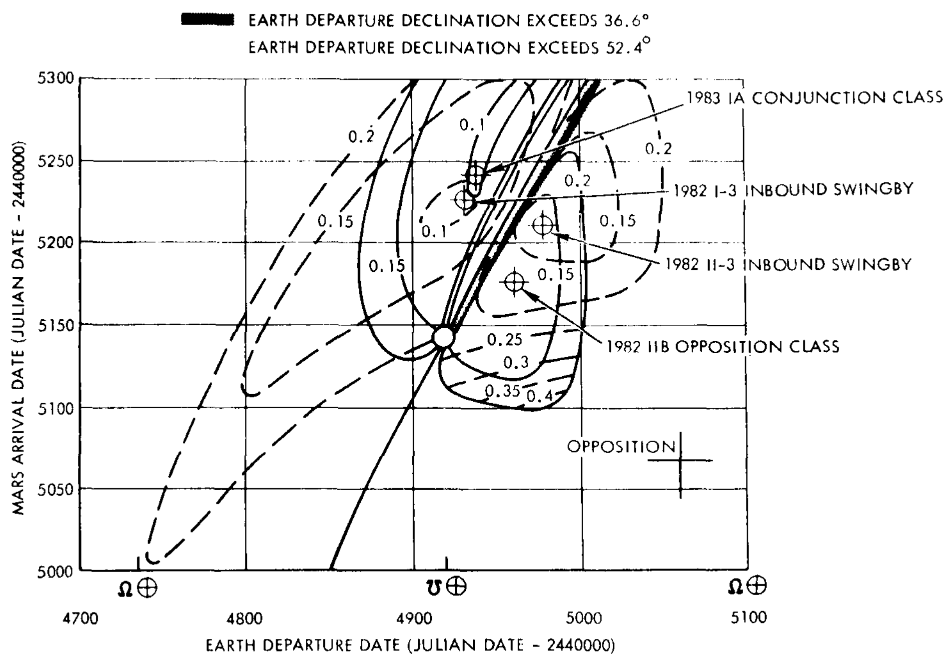

Figure IV-1. 1982 Earth Departure Declination Limits

If a nominal azimuth range of 44° to 114° (maximum declination of 52.4°) 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° (maximum declination of 36.6°), then five of the optimum missions analyzed will require adjustments to compensate for the declination constraints.

The mission most affected by the declination constraints is the 1986 opposition class mission, shown in Figure IV-2. The optimum IIB mission requires a very low vehicle weight, but the departure declination associated with the optimum trip is -51.2^{o}. The weight penalties to compensate for the declination constraint if the non-optimum or the opposite type outbound trajectories are used are 14.8 percent and 19.7 percent, respectively. In addition to the 1986 opposition class mission, if the departure declination limit of 36.6^{o} is imposed, the optimum 1975, 1978, 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.

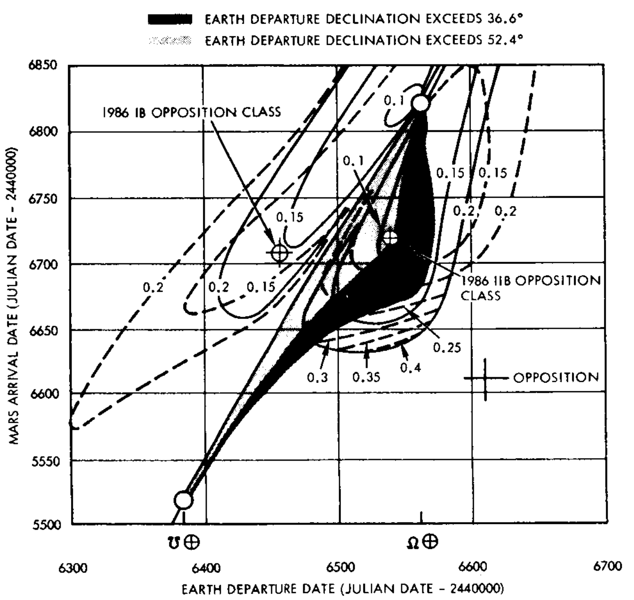

Figure IV-2. 1986 Earth Departure Declination Limits

The necessary declination of the Earth departure hyperbolic asymptote for any given mission during a given launch opportunity determines the nominal inclination of the Earth parking orbit. As mentioned previously, the Saturn vehicle Earth orbital payload capability is a strong function of the desired orbit inclination for launches from a given site. Since the departure asymptote declinations required for the various missions and opportunities investigated in this task varied from as low as 0.2 degrees to as high as -51.2 degrees, it follows that the Saturn payload weight available in the parking orbit for a single launch will be significantly different for the various missions.

In this study, as in most past studies, the interplanetary vehicles were configured on the basis of a nominal Saturn vehicle payload criterion. In future studies, 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. Also the assumed vehicle configuration can influence to a marked degree the initial weight of the vehicle. Therefore, mission mode and launch opportunity comparisons such as those performed in this study must eventually consider the necessary parking orbit inclinations and their effect on available parking orbit payloads if these comparisons are to reflect accurately the overall operational requirements.

Consequently, studies should be initiated to determine the latest projected Saturn payload capabilities for the range of required orbit inclinations and the resulting data should be integrated into the mission analyses.

FUTURE MISSION ANALYSIS IMPLICATIONS

In addition to the more rigorous interfacing of realistic Saturn payloads in future mission analyses, other factors associated with launch azimuth constraints must be considered. Since it will be necessary to impose weight penalties upon the vehicle in order to circumvent these constraints for certain modes and launch opportunities, any analyses leading to comparisons of these modes must involve an analysis of these constraints. First it must be determined if a nominal launch is possible and if not, the best method to be employed in circumventing the constraints and the attendant weight penalties must be ascertained.

In addition, in the analysis of launch window provisions, constraints placed on the start of the launch window may require the nominal date or center of the window to shift to a later depart date. Therefore, the analysis of penalties incurred for providing Earth launch window also interacts with the problem of launch azimuth constraints (declination restrictions) and the analyses cannot be disassociated.

In summary, in order that future analyses of the standard as well as new alternative manned mission modes be completely valid, the areas of Saturn payload capability, launch window provisions, and launch azimuth constraints must be considered simultaneously.

V. REFERENCE

1. Mission Oriented Advanced Nuclear System Parameters Study, Volume III, Parametric Mission Performance Data, TRW/STL, 8423-6007-RU000, March 1965.