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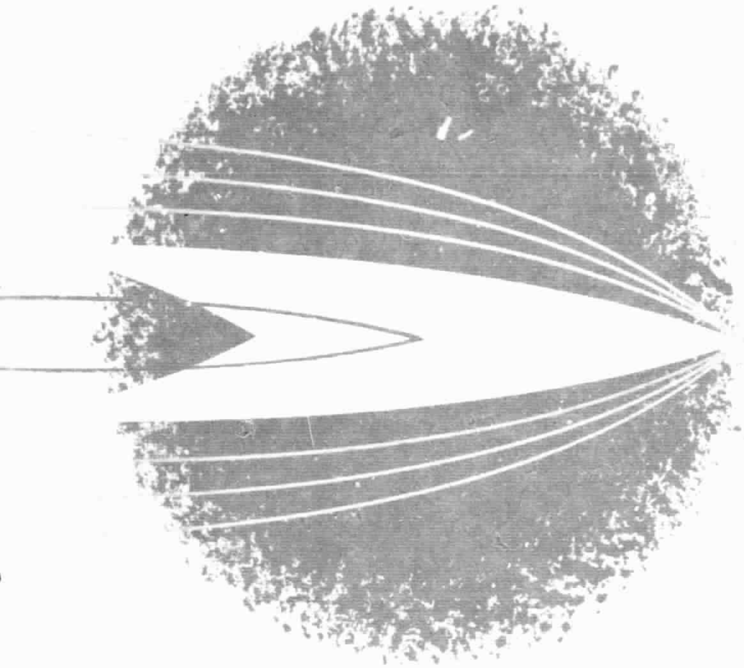

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INVESTIGATION OF SPACE RENDEZVOUS PROPULSION SYSTEM REQUIREMENTS

Vol. I - Summary
Contract NAS7-87

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INVESTIGATION OF SPACE RENDEZVOUS PROPULSION SYSTEM REQUIREMENTS

Contract NAS 7-87

Period Covered: 20 November 1961 to 19 December 1962

Approved by:

R.L. Phen

R.L. Phen
Project Engineer

SPACE-GENERAL CORPORATION

C.A. Lysdale

C.A. Lysdale, Manager
Space Systems Study Center

AEROJET-GENERAL CORPORATION
Azusa, California

Report No. 2555

CONTRACT FULFILLMENT STATEMENT

This final report, documenting all work under the contract, constitutes partial fulfillment of Contract NAS 7-87.

AEROJET-GENERAL CORPORATION
Approval:

M. L. Stary, Manager
Spacecraft Division

SPACE-GENERAL CORPORATION
Approval:

C. E. Roth, Jr., Vice President
Space Operations

FOREWORD

This summary document is the first of five volumes that present the work completed by Space-General Corporation, and the Spacecraft and Space Propulsion Divisions of Aerojet-General Corporation on the "Investigation of Space Rendezvous Propulsion System Requirements". Other volumes completing the report are Volume II - Phase Analyses, Volume III - Mission and Design Analyses, Volume IV - Tables and Figures, and Volume V - Appendices.

The study was conducted under National Aeronautics and Space Administration Contract NAS 7-87.

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CONTENTS (cont.)

I. INTRODUCTION

This is the first of five volumes presenting the results of contract NAS 7-87, "Investigation of Space Rendezvous Propulsion System Requirements". This volume summarizes the work presented in the other four volumes.

The study was divided into four basic areas of study: (1) literature review, (2) independent analysis of the rendezvous phases, (3) analysis of representative rendezvous missions, and (4) conceptual and preliminary design of rendezvous propulsion systems.

Prior to discussion of the above phases of the study, it is necessary to define rendezvous propulsion. Rendezvous propulsion has been defined as those propulsion systems, utilized in the rendezvous operation, which have the capability for completion of closed-loop vehicle closure and docking. In addition, the propulsion system may be required to satisfactorily perform other operations connected with the rendezvous maneuver. The rendezvous propulsion system is normally contained in the last stage of the rendezvous vehicle.

The initial effort under the study involved a comprehensive literature review; the resulting bibliography is presented in Appendix A. References on all phases of the rendezvous maneuver are included.

To gain a better understanding of the rendezvous maneuver, the maneuver was divided into phases, each of which was initially examined independently. The phases studies include ascent, orbit transfer, midcourse correction, closure, and docking. In addition, two areas of study, although not strictly rendezvous phases, were subjected to independent investigation. These involved an attitude control analysis and an analysis of engine transient and control effects. The individual phases are illustrated graphically in Figure 1 of this volume.

Three different missions, which were considered representative of the rendezvous missions to occur in the next 10-15 year period, were examined. These missions are: (1) earth-orbit rendezvous of Saturn C-5 payloads, (2) earth-orbit rendezvous with a space station, and (3) lunar-orbit rendezvous.

The primary purpose of examining the rendezvous phases and missions was to establish propulsion requirements such as characteristic velocity, thrust-to-mass ratio, thrust variability, number of restarts, thrust vector control requirements, etc.

The subsequent design studies define the best type of propulsion system and vehicle for completion of the rendezvous missions. Included in the propulsion design analysis are the selection of propellant combination, engine design and cooling method, feed system, tankage type and material, etc. Configuration studies were made to determine the best vehicle configuration, best number of engines, number of tanks, etc.

The design analysis was carried out in three basic steps: (1) conceptual design of the propulsion system to perform the earth-orbit rendezvous of Saturn C-5 payloads, (2) preliminary design of the propulsion system to perform the Saturn C-5 rendezvous mission and (3) mission variation effects on the propulsion system design.

II. PHASE ANALYSES

A. THE ASCENT PHASE

The ascent or launch phase was examined during the rendezvous study to determine the effect of the launch trajectory on the rendezvous propulsion requirements and payload. For the case in which no specific launch vehicle is stipulated, the ascent analysis would consist basically of establishing the approximate size of the final stage by distributing the characteristic velocity judiciously among the n stages of the vehicle, taking into consideration out-of-plane and launch-time delay requirements. The result consists of a table of multistage systems for a range of rendezvous mission parameters. It was recognized, however, that the design of the rendezvous system is actually constrained by existing launch vehicle designs and practical requirements on rendezvous orbit altitudes. Consequently, no attempt was made to optimize the staging of the over-all vehicle.

The ascent phase analysis does, however, show the variation in size of the rendezvous vehicle, and the final payload, with launch vehicle configuration and rendezvous altitude. Launch vehicle configurations have been selected as representative of the capabilities that will be available. They cover a rendezvous vehicle weight range of approximately 10,000 lb to 400,000 lb (based on a 100 n.m. initial orbit altitude). The data of the ascent analysis is presented largely in terms of the use of an intermediate orbit in the rendezvous flight profile, although approximate means are provided (Appendix B) for interpreting the data in terms of direct ascent.

Also included in the analysis is a comparison between the direct ascent and the parking orbit techniques. The parking orbit technique offers numerous advantages over the direct ascent and can, by staging in the intermediate orbit, provide a payload advantage over the direct ascent method.

Other factors considered in the ascent phase analysis are the effects of a launch delay, and the effects of orbit inclination and orbit altitude on the ascent phase velocity requirements. The velocity penalty associated with a launch delay has been developed. Factors affecting the selection of orbit altitude have been evaluated with emphasis on the utilization of rendezvous compatible orbits.

B. THE ORBIT TRANSFER PHASE

Orbit transfers for rendezvous maneuvers are unique in that the prime requirement of the transfer is to place the rendezvousing vehicle in the vicinity of the target with the initial conditions required for the closure phase. That is, in most transfers for rendezvous maneuvers, the final correction of a typical orbital transfer is replaced by the closure phase maneuver. The philosophy of reserving the final orbital correction for the closure phase is practiced throughout the analysis.

Based on the contemplated rendezvous missions, it is assumed that both the original orbit from which the vehicle is launched and the final target orbit will be circular. Small eccentricities in the actual orbits, resulting from the various errors, will exist; however, these eccentricities should be small enough to be ignored when determining the propulsion requirements for most transfer operations. Transfers between elliptical orbits and between circular and elliptical orbits are not contemplated for the friendly rendezvous missions currently projected during the next 10 years; thus these transfers are not considered. Therefore, only coplanar and non-coplanar transfers between circular orbits, and epoch changes in circular orbits, are analyzed. Hohmann transfers are assumed throughout for simplicity; non-Hohmann transfers will generally result in larger propulsion requirements than those given. All velocity requirements were determined on the basis of impulsive thrusting.

The results of the analysis can be stated briefly as follows. For the in-plane transfers the velocity requirements can range from 100 to 5000 ft/sec, but since most transfers will be to and from orbits below the Van Allen belt, a maximum of about 500 ft/sec can be expected. The desirable initial thrust-to-mass ratios will vary between 0.05 lbf/lbm and 1.0 lbf/lbm. The propulsion system will not be required to be restartable unless it is also used for the closure phase. Thrust vector control by the main engine may be required for high accelerations, but the vehicle attitude control system will generally be adequate for the low acceleration cases.

For transfers involving a plane change, the velocity requirements can be significantly larger than for the in-plane transfers. Requirements as large as 11,000 ft/sec are possible, but the plane changes required will generally be small giving a typical maximum requirement of about 1000 ft/sec. The desirable initial thrust-to-mass ratios are between 0.05 lbf/lbm and 1.0 lbf/lbm.

C. MIDCOURSE CORRECTION PHASE

The midcourse correction phase of the rendezvous maneuver is generally characterized by impulsive-type, intermittent corrections applied after the orbit transfer maneuver (or after burnout of a direct ascent) in order to provide an interception course prior to initiation of the closure phase. The total midcourse correction velocity requirement will be in the range from 100 to 200 ft/sec when limited ground tracking is used. A typical velocity requirement for the largest pulse is less than 70 ft/sec. The thrust-to-mass ratio will normally range between 0.02 and 0.1 lbf/lbm. The propulsion system must be restartable and thrust vector control can be provided by a capable attitude control system.

D. THE CLOSURE PHASE

The closure phase is the primary rendezvous phase. During this phase the interceptor vehicle is brought into close proximity with the target and the position and velocity of the two vehicles are matched. The closure phase is efined in this study as the phase of the rendezvous maneuver which is characterized by closed-loop propulsion system control through target lock-on. (It should be noted that the midcourse phase, previously examined in Section IV, is a special case of the closure phase involving interception only which, because of its importance, was examined separately.)

The purpose of the closure phase analysis is to establish propulsion requirements for the terminal or closure phase of the rendezvous maneuver. The analysis has been directed toward the review and evaluation of various closure techniques. The kinematics and dynamics of closure paths have been examined and the equations of motion applicable to this phase have been established. The equations were then programmed on both digital and analog computers and solved for a variety of closure thrust programs.

A number of terminal guidance systems, both manual and automatic, located in either the interceptor or target, have already been proposed for the satellite rendezvous problem[1]. The majority of these systems utilize proportional navigation with constant bearing guidance schemes. Therefore, these schemes and modifications of them proposed by Cicolani[2], Sears and Felleman[3], and Harrison[4] have been used in this study.

A number of different thrust programs using the above proposed guidance systems have been considered. These include:

1. Continuous thrust with variable thrust level.
2. Continuous thrust with limited thrust variability.
3. Frequency-modulated pulse thrust.
4. On-off thrust with constant thrust level.
5. On-off-constant thrust to maintain interception, followed by a continuous variable thrust program to obtain the final closure.

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[1] Appendix A, Bibliography

[2] Reference 1

[3] Reference 2

[4] Reference 3

6. A modification of (5) in which continuous-variable thrust is used to maintain interception.

7. A modification of (5) in which on-off constant thrust is used to obtain the final closure.

Propulsion requirements, including characteristic velocity, specific thrust, specific thrust variability, and burning time, have been established for the above thrust programs. A variety of initial conditions, such as different initiation criteria, orbit altitudes, and orbit transfer errors were used in the analysis.

In the final analysis the propulsion requirements for each of the above programs were examined in detail and the thrust programs and guidance schemes were compared. The results of the closure phase analysis are summarized in the following paragraphs.

The best initial position of the interceptor with respect to the target at initiation of closure phase thrust is ahead of the target when the interceptor is transferring from a lower altitude and behind the target when the interceptor transfers from a higher orbit. Adherence to these criteria assures that a 180° reversal of the main engine thrust will not be required.

The best aim point for apogee of the transfer ellipse was found to vary with the transfer errors. (This apogee position will, of course, never be achieved because closure will be initiated prior to it. Designation of the Hohmann transfer in terms of its apogee with respect to the target does provide a simple way of classifying the transfer trajectories, however.) When the transfer errors are very small, commensurate with extensive tracking, the aim point should be slightly below and behind the target (at apogee passage). As the errors are increased, the aim point moves to a position ahead of and above the target.

Comparison of the guidance schemes allows the following generalizations. The basic method proposed by Cicolani results in minimized propulsion requirements, (i.e., lower velocity increment and lower thrust variability) while the method proposed by Sears and Felleman is more practical to implement in a flight vehicle.

Comparison of the thrust programs resulted in the selection of a continuous thrust scheme with limited thrust variability. This thrust program was found to be ideally suited to transfers involving small transfer errors. When the transfer errors were large the best thrust program involved a continuous thrust mode preceded by a constant intermittent thrust interception or midcourse phase.

The criteria for thrust initiation depends upon both the transfer errors and the thrust program. Generally, for small errors the continuous-variable thrust program is best initiated on a range criterion (20,000 ft from the target). For larger errors, the interception phase should be initiated at minimum lead angle (the angle between the line-of-sight and relative velocity vectors) while the continuous thrust phase is again initiated at a specified range.

Rendezvous operations for low earth orbit altitudes were investigated thoroughly, defining the following general propulsion requirements. For the main closure engine, the ΔV will vary between 200 and 300 ft/sec depending upon the initial conditions of the maneuver. The initial thrust-to-mass ratio required is about 0.06 lbf/lbm, while a thrust variability of 3:1 is normally adequate. If an interception phase is required, there will be an increase in ΔV requirement of up to 150 ft/sec; the nominal thrust level for interception is 0.015 lbf/lbm.

E. THE DOCKING PHASE

The orbital rendezvous docking phase is defined as the final phase of the rendezvous maneuver beginning after the last correction of the closure phase and ending when the interceptor is attached to the target. (The term attached is used in the broadest sense and includes the station-keeping maneuver). To establish the propulsion requirements necessary to make any velocity changes required during the docking phase, the initial and final conditions of the docking phase must be defined. A study was made of four technological areas directly influencing these docking conditions. The four areas considered are guidance system characteristics, interceptor exhaust-plume effects on the target, impact dynamics and coupling techniques, and liquid propulsion system cutoff accuracies. Based on the study of the above four areas the following nominal initial and final conditions were established:

(1) Initial:

Range = 1500 to 100 ft (1000 ft nominal)

Relative velocity = -5 to -10 ft/sec

(2) Final

Range = 0 $\pm$ 2 to 3 ft

Relative velocity = 0 to -1 ft/sec

A representative docking technique was established in which the interceptor is held within a control cone while the distance between itself and the target is closed. This procedure is represented graphically in Figure 2. Using this technique the total velocity requirement for all axes is about 35 ft/sec maximum. The thrust-to-mass ratios for docking should be in the range 0.005 lbf/lbm to 0.02 lbf/lbm. For the docking system, throttleable engines are not required, but restartability is a definite requirement with the capability for pulse thrusting. Thrust vector control is provided by the attitude control system.

In addition to the nominal docking requirements, propulsion requirements to perform a station keeping maneuver were established. The total impulse to mass ratio depends upon the station keeping drift rate and amplitude of oscillation, but will generally be of the order of 0.1 $\frac{\text{lbf-sec}}{\text{lbm}}$ per hour of operation. The thrust to mass ratios required are in the range between 10^{-3} and 10^{-2} lbf/lbm.

F. ATTITUDE CONTROL

The specification of the attitude control system propulsion requirements is complicated by the dependence on vehicle size and shape information (i.e., moments of inertia) and by the interaction of control requirements and characteristics with the particular type of attitude control systems under consideration. It is therefore necessary to define (a) methods for evaluation of control requirements based upon the various types of attitude control systems, as well as (b) methods for comparison and selection of the most advantageous control system for a specific mission and vehicle.

In this report the competitive systems are defined and the comparisons are presented. System comparisons were made for three basic types of systems - reaction jets, thrust vector control (gimballed-engine) systems, and reaction wheels.

Three types of operational periods which may arise during a rendezvous maneuver were considered: correction of main engine thrust misalighment, attitude limit cycles, and control of initial rates.

Parametric weight curves were developed for the three types of systems for each type of operation. Weight comparisons of all feasible system cobminations indicated that an all-reaction-jet system is generally best. There may be times, however, when the use of thrust vector control during main engine thrusting may be desirable.

G. ENGINE TRANSIENT AND CONTROL ANALYSIS

A single degree-of-freedom rendezvous engine transient analysis was performed to determine the effects of engine thrust dynamics on terminal guidance, stability, and control in the rendezvous of space vehicles. Based on the analysis, it can be concluded that, for a friendly target, there are no serious stability and control problems created by variable-thrust dynamics.

III. MISSION ANALYSES

A. EARTH-ORBIT RENDEZVOUS OF SATURN C-5 PAYLOADS

The single rendezvous of Saturn C-5 payloads for subsequent lunar operations was examined in detail. During the course of the study of this mission, numerous methods for achieving the rendezvous with maximum simplicity and maximum accumulated payload were examined. Included in the evaluation were methods involving the accumulation of the payload by the assembly in space of two discrete parts, as well as methods involving the transfer of propellant to a partially empty orbit-launch vehicle. Rendezvousing discrete parts for this mission does not provide the maximum payload with the discrete peices which are available. These are assumed to be the S-IVB orbit-launch vehicle and the Apollo lunar capsule and service module. Instead, it appeared best from the standpoint of maximizing payload to select the propellant transfer method. The propellant transfer method is possibly more complex, but this disadvantage does not outweigh the advantage of substantially increased total payload available for the subsequent lunar operation.

The mission profile for this mission is based on the one selected by the George C. Marshall Space Flight Center and an independent investigation made to substantiate certain of the profile's characteristics. Included in the mission profile is the use of the parking orbit technique to obtain maximum launch flexibility. The mission is illustrated in Figure 3.

Using the selected mission profile, the propulsion requirements for the entire rendezvous mission have been established. Included are the propulsion requirements to correct orbit plane errors and perform the transfer from the parking orbit to the operational orbit, as well as the propulsion requirements for the closure maneuver, docking, and attitude control.

In order to establish these propulsion requirements, a series of independent analyses were performed. These include analysis of the transfer errors, selection of the best closure thrusting scheme, and selection of the closure phase initiation criteria.

Considering orbit transfer errors commensurate with extensive ground tracking of the target, the best closure phase thrusting scheme was found to be a continuous-thrust program with limited variability; initiation of the closure phase was best accomplished at a range of 20,000 ft.

Two different propulsion systems are required - a primary system to perform the plane change, orbit transfer, and closure maneuvers and a secondary system to perform the docking, attitude control, and settling jet functions. The velocity and thrust requirements for these systems are given in Table 1.

The total velocity requirement for the primary system including an addition of approximately 10% for contingency, is 700 ft/sec. The acceleration at maximum thrust is 2.0 ft/sec^2 and the maximum thrust variability is 3:1. Two restarts are required; thrust vector control is provided by the attitude control system.

The docking and attitude control system used constant intermittent thrust. The thrust levels required are 1000 lbf along each axis for docking and 250 lbf about each axis for attitude control.

B. SPACE STATION RENDEZVOUS

The requirements for earth-orbit rendezvous with a space station are very similar to those for the rendezvous of Saturn C-5 payloads in an earth-orbit.

The rendezvous altitude was assumed to be 300 n.m. to allow for a reasonable station lifetime with little radiation hazard from the Van Allen belts. To be as general as possible, the assumption was made that a minimum of ground tracking was available, so that a midcourse or interception phase was required. The characteristic velocity and specific thrust requirements for this mission are given in Table 2. The total velocity requirement is about 850 ft/sec for the main engine, about 200 ft/sec maximum for interception and docking, and about 50 rad/sec for attitude control. (The units rad/sec for attitude control are equivalent to the ΔV applied divided by the moment arm from the bodies C.G. to the engine location. Use of the units allows general application to any vehicle shape.) The main engine acceleration level varies less than 3:1 with a maximum value of 3.8 ft/sec^2, equivalent to a thrust to mass ratio of about 0.12 lbf/lbm. The docking and attitude control thrust to mass ratios are about 0.015 lbf/lbm and 0.001 lbf/lbm, respectively.

C. LUNAR-ORBIT RENDEZVOUS

The lunar-orbit rendezvous mission differs considerably from the earth-orbit missions. The primary difference is that the lunar-orbit rendezvous propulsion system must perform the ascent phase as well as an orbit transfer, midcourse or interception, closure, and docking phases. The requirement to perform an ascent phase greatly increases the complexity of the mission. Four different mission profiles are considered and propulsion requirements are determined. These mission profiles considered include;

1. Mission Profile A

The Lunar Excursion Module (L.E.M.) ascends directly to the 100 n.m. rendezvous orbit altitude. An interception phase precedes closure and the final docking phases.

2. Mission Profile AA

The L.E.M. is launched into a 10 n.m. parking orbit and subsequently transfers to the 100 n.m. orbit. Closure and docking follow the transfer maneuver.

3. Mission Profile B

The L.E.M. is launched directly into an orbit having a period equal to that of the target orbit. The orbit has an apogee of 194 n.m. and perigee of 10 n.m.; varying injection altitudes were investigated. Interception, closure and docking follow the ascent.

4. Mission Profile C

A modification of mission B in which ascent is made to equal period orbits of different eccentricities, with perigee as the injection point of each orbit.

Two propulsion systems are required for each of the missions; the functions to be performed vary depending upon the mission profile used. The propulsion requirements for each mission profile are given in Table 3.

Mission profiles A and B require either continuous or stepwise variability of the main engine with intermittent thrusting. Profile A has the lowest Δ V requirements. Profiles AA and C require no variability and only one restart of the main engine. Closure and docking are performed by a secondary intermittent thrusting propulsion system.

IV. PROPULSION SYSTEM DESIGN

A. CONCEPTUAL DESIGN FOR THE EARTH-ORBIT RENDEZVOUS OF SATURN C-5 PAYLOADS

1. Transfer and Closure Propulsion System

For purposes of comparison, the main propulsion system was divided into four elements: propellants, thrust chamber and nozzle, pressurization system, and propellant tankage.

The major criteria for the selection of the best systems were: weight and performance, reliability, practicability, and compatibility with configuration constraints. The propulsion system concepts selected for evaluation were based upon choices made among the following: propellant combination, type of thrust chamber, number of engines, direction of thrust (orthogonal or nonorthogonal), pressurization system, type of propellant tankage, number of propellant tanks, and engine location.

Cryogenic, earth-storable, and high-performance space-storable propellants were selected for comparison. It was concluded that the LO_2/LH_2 propellant combination was most suitable for initial e rth-orbit rendezvous propulsion systems in the C-5 payload range, with future effort being devoted to OF_2/B_2H_6 or alternate high-performance space-storable systems, as experience with these propellant combinations increases. The propulsion system using OF_2/B_2H_6 propellants showed a weight advantage of about 4% over the system using LO_2/LH_2, while the system using N_2O_4/Aerozine-50 was about 30% heavier than the LO_2/LH_2 system.

Four basic thrust chamber types were considered in the study; regeneratively-cooled deLaval, ablation-cooled deLaval, radiation-cooled deLaval and unconventional plug nozzle and expansion-deflection engines. The conventional engines were found to have no advantage over the conventional engines in the selected vehicle configuration. Ablatively cooled engines were found to be generally best, except at high chamber pressures where the regeneratively-cooled chambers have a definite weight advantage. Pyrolitic graphite radiation chambers were not found to be competitive because of the prohibitive weight attendant with the use of a required protective shield. Ablative chambers were therefore selected for the pressure-fed systems, whereas regeneratively-cooled chambers were selected for pump-fed systems.

A comparison between turbopump and pressure feed systems was made. The two systems were found to be comparable in terms of weight and reliability. A pressure-fed system with ablative-cooled chamber was selected based upon greater flexibility and growth potential to other propellants.

Various vehicle configurations and numbers of engines were compared with the selection of the configuration shown in Figure 4. The vehicle utilizes a toroidal LO_2 payload tank of segmented spherical construction. The single engine is mounted in the center of the toroidal tank.

2. Docking and Attitude Control Propulsion System

After consideration of numerous propellants and conceptual system designs the docking-attitude control system selected utilizes earth-storable propellants (N_2O_4/Aerozine-50) in a single set of tankage, with teflon bladders for positive expulsion. The system is pressurized with stored helium and ablative engines used throughout were found to be best.

B. PRELIMINARY DESIGN FOR THE EARTH-ORBIT RENDEZVOUS OF SATURN C-5 PAYLOADS

Under the preliminary design analysis, the vehicle was described in detail and a weight and structural analysis was performed. An inboard profile of the vehicle is illustrated in Figure 5. The primary and secondary propulsion systems, including the ablative chamber and nozzle, pressurization system, tankage, and insulation are described in Volume III. The chamber pressure and expansion ratio for the main engine optimized at 95 psia and 50:1, respectively. A multiple-pintle variable-area injection system was selected over other injection systems for the variable thrust primary system. The primary system's thrust chamber and nozzle are shown in Figure 6. The pressurization system includes a gas generator-heat exchanger which heats auxiliary hydrogen and helium for pressurization of the hydrogen and helium tanks, respectively. For the secondary system optimizations of the engines' expansion ratios and chamber pressure gave results of 40:1 and 100 psia, respectively. A number of engine configurations were compared with the selection of that shown schematically in Figure 7 and in detail in Figure 5. The complete propulsion system specification for the rendezvous of Saturn C-5 payloads in a low earth orbit are given in Table 4.

C. MISSION VARIATION CONSIDERATIONS

The propulsion requirements for two missions, in addition to the rendezvous of Saturn C-5 payloads in a low earth orbit, have been established. The requirements were established basically to determine how the nominal propulsion system specification for the Saturn C-5 mission would vary for other rendezvous missions contemplated for the next 10 to 15 years, and what effect these changes would have on the propulsion system design philosophy. Rendezvous with an earth-orbiting space station and lunar-orbit rendezvous were selected as missions typical of those which are scheduled for the prescribed time interval.

Analysis of the requirements of the space-station missions indicates that the propulsion system design requires little modification from that specified for the Saturn C-5 rendezvous mission, until the rendezvous vehicle payload exceeds about 400,000 lb. For larger payloads, a turbopump-fed system achieves a moderate weight advantage over the pressure-fed system, based on an analysis using LO_2/LH_2 propellants. This recommendation should be verified for each application, however, particularly if OF_2/B_2H_6-type propellants are selected. The selection of high-performance space-storable propellants is considered desirable since space station rendezvous missions will probably occur in a time period in which the technology for these propellants will be sufficiently advanced to allow their use. A possible additional requirement for a propulsion system performing space-station rendezvous might involve reuse. For such applications, ablative chambers would not be acceptable and regeneratively-cooled chambers or radiation-cooled chambers would be applied. The propulsion system specifications for the earth-orbit space station mission are given in Table 5.

The advanced-technology lunar-rendezvous propulsion system differs from the current LEM system concept primarily in that OF_2/B_2H_6-type propellants are also specified in order to take advantage of their high performance and space storability. In addition, the propulsion system differs from the Saturn C-5 rendezvous propulsion system in that the primary propulsion system performs the ascent and transfer phases while the constant-thrust secondary system performs both the closure and docking maneuvers. No thrust variability is required. Both primary and secondary systems are pressure fed and ablatively-cooled engines were specified in all cases. Table 6 is a summary of the propulsion system specifications for the lunar-orbit rendezvous mission.

V. CONCLUSIONS AND RECOMMENDATIONS

The following is a brief presentation of the conclusions arrived at during the course of the program and the recommendations for improving the space-rendezvous propulsion-system technology.

A. CONCLUSIONS

1. The closure and docking phases are unique to the rendezvous maneuver.

2. The characteristics of the rendezvous maneuvers are low velocity increment, low acceleration levels and extensive thrust control which can be reasonably limited in control range.

3. The closure phase propulsion system characteristics are flexible in that a variety of methods are workable; however, the following

generalizations can be made:

a. for large payloads a variable thrust single-engine for closure and on-off orthogonal thrust for docking is best.

b. for small payloads on-off control with positive expulsion is applicable for both maneuvers.

4. Significant payload improvement is possible through unusual vehicle design concepts.

B. RECOMMENDATIONS

1. System control interactions with the propulsion system suggests further analysis and demonstration in the following areas:

a. a verification that the variable-thrust system controllability during closure will not be a major problem by examination of the interaction of the attitude control system and the closure system.

b. an investigation to determine the variable-thrust system dynamics for the rendezvous system designs developed.

c. ground test demonstration of the rendezvous propulsion system feasibility.

2. The following additional mission analyses are recommended:

a. the determination of rendezvous propulsion requirements for an orbital "tug" required for space station assembly.

b. the determination of propulsion requirements for a translunar shuttle.

c. the determination of propulsion requirements for a recoverable and reusable system for space-station rendezvous.

3. More detailed examination of the design aspects of propulsion systems using OF_2/B_2H_6 propellants is required.

4. Further study of the applicability of pryolitic graphite chambers for rendezvous mission applications should be made.

5. Injector programs to reduce the maximum injector pressure drop and chamber programs to develop light-weight full-ablative chambers should be continued.

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TABLE 1

SUMMARY OF PERFORMANCE REQUIREMENTS
FOR THE SATURN C-5 EARTH-ORBIT RENDEZVOUS

Maneuver	Characteristic Velocity	Specific Thrust
Plane change	223.0 fps	2.0 fps^2
Transfer	180.0 fps	2.0 fps^2
Closure (maximum)	225 fps	Variable 2.2 fps^2 to 0.75 fps^2
Docking (small errors)	43 fps	0.145 fps^2
Attitude control	48 rad/sec	0.03 rad/sec^2

TABLE 2

SUMMARY OF THE PERFORMANCE REQUIREMENTS ASSOCIATED WITH THE SPACE STATION RENDEZVOUS

	Acceleration (fps^2)	Operating Time (sec)			Characteristic Velocity (fps)*		
		Main Syst.	Dock Syst.	ACS	Main Syst.	Dock Syst.	ACS
PLANE CHANGE	2.0 (avg.)	112	-	-	223	-	-
TRANSFER IMPULSE	2.0 (avg.)	145	-	-	291	-	-
CLOSURE							
Interception	.5 (max.)	-	293***	-	-	161	-
Rendezvous	3.8 to 2	138	-	-	338	-	-
DOCKING	.5 (max.)	-	40***	-	-	43	-
		-	12@		-	-	-
ATTITUDE CONTROL**	.03 (avg.)	-	-	310@@	-	-	48
TOTALS	-	395	333*** 12@	310@@	852	204	48

* For attitude control, the total ΔV is in rad/sec

** The requirement for control of thrust misalignment shall be assumed to be zero either because the engine is mounted at the c.g. or because the engine is gimballed.

*** Longitudinal

@ Lateral

@@ Axis

TABLE 3

SUMMARY OF MISSION PROFILES

	A	AA	B	C	Units
Ascent Phase					
Transfer Orbit Apogee Altitude	100	100	194	120	n.m.
Transfer Orbit Perigee Altitude	-453	-49.4	10	80	n.m.
Launch Window	343.2	1321	313.4	296.4	sec
Launch Time	260	-367.9	210.4	203.2	sec
Launch Opportunities per 24 Hour	13.3	∞	13.3	13.3	
Time from Launch to Orbit Injection	210.46	52000*	656	811.8	sec
Characteristic Velocity Increment	5203	6267	6284	6290	ft/sec
Thrust Level	6000	6000	6000	6000	lb
Closure Phase					
Aim Point Conditions					
Range	9.22	1.96	20.2	5.96	n.m.
Line-of-Sight Angle	180.28	182.7	280.0	285.C	deg
Relative Velocity	1031.5	207.	487	112.7	ft/sec
Lead Angle	.7869	2.74	11.81	14.59	deg
Initiation Opportunities	1	1.7**	13.3**	13.3**	
Time from Orbit Injection to Aim Point	820	1940	0.0	1520	sec
Characteristic Velocity Increment					
Interception Maneuver	159.0		15.19	172.9	
Rendezvous Maneuver	1032	220	481.15		ft/sec
Total	1199	220	497.34	172.9	ft/sec
Thrust Level					
Interception Maneuver	184.0	70	66.15	70	lb
Rendezvous Maneuver	5030 3465		1636 1562		lb
Total Time of Closure Phase	171.3	350.	288.49	331.3	sec
Weight at end of Closure Phase	5256	5228	5066	5230	lb
Complete Mission Parameter					
Total Characteristic Velocity	6402***	6487***	6781.***	6463***	ft/sec
Total Time to Rendezvous	1201.76	54290.*	944.5	2763	sec

* Includes maximum waiting time in parking orbit

** Per 24 hour period

*** Maximum values within error region

TABLE 4 PROPULSION SYSTEM SPECIFICATIONS FOR THE EARTH-ORBIT RENDEZVOUS OF SATURN C-5 PAYLOADS

	Transfer and Closure Propulsion System	Docking Propulsion System (Small Apogee Errors)	Attitude Control Propulsion System
CHARACTERISTIC VELOCITY (TOTAL), ft/sec	700	45	-
TOTAL IMPULSE, MAX. lbf-sec	5.2×10^6	3×10^5	6.1×10^5
THRUST: Thrust Level, lbf	15,000 (single engine)	1000 lbf total in each of six orthogonal directions	250 lbf min. about pitch yaw axes 500 lbf min. about roll axis
Max. Burning Time (per chamber) sec	500	35 (Lateral) 110 (longitudinal)	200 (Pitch and Yaw) 240 (Roll)
Type of Thrust Control	Limited Variable	Not Required	Not Required
Control Range	3:1	-	-
Accuracy of Thrust Programming Req.	Closed loop	-	-
Restart Requirements	2	7	1000 per engine
Starting Transients (Ignition/Response) Typical,[1] sec.	0.04/0.3	0.005/0.025	0.005/0.025
Cutoff-Impulse Tolerance, 3σ, lbf-sec	500	Required Value: 680 Expected Value: 15	Required Value: 33 Expected Value: 7
Startup-Impulse Tolerance,[2] lbf-sec	Not critical	Less than 40	Less than 20
Vector Control Requirement	-	-	-
Vector Control Implementation	Attitude control system	Attitude control system	-
PROPELLANTS:			
Composition, Mixture Ratio, O:F	LO_2/LH_2 (5:1)	N_2O_4/Aerozine-50 (2.1:1)	N_2O_4/Aerozine-50 (2.1:1)
Nominal Delivered Specific Impulse, lbf-sec/lbm	430	310	310
I_{sp} Uncertainty (Computed/delivered), lbf-sec/lbm	< 3/6	< 3/6	< 3/6
Operational Compatibility	Excellent	Good	Good
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:			
Zero 'g' Propellant Supply	Docking engines used as settling jets	Teflon bladders	Teflon bladders
Space Boil-off Losses	< 1 percent	None	None
Insulation	70 lb, 1.0 in thickness	None Anticipated	None Anticipated
Micrometeorite Protection[3]	No special protection req.	None Anticipated	None Anticipated
Ionizing Radiation Protection[4]	None required	None required	None required
Target and Payload Contamination	No problems anticipated	None Anticipated	None Anticipated
Ground Support Considerations	Cryogenic propellant handling	Hazardous propellant handling	Hazardous propellant handling
SYSTEM GEOMETRY AND CHARACTERISTICS:			
Configuration	7 (Figure 124)	(Figure 124)	(Figure 124)
Vehicle Envelope Restrictions	396 in diameter, max.	-	-
Propellant Feed System	Pressure	Pressure	Pressure
Thrust Chamber Cooling Method	Ablative	Ablative	Ablative
Thrust Chamber Characteristics	ϵ = 50:1, P_c 95 max, 30 min.	ϵ = 40, P_c = 100 psia	ϵ = 40, P_c = 100 psia
System Weight, lb	13,848	2704	-
Stage Propellant Fraction	0.898	-	-
RELIABILITY FACTOR:[5] (Total Propulsion System)	0.947	-	-

NOTES:
1. Assumes tanks pressurized prior to start
2. Guidance/control systems monitor delivered velocity increments in real time for all maneuvers.
3. Micrometeorite environment not well defined. Vehicle skin and tank wall provide a double wall and a high probability of no disabling damage.
4. Protection for the bladders should be provided by the tanks and vehicle skin.
5. Inherent system reliability factor based on current component failure rate data.

TABLE 5 PROPULSION SYSTEM SPECIFICATIONS FOR EARTH-ORBIT SPACE-STATION RENDEZVOUS

(400,000 lb initial weight in a 120 n.m. orbit)

	Transfer and Closure Propulsion System	Interception and Docking Propulsion System	Attitude Control Propulsion System
CHARACTERISTIC VELOCITY (TOTAL), ft/sec	930	200	-
TOTAL IMPULSE, lbf-sec	1.15×10^7	2.3×10^6	3.1×10^6
THRUST: Thrust Level, lbf	50,000 lb (single engine)	6000 lb in each of the six orthogonal directions	2000 about each axis[6]
Max. Burning Time (per chamber), sec.	400	330 (longitudinal) 12 (lateral)	310[6]
Type of Thrust Control	Limited Variable	Not Required	Not Required
Control Range	3:1	-	-
Accuracy of Thrust Programming Req.	Closed Loop	-	-
Restart Requirements	2	20 (longitudinal) 5 (lateral)	1000 per engine
Cut-off-Impulse Tolerance 3σ, lbf-sec	2250	160	40
Startup-Impulse Tolerance[2], lbf-sec	Not Critical	Less than 250	Less than 100
Vector Control Requirement	-	-	-
Vector Control Implementation	Attitude Control System	Attitude Control System	-
PROPELLANTS:			
Composition, Mixture Ratio, O:F	OF_2/B_2H_6 (3:1)	OF_2/B_2H_6 (3:1)	OF_2/B_2H_6 (3:1)
Nominal Delivered Specific Impulse, lbf-sec/lbm	435	435	435
Operational Compatibility	Fair[3]	Fair[3]	Fair[3]
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:			
Zero 'g' propellant supply	Docking System used as settling jets	Positive Expulsion bladders	Positive Expulsion bladders
Space Boil-Off Losses	Negligible	Negligible	Negligible
Insulation	Negligible	Negligible	Negligible
Micrometeorite Protection[4]	No Special Protection reqd.	No Special Protection reqd.	No Special Protection reqd.
Ionizing Radiation Protection[5]	None Required	None Required	None required
Target and Payload Contamination	No problems anticipated	No problems anticipated	No problems anticipated
Ground Support Considerations	Hazardous and cryogenic propellant handling	Hazardous and cryogenic propellant handling	Hazardous and cryogenic propellant handling
SYSTEM GEOMETRY AND CHARACTERISTICS			
Configuration Vehicle Envelope Restrictions	Depends upon the rendezvous functions and launch vehicles	Depends on the rendezvous functions	Depends on the rendezvous functions and vehicle shape
Propellant Feed System	Turbopump	Pressure	Pressure
Thrust Chamber Cooling Method	Regenerative	Radiation	Radiation
Thrust Chamber Characteristics (estimated)	$\epsilon = 60:1$, $P_c = 450$ max, 150 min	$\epsilon = 40:1$, $P_c = 100$ psia	$\epsilon = 40:1$, $P_c = 100$
System Weight, lb (estimated)	28,000	6100	8200
Stage Propellant Fraction	0.93	-	-

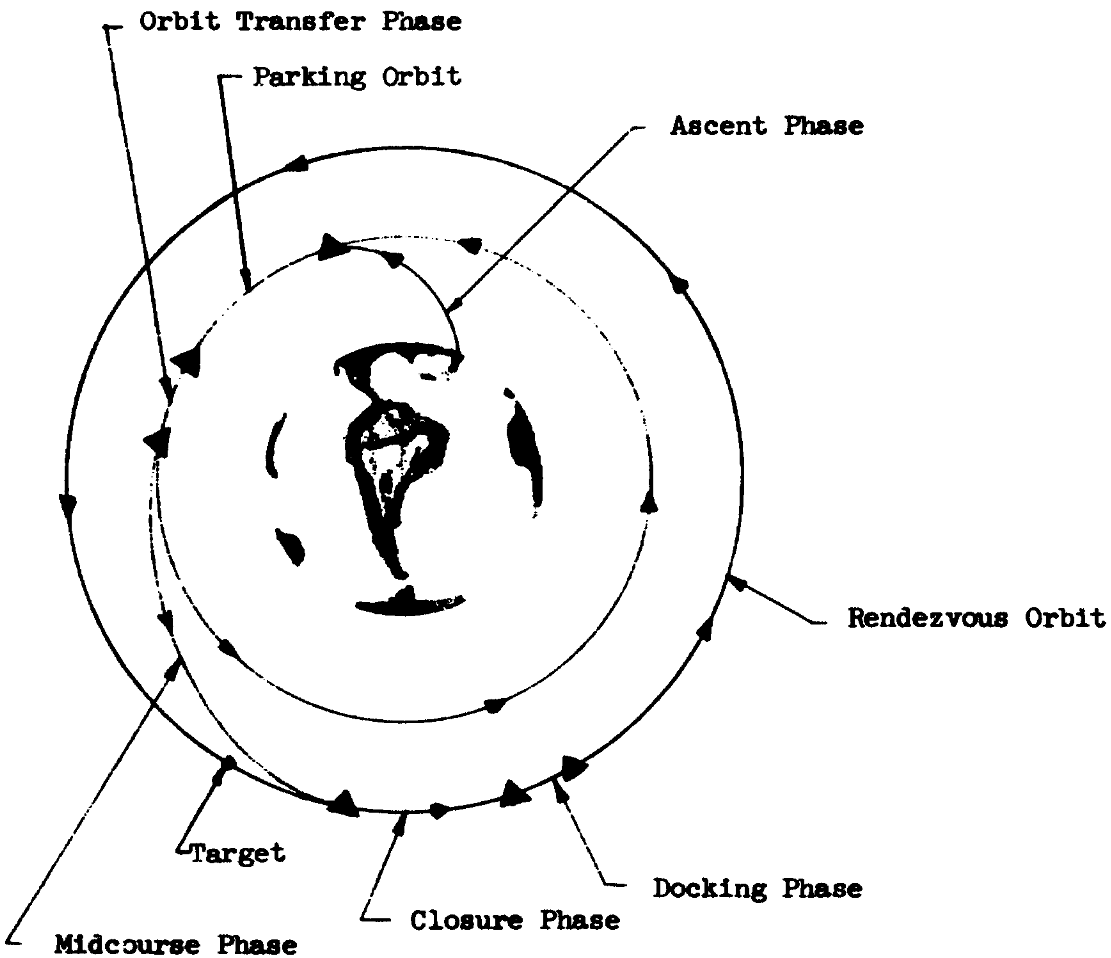
NOTES:
1. Assumes tanks pressurized prior to start.
2. Guidance/control systems monitor delivered velocity increments in real time for all maneuvers.
3. This judgement is based on the lack of present technological advancement.
4. Micrometeorite environment not well defined. Vehicle skin and tank wall provide a double wall and a high probability of no disabling damage.
5. Protection for the bladders should be provided by the tanks and vehicle skin.
6. This is an estimate; the actual value depends strongly on the vehicle configuration.

TABLE 6 PROPULSION SYSTEM SPECIFICATIONS FOR LUNAR-ORBIT RENDEZVOUS

(10,000 lb initial weight)

	Ascent Propulsion System	Closure and Docking Propulsion System	Attitude Control Propulsion System
CHARACTERISTIC VELOCITY, ft/sec	6300	265	-
TOTAL IMPULSE, lbf-sec	1.57×10^6	5.25×10^4	2.5×10^4
THRUST: Thrust Level	6000	70	50
Max. Burning Time (per chamber), sec	240	15 (lateral) 475 (longitudinal)	310
Type of Thrust Control	Not Required	Not Required	Not Required
Control Range	-	-	-
Accuracy of Thrust Programming Req.	-	-	-
Restart Requirements	1	20	1000 per engine
Cut-off-Impulse Tolerance, max/min, 3σ, lbf-sec	160	1.5	1.5
Startup-Impulse Tolerance[2], lbf-sec	Not critical	Less than 2.0	Less than 2.0
Vector Control Requirement	-	-	-
Vector Control Implementation	Attitude control	Attitude control	-
PROPELLANTS:			
Composition, Mixture Ratio, O:F	OF_2/B_2H_6 (3:1)	OF_2/B_2H_6 (3:1)	OF_2/B_2H_6 (3:1)
Nominal Delivered Specific Impulse, lbf-sec/lbm	435	435	435
Operational Compatibility	Fair[3]	Fair[3]	Fair[3]
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:			
Zero 'g' propellant supply	Positive expulsion bladders	Positive expulsion bladders	Positive expulsion bladders
Space Boil-Off losses	Negligible	Negligible	Negligible
Insulation	Negligible	Negligible	Negligible
Micrometeorite Protection[4]	No special protection reqd.	No special protection reqd.	No special protection reqd.
Ionizing Radiation Protection[5]	None required	None required	None required
Target and Payload Contamination	No problems anticipated	No problems anticipated	No problems anticipated
Ground Support Considerations	Hazardous and cryogenic propellant handling	Hazardous and cryogenic propellant handling	Hazardous and cryogenic propellant handling
SYSTEM GEOMETRY AND CHARACTERISTICS:			
Configuration	Not established	Not established	Not established
Vehicle Envelope Restrictions	Not established	Not established	Not established
Propellant Feed System	Pressure	Pressure	Pressure
Thrust Chamber Cooling method	Ablative	Ablative	Ablative
Thrust Chamber Characteristics (estimated)	$\epsilon = 50:1$, $P_c = 100$	$\epsilon = 40:1$, $P_c = 100$	$\epsilon = 40:1$, $P_c = 100$
System Weight, lb (estimated)	4000	200	100
Stage Propellant Fraction	0.90	-	-

NOTES:
1. Assumes tanks pressurized to start.
2. Guidance/control systems monitor delivered velocity increments in real time for all maneuvers.
3. This judgement is based on the lack of present technological advancement.
4. Micrometeorite environment not well defined. Vehicle skin and tank wall provide a double wall and a high probability of no disabling damage.
5. Protection for the bladders should be provided by the tanks and vehicle skin.

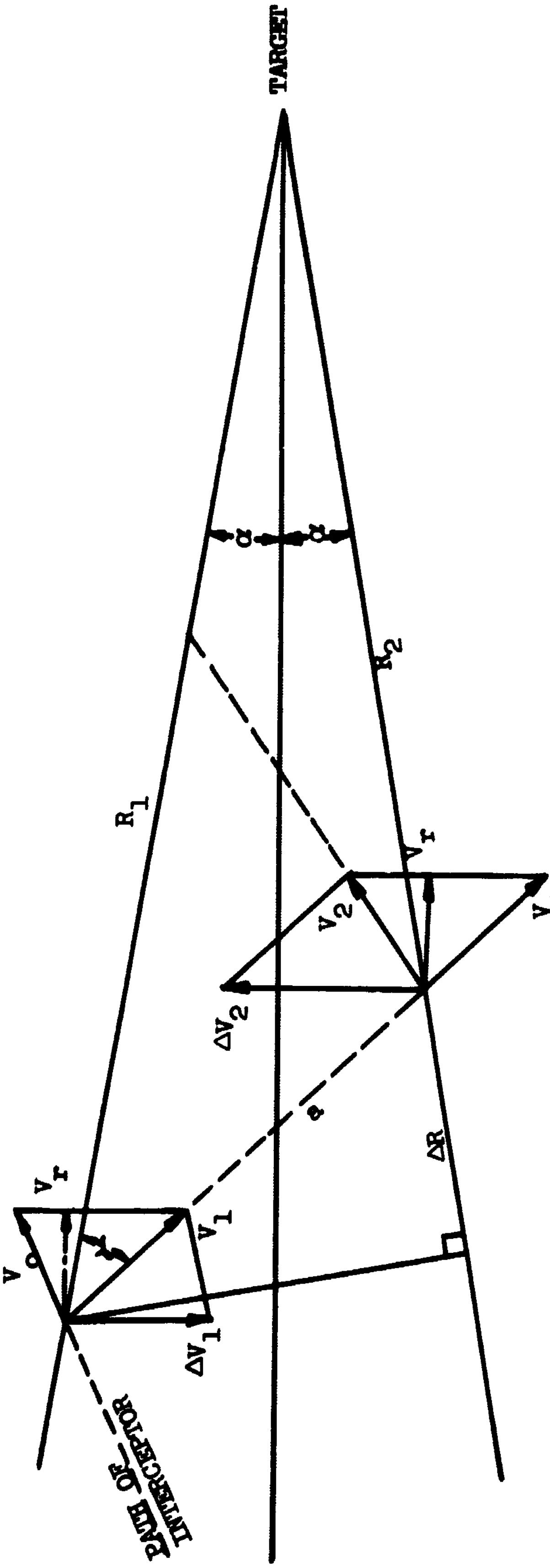


1. Boost & Ascent
2. Orbital Transfer
3. Midcourse
4. Closure
5. Docking

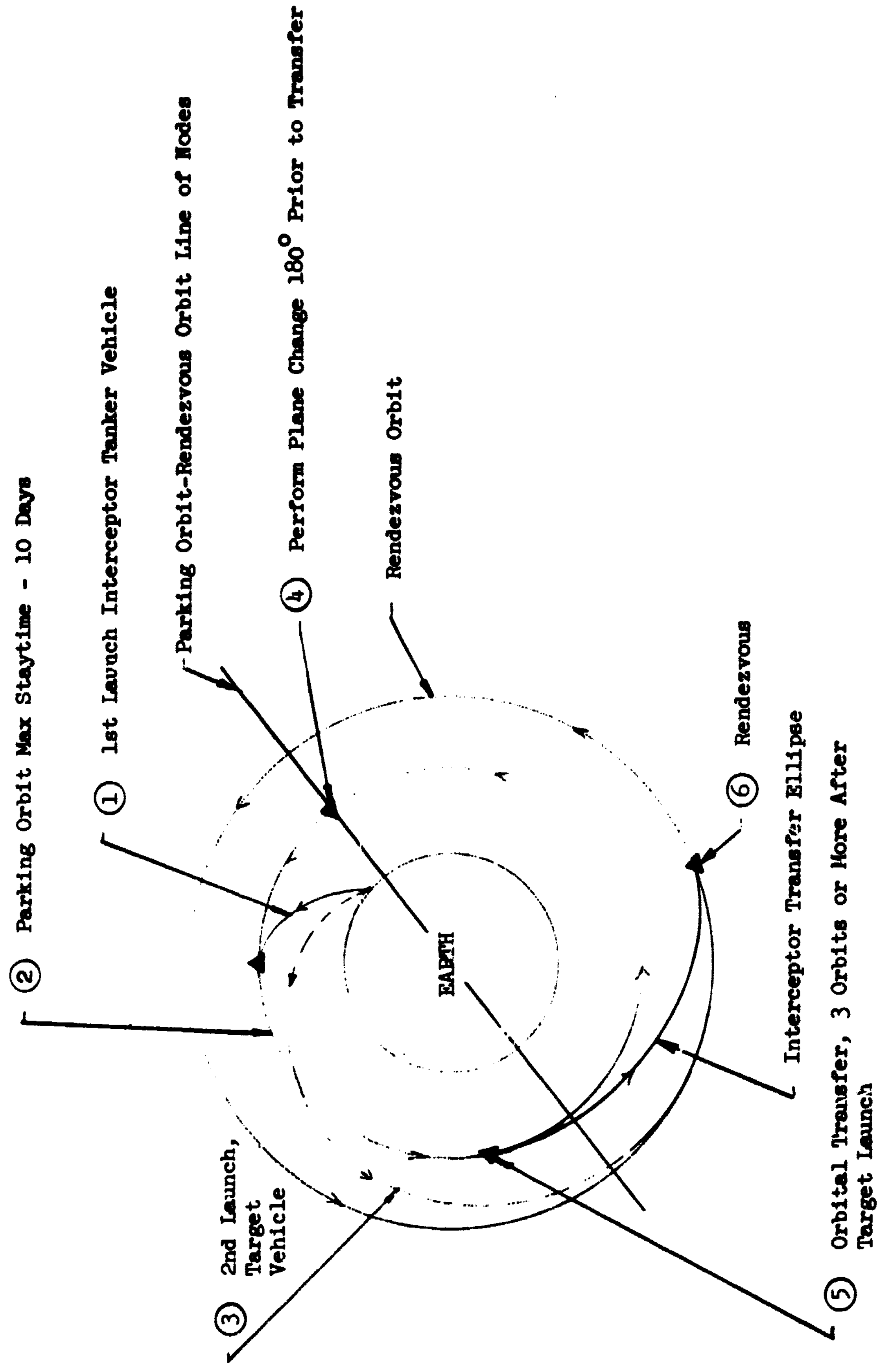
Rendezvous with Parking Orbit,
Phases 1 - 5

Direct Ascent Rendezvous,
Phases 1, 3, 4, & 5

PHASES OF THE RENDEZVOUS MANEUVER

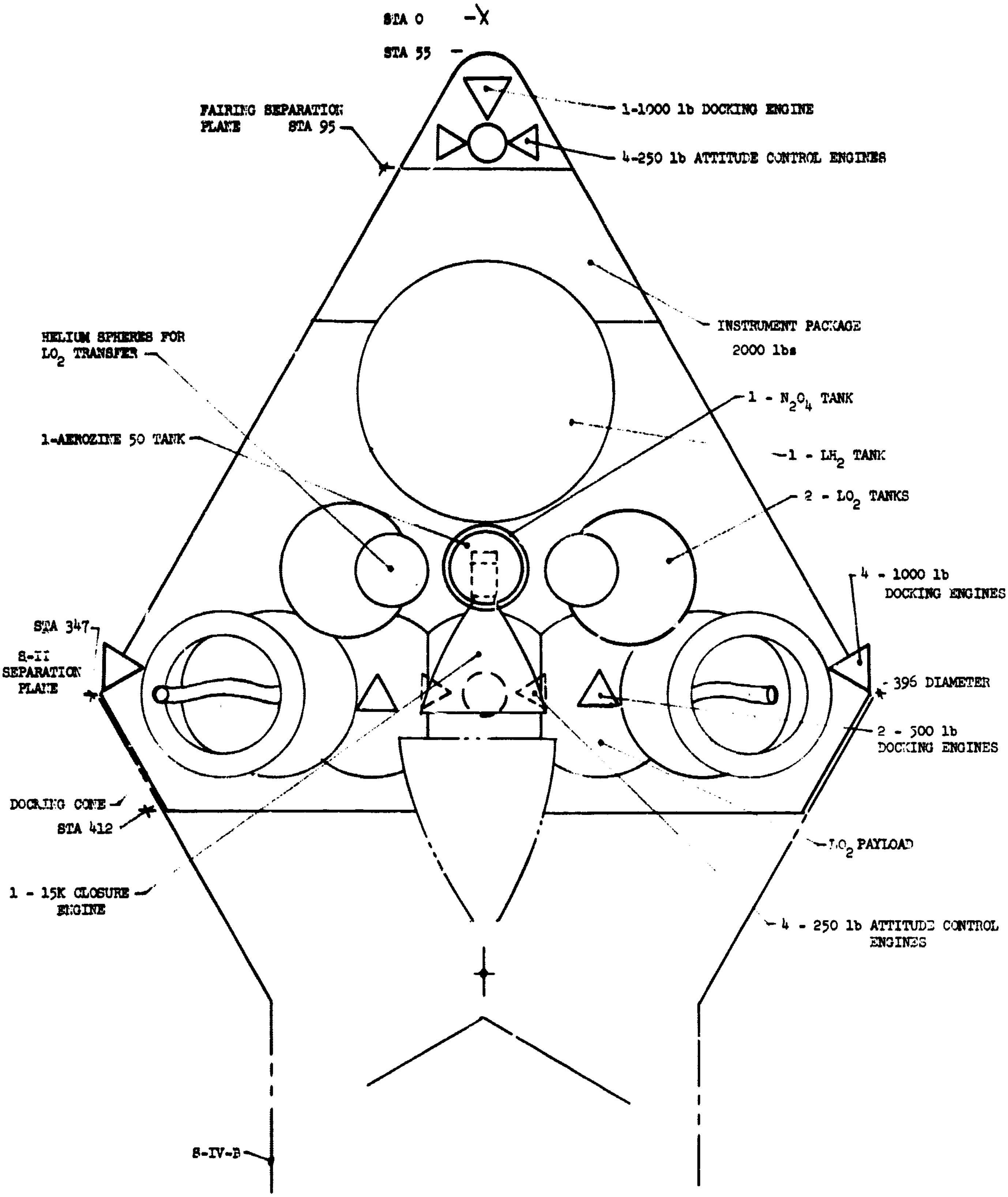


DOCKING CONTROL CONE GEOMETRY



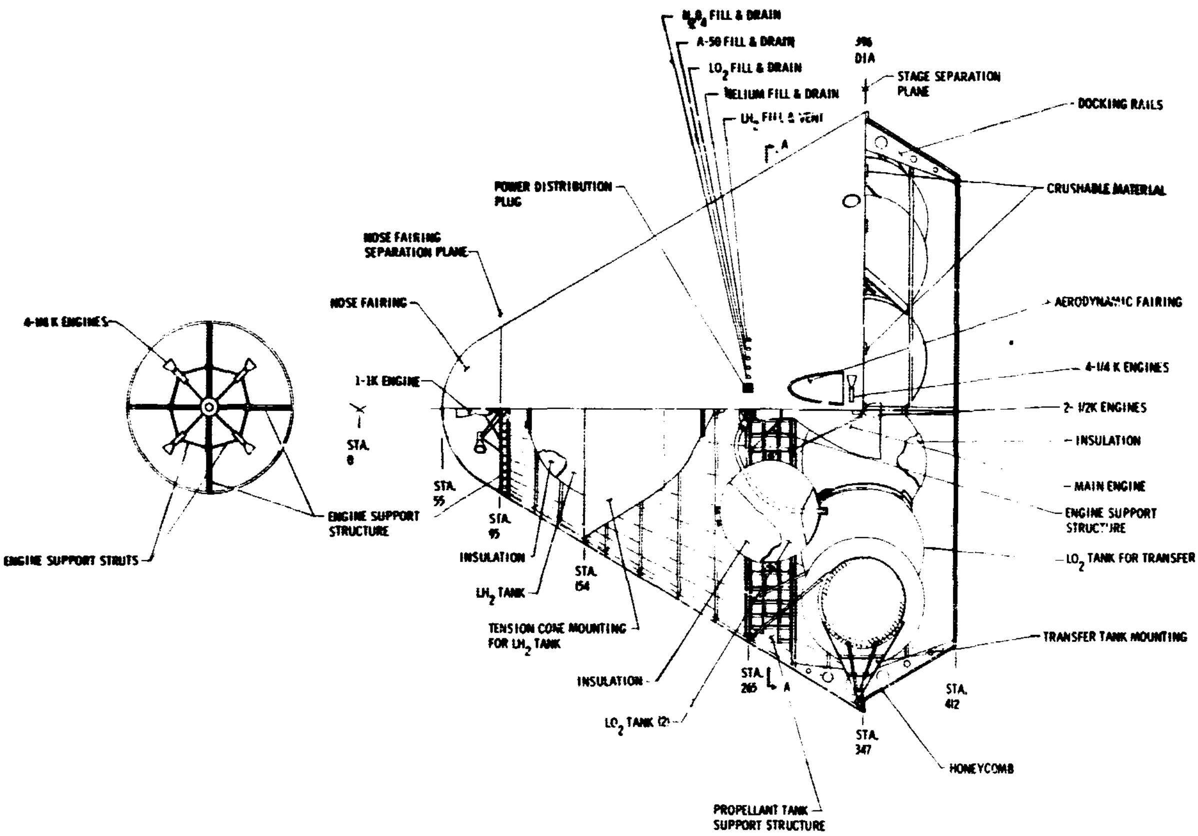
EARTH ORBIT RENDEZVOUS OF SATURN C-5 PAYLOADS

Figure 3
Volume I

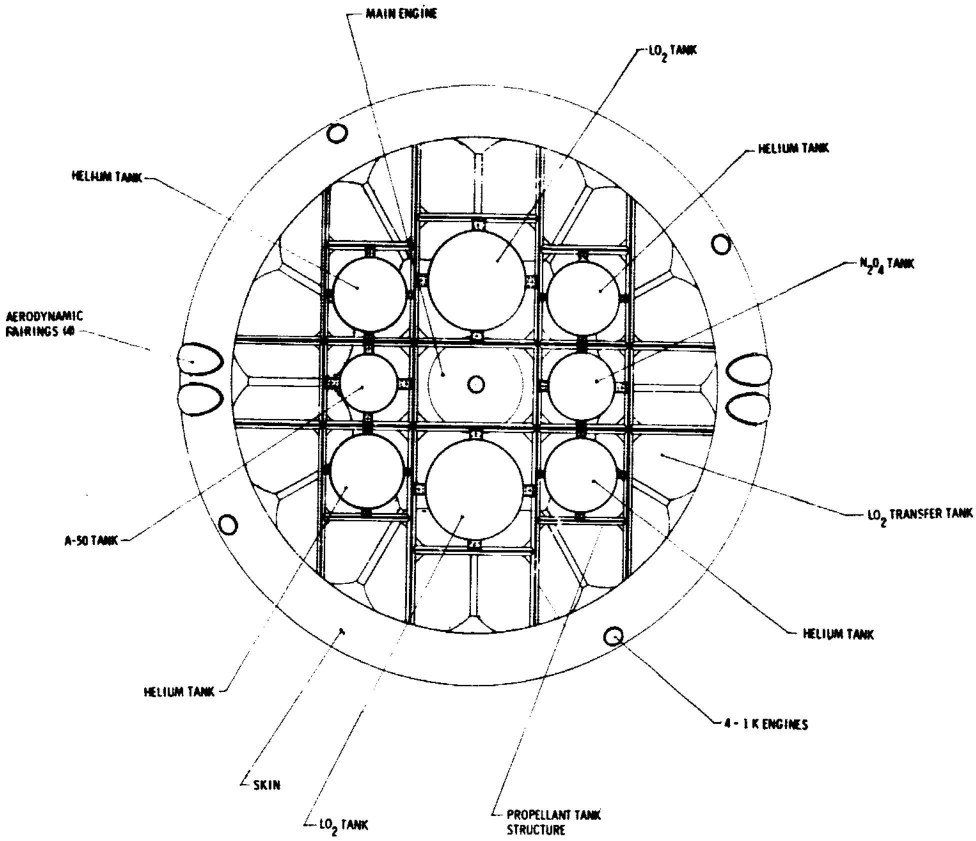


CONFIGURATION 6

Figure 4
Volume I

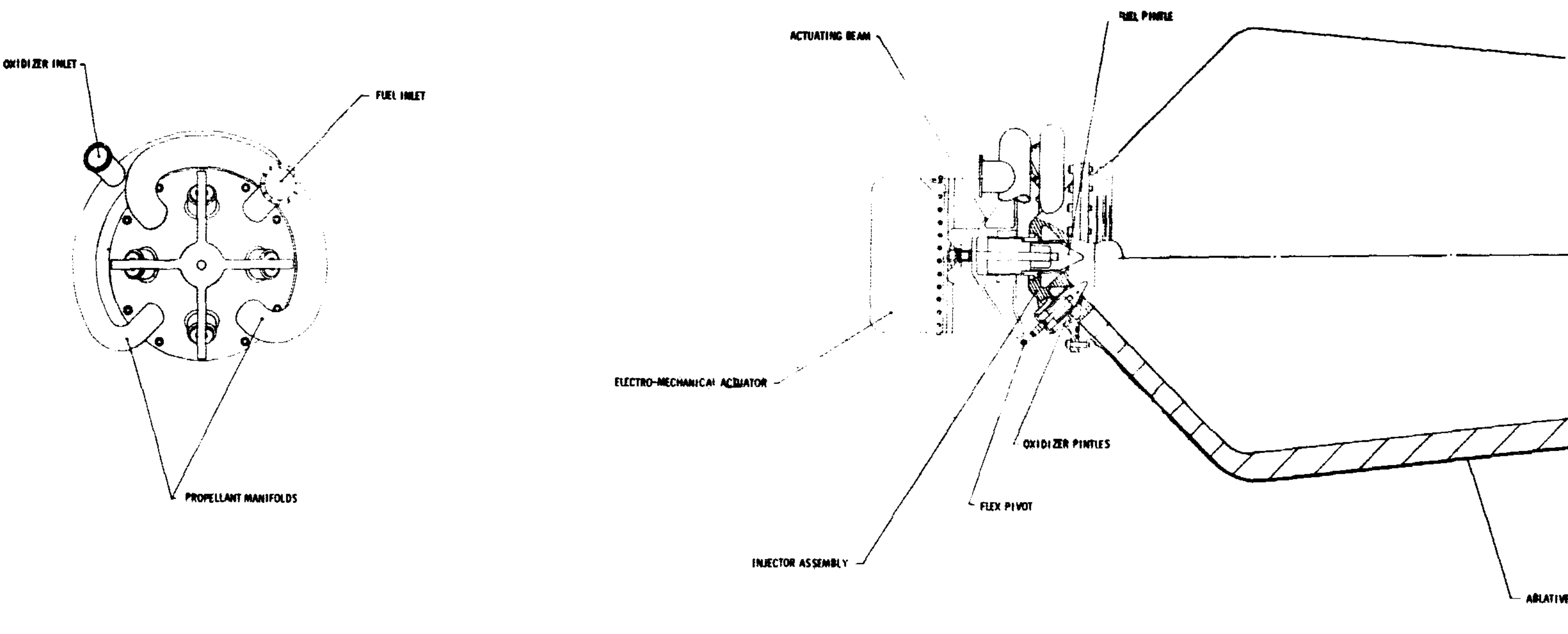


INBOARD PROFILE



SECTION A-A

Figure 5-2
Volume I



THRUST CHAMBER ASSEMBLY

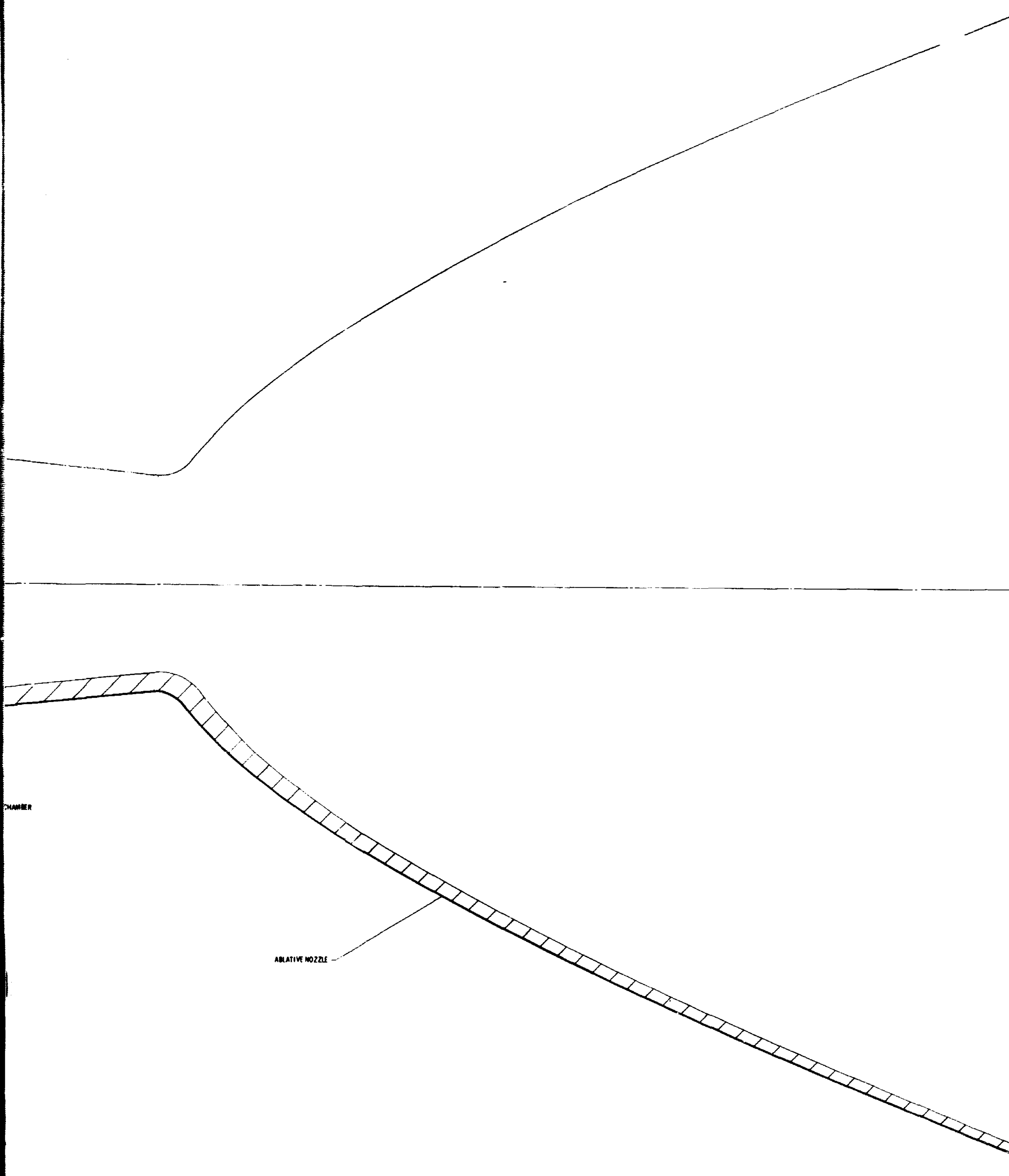
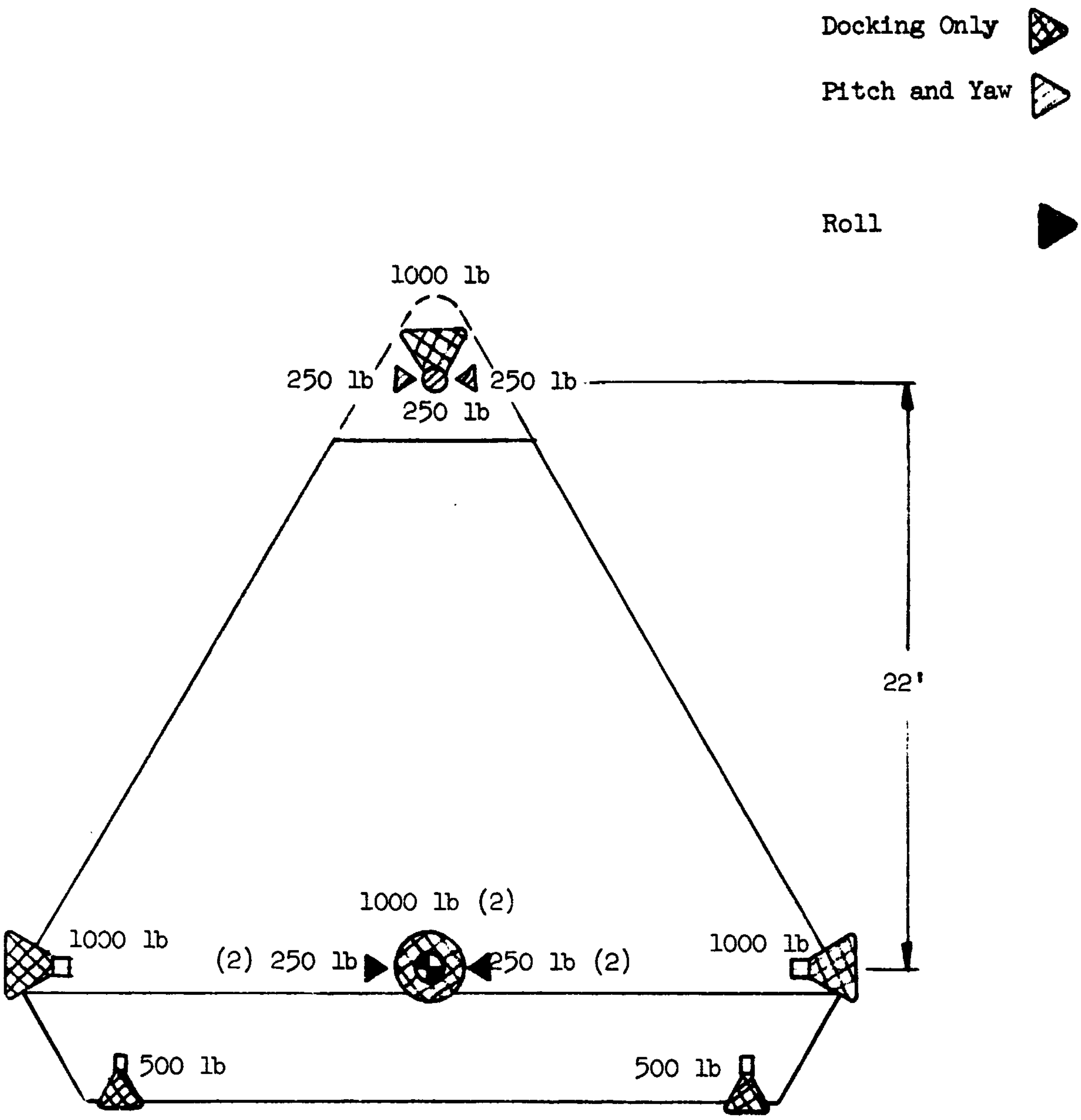


Figure 6-2
Volume I



DOCKING-ATTITUDE CONTROL SYSTEM THRUST CHAMBER LAYOUTS