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Page 42, first reference should read:
References 9 and 10, Volume III.

Page 42, second reference should read:
Reference 11, Volume III.

Since the distribution of Aerojet Final Report 2150 was completed, the above errata has been noticed.

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FOREWORD

This summary document is the first of six volumes that present the work completed by the Spacecraft Division of the Aerojet-General Corporation on the "Research Study to Determine Propulsion Requirements and Systems for Space Missions". This study, initiated on 1 February 1961 for the National Aeronautics and Space Administration, under Contract NAS 5-915, has two major objectives: first, establishment of the general propulsion requirements which are anticipated for future space missions, including earth-orbital, lunar, and interplanetary operations; and, second, optimization of system parameters and characterization of space-propulsion systems for several specific space missions. Technical efforts on the study were completed on 31 October 1961.

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I. INTRODUCTION

This volume summarizes the material presented in detail in Vols. IIa, IIb, III, IVa, and IVb.

The research study was organized in two phases. Phase I covered mission analysis, system concepts, and mission/system classification.

The work completed during Phase II of the study consists primarily of (a) evaluation of propulsion requirements and criteria; (b) selection and evaluation of alternate propulsion-system concepts; and (c) specification of selected integrated conceptual system design for each of four space missions which were specified for further study by NASA at the completion of Phase I. The four space missions include: (1) manned circumlunar missions, (2) manned lunar orbiting and return missions, (3) manned lunar landing and return missions, and (4) an unmanned 24-hr satellite mission. Injected spacecraft weights consistent with the capabilities of the Nova, Saturn C-3, Saturn C-2, and Centaur vehicles were considered.

The evaluation of propulsion requirements and criteria consisted primarily of reference to and extension of the generalized maneuver requirements as established in Phase I. This extension included: (a) selection and verification of appropriate three-dimensional nominal trajectories for each lunar mission; (b) verification of the propulsion requirements for trajectory corrections; and (c) further analysis of requirements and criteria for the specific maneuvers at the moon and for the 24-hr satellite operation.

The selection and evaluation of alternate propulsion concepts involved establishing a series of basic alternate propulsion-system concepts through the combination of available propulsion-system characteristics for the specified maneuver requirements. The alternate integrated systems were evaluated primarily on the basis of performance, with secondary consideration, including reliability, operational characteristics, and system flexibility.

The final part of the study consisted in detailed specification of the systems recommended for each mission. The accuracy of the performance and weight calculations was increased by a review of configuration and structure requirements. Optimizations were carried out and presented for the major propulsion-system parameters, and the operational sequence and the utilization of specified propulsion systems for the required maneuver was outlined. Tabular specification of system parameters and characteristics of the selected system is included.

II. MISSION ANALYSIS

The following categories of space missions and maneuvers are considered representative of the various space activities which are currently being undertaken, or will be initiated, in the foreseeable future: (a) orbital corrections, (b) orbital rendezvous, (c) lunar and interplanetary trajectory corrections, (d) lunar and planetary orbiting maneuvers, and (e) lunar and planetary landings and takeoffs.

Throughout the study the effort was made to obtain generalized coverage, rather than to consider specific applications, thus providing a comprehensive analysis of the possible variations and ramifications of the specified missions and maneuvers. The example considered in Appendix III-M, Volume III demonstrates the versatility available as a result of this approach.

Specific payloads and vehicle sizes were considered only in developing representative system characteristics for the mission/system classifications. Most parameters were presented on the basis of per-unit-initial-mass, in order to allow direct scaling of the results with vehicle gross weight. The analyses of orbital maneuvers were generally based on impulsive thrusting assumptions. A simplified non-optimum trajectory model was then used to establish upper limits on the effects of finite burning time. This model predicted excessive velocity penalties for some cases, and a more accurate finite-thrust orbit analysis was also considered. However, the simplified model was useful in guaranteeing the feasibility of straightforward, orbital-maneuver trajectories with only nominal finite-thrust penalties for many cases of interest.

The characteristics which were evaluated as basic mission-related propulsion requirements included: (a) ideal velocity-increment requirements, (b) desirable initial thrust to mass ratios, (c) required total impulse accuracy, (d) required thrust variability, (e) restart requirements, (f) thrust vector control characteristics, and (g) storability requirements.

The ideal velocity requirements are established directly from the nature and characteristics of the maneuver, while the required total-impulse accuracy is normally determined by the accuracy with which the maneuver must be completed. The desirable initial thrust-to-mass ratios were established from considerations including: (a) maximum acceleration tolerance of payload, (b) required cutoff impulse accuracy, (c) increase of propulsion system weight with thrust, (d) variations in ΔV requirement with thrust level, such as those due to gravity and drag losses, (e) effects of maneuver duration on guidance complexity, as for orbital maneuvers, and (f) effects of accelerometer bias errors on monitoring accuracy for the maneuver. The requirements for thrust variability, restart, thrust-vector control, and storability were established directly from the characteristics of the maneuvers.

A. ORBITAL CORRECTIONS

In determining the propulsion requirements for orbital corrections, most of the orbital maneuvers which it might be desirable to accomplish have been considered. These include: (1) control of orbital perturbation including atmospheric drag, earth oblateness effects, solar pressure and solar and planetary gravitational effects, (2) control of orbit eccentricity, (3) orbital plane changes, (4) orbital altitude variation and control, (5) orbital epoch changes, and (6) correction of injection errors.

1. Control of Orbital Perturbations

a. Atmospheric Drag

The most significant disturbance from the standpoint of absorption of orbital energy confronts a satellite that must pass over the earth at low altitudes. Observation and communication satellites, which may be required to pass within less than 200 n.mi. of the earth's surface, will become subject to significant and persistent aerodynamic-drag forces. The analysis of propulsion requirements to overcome atmospheric drag was concentrated on the range of practical orbits for observation and communication satellites. Circular orbits with altitudes as low as 60 n.mi. and elliptical orbits with perigee altitude of 100 n.mi. have been examined. Right-circular, aluminum-surfaced, cylindrical and spherical satellites were considered.

The work of Sanger, Tsien and Shamberg* was utilized in determining propulsion requirements necessary to counteract the effects of aerodynamic drag. It was determined that the total impulse required from a propulsion system is the product of the drag force and the duration of its action. The propulsion requirements to overcome atmospheric drag were summarized as follows:

Thrust levels will range from small fractions of a pound, for small satellites at high altitudes (200 n.mi. or higher), to 20 lb for 20-ft-dia satellites at low (60-n.mi.) altitudes. The total impulse required may range from less than 100 lb-sec/day, for small satellites at high altitudes, to over 1×10^6 lb-sec/day, for large satellites at low altitudes.

Objectionable changes in altitude may result from the fact that the applied thrust to overcome atmospheric drag does not equal the aerodynamic-disturbance force. In most cases, the required thrust-vector control can be accomplished by the attitude-control system, rather than the propulsion system itself performing the operation.

b. The Earth's Oblateness Effect

The earth's oblateness effects the motion of a satellite in a number of ways. D. G. King-Hele** examined orbits with eccentricities of 0.05 or less and determined that the four elements of the orbit which are effected are: (1) the period of revolution, (2) the rate of rotation of the orbital plane, (3) the rate of rotation of the major axis of the orbit, and (4) the oscillation in the radial distance. To correct all four conditions, it would be necessary to offset the increase in gravitational attraction, which is an increasing function with latitude, by a variable force applied to the satellite, in the radial direction. A continuous, variable thrust would be required to overcome the oblateness effect completely, which seems impractical. However, the effect of the earth's oblateness which seems most likely to require correction, is the rotation of the orbit plane.

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*References 2, 3, and 4, Volume II.
**Reference 5, Volume II.

The propulsion requirements to compensate for the rotation of the orbital plane due to the earth's oblateness were computed as follows:

Velocity requirements vary between values of less than 100 ft/sec/day, for high altitudes and large inclination angles, to about 4000 ft/sec/day, for a 100-n.mi. equatorial orbit. Initial thrust-to-mass ratios may range from approximately 0.05 to 1.0 lbf/lbm. The use of initial thrust-to-mass ratios in this range will result in realistic burning times for all cases.

c. Solar Radiation Pressure

Solar radiation pressure can cause the altitude of satellites with large surface-area-to-mass ratios to vary; but the corrections required to counteract this change would be small, and could be made with a system which combined attitude control and station keeping. Since attitude-control systems were not a subject for consideration in this study, no further determination of propulsion requirements for controlling solar-radiation pressure was made.

d. Satellite Perturbations Due to Lunar and Solar Gravities

The only effects of solar and lunar gravities which significantly change the motion of the satellite are the regression of the nodes and the oscillation of the orbit-inclination angle. Perturbations due to these effects can be controlled by a combined attitude-control and station-keeping system. The requirements of such a system therefore were not defined in this study.

2. Orbit Eccentricity Control

Operational requirements may make it necessary to change the eccentricity of satellite orbits so that a large spacial coverage can be obtained with one satellite. Propulsion requirements necessary to effect these changes were determined as follows:

The velocity requirements range from 100 ft/sec for very small changes in orbit eccentricity, to values of about 5000 ft/sec for large changes in eccentricity. Initial thrust-to-mass ratios may vary between 0.05 and 1.0 for most operations. The desirable upper limit will usually not exceed 1.0 lbf/lbm, so that burning times and burnout accelerations remain realistic.

Thrust modulation and restart capability will generally not be a requirement for a propulsion system required to change the orbit eccentricity. Thrust-vector control may be required for high accelerations, but the attitude-control system of the vehicle will generally be sufficient at low accelerations.

3. Orbital Plane Changes

Plane-changing maneuvers may be required to perform various functions required of an earth satellite, such as correction of regression of the modes due to the earth's oblateness, interception and rendezvous, and varying spacial coverage. An analysis was performed to determine the propulsion requirements to rotate the plane of an orbit through a given angle. Both circular and elliptic orbits were examined, and rotation angles to 45° for orbital altitudes between 300 n.mi. and 19,310 n.mi. were considered. The propulsion requirements determined by this analysis are summarized as follows:

The velocity increment required to change the orbital plane varies as a function of the altitude and rotation angle within the region considered. The required velocity increment could vary between 200 ft/sec (the ΔV required to rotate a very high altitude orbit 1°) to 19,000 ft/sec (the ΔV required to rotate a 300-n.mi. orbit 45°).

Initial thrust-to-mass ratios will range between about 0.5 and 2.0 at the 300-n.mi. orbit, and 0.15 to 1.0 at a 24-hr orbit for the maximum requirements. To accomplish very large plane changes at low altitudes by a single thrusting maneuver, an initial thrust-to-mass ratio of about 3.0 would be desirable; however, if the system has no variability, the final thrust-to-mass ratio may be as high as 15.0, which exceeds the allowable acceleration for manned vehicles. This indicates that either staging or a variability of about 2.0 is desirable for such maneuvers.

Thrust-vector control will probably be required for the larger plane changes; however, the attitude-control system will probably be adequate for the smaller plane changes.

4. Orbital Altitude Variation

Propulsion requirements to transfer from one circular orbit to another coplanar, circular, orbit of different altitude were determined for both impulsive and continuous-thrust assumptions. The propulsion requirements were established as follows:

The velocity increment requirement will be a few hundred ft/sec for small altitude changes. For large variations in altitude, the velocity increment requirement may be on the order of 14,000 ft/sec, if two impulsive thrusts are used, (one at perigee and one at apogee), or 19,000 ft/sec if continuous thrust is used.

Initial thrust-to-mass ratios for the maximum requirements will vary between 1.0 and 2.0 at the 300-n.mi. orbit and 0.1 and 2.0 at a 24-hr orbit.

Thrust modulation will generally not be required; different velocity requirements at perigee and apogee can be achieved by two different burning times.

Thrust variability or staging may be required to perform maneuvers with manned vehicles, at the maximum velocity requirements, in order to stay within the bounds of maximum thrust-to-mass ratio; i.e., about 8 lbf/lbm. For the largest requirement, thrust variability will be approximately 1.2 maximum.

A zero-g, restartable, propulsion system will be required to perform the perigee and apogee operations unless a continuous, low-thrust propulsion system is used.

Short-term storability will be required; however, if very low, continuous thrust is specified, long-term storability may be necessary.

Thrust-vector control may be required if the vehicle's attitude-control system does not have adequate capability at the higher thrust-to-mass ratios.

5. Orbital Epoch Change

a. Types of Maneuvers

Three types of maneuvers for achieving an epoch change were analyzed: (1) the use of continuous thrust, (2) impulsive transfer to a new path for a fast or emergency transfer, and (3) a special case of the fast transfer in which the satellite is required to achieve the epoch change in one orbital revolution.

b. Continuous Thrust

When an epoch change is made using continuous thrust, velocity is increased during the first half of the transfer, and decreased during the second half, or vice versa, depending on whether the epoch change is "leading" or "lagging". The original circular-orbit path is maintained during the maneuver by directing an appropriate thrust component along the radial axis. The radial thrust component is directed inward, when the velocity is greater than that for the normal circular orbit, and outward when the velocity is below orbital.

Generally, the continuous-thrust method appears most applicable when the epoch change to be made is very small, such as in terminal phases of a normal rendezvous maneuver.

c. Fast, or Emergency Transfer

Fast, or emergency transfers, require transfer by impulsive thrust to new trajectories. If the desired position "leads" the satellite, the new trajectory is either elliptical or hyperbolic, depending on the magnitude of the change required. If the desired position "lags" the satellite, the new trajectory is elliptical. The new trajectory intersects the original circular orbit in such manner that the satellite achieves the epoch transfer at the time of intersection. At this instant, a velocity increment, equal to that applied to transfer it to the new trajectory, returns the satellite to the original circular orbit.

d. Special Case of the Fast Transfer

The special case of the fast transfer was considered, in which the satellite is transferred by impulsive thrust to an elliptical path, in such manner that it takes one revolution of the satellite to reach its desired position in orbit. Cases where the desired position leads the satellite, and where the desired position lags the satellite, were considered.

Propulsion requirements to achieve an epoch change dictate two general types of systems. When the change to be made is small, the propulsion system required will generally be a low continuous-thrust system, and the velocity requirements will usually be less than 1000 ft/sec. The thrust must be variable in magnitude, with some means of controlling the thrust vector.

For general epoch changes in which the maneuver duration is not tightly restricted, the operational mode described as a "special case" can be used. This method will generally require velocity increments which are less than those necessary for continuous thrusting. For this type of system, the velocity increment will range from a few hundred ft/sec, for small epoch changes, to a maximum of about 8000 ft/sec for two impulses.

Initial thrust-to-mass ratio will range between 0.5 to 2.0 at the 300-n.mi. orbit, and 0.1 to 2.0 at the 24-hr orbit altitude. Thrust modulation will generally not be necessary unless this operation is part of a rendezvous maneuver, in which case modulation will be required for the terminal correction. Since two thrusting operations are necessary, the propulsion system must have a zero-g restart capability with short-term storability.

The propulsion requirements described above hold true for the fast-transfer method, with the exception that total velocity increment may be as high as 20,000 ft/sec for very fast transfer times, in which case the initial thrust-to-mass ratios will vary from about 1.0 at the 300-n.mi. altitude, to between 0.1 and 1.0 at the 24-hr orbit altitude. At the very high requirements, it may be necessary to have thrust variability, or staging, to maintain acceptable accelerations for manned vehicles. For the maximum ΔV requirements with a manned vehicle, the required variability could be as large as 2.0. Also, the propulsion system will probably have to provide for thrust-vector control, since the attitude-control system will normally not be sufficient at the higher accelerations.

6. Correction of Injection Errors

Two methods which could be used to correct injection errors were analyzed. The first method consists of correction of the errors in each orbital parameter separately; this is termed the three-impulse transfer. In the second method, the errors are corrected simultaneously by one maneuver; this result can be achieved by selecting a point in the desired orbit and then utilizing continuous thrust to attain that position.

In the three-impulse transfer method, errors in eccentricity and perigee altitude can be corrected simultaneously, and errors in the orientation of the orbit plane can be corrected by an additional impulse, applied at the proper location. The propulsion requirements to correct anticipated nominal injection errors, using the three-impulse transfer, are velocity increments ranging from a few ft/sec to approximately 1000 ft/sec, and initial thrust-to-mass ratios ranging from 0.1 to 0.5 at low orbital altitudes, and 0.01 to 0.5 at the 24-hr-orbit altitude.

If the continuous-thrust method is used, the propulsion system will require continuously modulated thrust. This method actually amounts to the rendezvous technique. The general requirements for rendezvous-propulsion systems are discussed in the following section.

B. ORBITAL RENDEZVOUS

Two basically different types of rendezvous were analyzed. The first assumes that rendezvous is composed of two operations, a coarse injection maneuver and a fine correction of the injection errors. The second type employs a homing technique, in which the rendezvousing satellite homes onto the target satellite. The rendezvous is accomplished simultaneously with injection. Besides these two basically different types of rendezvous, specific rendezvous problems were analyzed; particularly, rendezvous with the "dogleg" maneuver and the emergency rendezvous.

1. Rendezvous with Nominal Injection Errors

In this analysis, it was assumed that out-of-plane errors are small compared to in-plane errors, resulting in a two-dimensional rendezvous. Two cases were analyzed, (a) continuous thrust was assumed, and the results are descriptions of actual trajectories; and (b) it was assumed that the rendezvousing and target satellites had rectilinear motion, and that impulsive thrust is used to obtain rendezvous.

Two rendezvous problems were included in the analysis of continuous-thrust rendezvous. In the first problem, it was assumed that the two satellites are in the same orbit, while in the second problem they are not. The propulsion requirements necessary when the satellites are in the same orbit were determined in connection with the orbital epoch charge. The general propulsion requirements for the terminal phase of the rendezvous maneuver are as follows:

a. Velocity requirements will range from a few ft/sec to as high as 1000 ft/sec.

b. Maximum accelerations should not exceed 25 to 50 ft/sec^2, with the minimum values as low as 1 or 2 ft/sec^2 being desirable. This results in an initial thrust-to-mass ratio range of approximately 0.01 to 1.5.

c. Thrust variability will generally be required, with a maximum variability of the order of 100 required between the initial closure thrust and the terminal maneuver.

d. Restart capability and short-term storability will be necessary. Thrust-vector control can probably be accomplished with an attitude control system.

2. Rendezvous by Combined Terminal Guidance and Injection

The technique used in this method combines orbital injection with terminal-homing guidance in a two-dimensional field. A trajectory analysis was based on a satellite interceptor rendezvousing with a satellite in a circular orbit. A specific case was examined in which the rendezvousing satellite injects from a parking orbit, and then homes on the target satellite at apogee.

The general propulsion requirements for rendezvous by combined terminal guidance and injection are as follows:

a. Velocity increments will range from 500 ft/sec to 5000 ft/sec.

b. Initial thrust-to-mass ratio will range between a maximum on the order of 1.0 to 3.0 and a minimum of 0.01. This wide range indicates that variability will be required. Variability can be accomplished by a very accurate injection, using a large primary-propulsion system and a separate, small, station-keeping system for the terminal phase of the rendezvous.

c. Thrust-vector control other than that supplied by an attitude-control system may be required for high accelerations. Short-term storability may be necessary; restart capability will often be required to augment thrust-variability control.

3. Rendezvous with the Dogleg Maneuver

Rendezvous with the dogleg maneuver will generally be comprised of two separate maneuvers, the first to make the orbital plane-change and injection, and the second to make the final rendezvous. Since requirements for the final rendezvous were determined previously, only requirements for the first maneuver were considered in the analysis.

Propulsion requirements for the coarse rendezvous correction are as follows:

a. Velocity requirements will range from approximately 500 ft/sec to 18,000 ft/sec depending upon the injection velocity and the dogleg angle required. The initial thrust-to-mass ratio is determined by the velocity requirement and the allowable burning time, and can vary within a range between 0.01 and 3.0, depending on the specific operation. Thrust variability may be required for the largest corrections, with maximum variabilities of the order of 2.0 required.

b. Restart capabilities for the fine maneuver will generally be required, and thrust-vector control, other than that supplied by the attitude-control system, may be necessary with the larger accelerations. The total system variability required to perform a dogleg rendezvous maneuver may be as high as 1000:1.

4. Emergency Rendezvous

An emergency rendezvous operation may include any or all of the following operations: (a) injection with the dogleg maneuver, (b) orbital epoch change, and (c) final rendezvous. Thus, velocity requirements for emergency rendezvous are the sum of those for the individual maneuvers. Propulsion requirements to perform an emergency rendezvous may be summarized as follows:

The velocity requirements can range from 1000 ft/sec to 25,000 ft/sec, with the maximum velocity needed for a rendezvous with dogleg and fast epoch transfer.

The initial thrust-to-mass ratios will vary widely depending upon the requirements. For the coarse maneuver, it may range between 1.0 and 3.0, with thrust variability as high as 3:1 needed for the extreme requirements. The thrust-to-mass ratio for the terminal rendezvous operation will vary between 0.01 and 1.5. The required overall system variability may be as high as 1000:1. Restart capability will be required for fast epoch changes, and thrust-vector control, other than the attitude-control system, will be required due to the high accelerations. Short-term storability will be required for almost all emergency rendezvous maneuvers.

C. LUNAR AND INTERPLANETARY TRAJECTORY CORRECTIONS

The propulsion requirements necessary to perform lunar and interplanetary trajectory corrections were established by error analyses for the nominal trajectories and particular missions considered. In the analysis of trajectory corrections, both midcourse corrections and terminal corrections were considered for the following missions: (1) midcourse corrections for earth-moon flights and earth-Mars flights, and (2) terminal corrections for outbound lunar flights, and return flights from the moon and Mars.

1. Midcourse Corrections

Midcourse correction capability for space missions will normally be required on ballistic flights, where the uncorrected trajectory results in miss-distances which are excessively large for terminal-phase correction. The propulsion requirements for midcourse corrections are affected by the following factors: (a) the initial burnout-velocity-vector accuracy, (b) the allowable miss distance at the target body, (c) the accuracy of midcourse navigation and guidance equipment, (d) the accuracy with which the corrective maneuvers are carried out, and (e) any significant inaccuracies in astro-physical data.

a. Earth-Moon Flights

Three outbound lunar flights of 1.50, 2.00, and 2.75 days duration were considered as representative of current and probable future lunar missions. It was assumed that the required midcourse correction would be established from earth-based radar tracking data.

The propulsion requirements determined for midcourse correction on earth-moon flights may be summarized as follows:

(1) Corrective velocity-increment capabilities between 25 and 250 ft/sec are required, with specific requirements primarily dependent upon the tracking system and initial burnout accuracies.

(2) The upper limit on thrust level for accelerometer-monitored midcourse propulsion is established either by payload acceleration tolerance (8-g maximum for manned vehicles and 20 g for unmanned payloads was assumed), or by the requirement for 0.1% cutoff accuracy in delivered impulse. The lower limit is established by the maximum burning time per correction of about 5 minutes, based on typical accelerometer bias errors.

(3) No requirement for thrust modulation is apparent for lunar flights. Total-impulse control must be accurate to within approximately 0.3%. Thrust-vector control will be required to maintain appropriate vehicle orientation during the correction, unless this function is provided by an auxiliary attitude-control system. Requirements for several restarts under zero-g conditions are definitely indicated, although a single accurate correction will be adequate for early flights. Storage durations, and times between restart, on the order of fractions of a day to several days, are indicated.

b. Earth-Mars Flights

Three missions to Mars were selected as being representative of outbound interplanetary flights. Examination of the parameters and method of analysis indicates that the midcourse correction requirements on flights to Venus and return will be similar, since the navigation accuracies and initial guidance errors are the predominant effects. Thus, the results presented in this portion of the study may be considered generally representative for midcourse corrections on the interplanetary flights which are considered of current major interest. Earth-Mars flight times of 100, 150, and 259 days were selected, and the trajectory parameters were chosen to result in the minimum total velocity-increment between parking orbits about the terminal planets. These trajectories require initial burnout velocities at earth of 45,400, 39,750, and 37,050 fps, respectively.

The primary navigation and guidance system which was considered for the midcourse phase includes (1) an optical system, relying on sun and planet bearings for position and velocity data, with stellar references for orientation information, and (2) a basic inertial guidance system for controlling propulsion during the actual midcourse corrections.

The propulsion requirements for midcourse correction on outbound Mars flights are summarized as follows:

(1) Corrective velocity increment capabilities between 50 and 1000 ft/sec are required, with specific requirements dependent upon the accuracy of the navigation system.

(2) The upper limit on thrust level is established either by payload acceleration tolerance, or by the requirements for 0.3% cutoff accuracy in delivered total impulse. The lower limit is established by the maximum burning time per correction of about 25 minutes, based on typical accelerometer bias errors.

(3) No requirement for thrust modulation is apparent. Total-impulse control must be accurate to within approximately 0.3%. Thrust-vector control will be required to maintain appropriate vehicle orientation during the correction, unless this function is provided by an auxiliary attitude-control system. Multiple restarts (up to 5 or 6) are indicated. Storage requirements, and times between restart for the midcourse-propulsion system will be on the order of 2 to 200 days.

2. Terminal Corrections

Terminal trajectory corrections are classed as the impulses applied to correct the final perigee distance after the target body's gravitational effects have become predominant. The nature of terminal trajectory corrections differs from that of midcourse corrections for two basic reasons: (a) the trajectory is being affected by the target body's gravitational field, and (b) position errors become increasingly important in determining the final perigee distance as the target is approached. Propulsion requirements for terminal corrections on outbound lunar flights and outbound Mars flights were considered first.

a. Outbound Lunar Flights

The ideal velocity increments required for terminal corrections on outbound lunar flights of 1.5, 2.0, and 2.75 days duration were obtained as a function of the initial miss distance. The terminal-navigation system considered consists of a vehicle-borne optical tracker, combined with a basic inertial guidance system to monitor the correction.

The propulsion requirements for terminal corrections on outbound lunar flights can be characterized as follows:

(1) ΔV: Ideal corrective velocity increments of between 25 and 500 ft/sec are indicated, with the specific requirement dependent primarily on the initial miss-distance and the accuracy of the terminal guidance.

(2) Thrust level: The upper limit on thrust level will be established either by the payload acceleration tolerance, or by the typical cutoff accuracy requirement of 0.5% on delivered total impulse. The lower limit on thrust level will result from a maximum burning time restriction of about 10 minutes, due to the effects of distance traveled during burning. For a representative maximum corrective ΔV of 400 ft/sec this limitation indicates a thrust-to-mass ratio of at least 0.02 lbf/lbm.

(3) Controllability: Total-impulse control must be accurate to 0.5%. No requirement for thrust variability is apparent. Since one or two accurate corrections appear most desirable, a limited number of restarts are required. Necessary thrust-vector control may often be provided by the conventional attitude-control system, if available.

(4) Storability: Storage times in transit before the terminal correction will be on the order of 1.5 to 3 days.

b. Outbound Mars Flights

The ideal velocity increments required for terminal corrections on outbound Mars missions of 100, 150 and 259 days duration were established vs the initial miss-distance.

A vehicle-borne terminal guidance system similar to that for the lunar mission was assumed. For the Mars mission, however, the sensor might be required to operate in the infrared spectrum rather than the visible spectrum, due to the effects of the Martian atmosphere.

The propulsion requirements for terminal corrections on outbound Mars missions may be summarized as follows:

(1) ΔV: Corrective velocity increments between 100 and 1000 ft/sec are indicated; specific requirements are determined by the terminal guidance accuracy, the energy of the approach trajectory, and the initial miss-distance.

(2) Thrust level: The upper thrust-level limit will be set by the payload-acceleration limits, or by the total impulse accuracy requirement of 1.0%. The lower limit will correspond to the maximum allowable burning time due to the increasing ΔV requirement; this duration limit is set at about 30 minutes, which indicates a minimum thrust-to-mass ratio of approximately 0.02 lbf/lbm.

(3) Controllability: A delivered total impulse accuracy of 1.0% is indicated. No controllable thrust capability is required, although limited restart capability seems to be desirable. Thrust-vector control will probably be provided by an available attitude-control system.

(4) Storability: Storage time, on the order of 100 to 250 days before the terminal-correction maneuver, is indicated.

c. Return Flights

The propulsion requirements for terminal corrections on lunar and Mars return flights were determined. The method of approach used was similar to that for determining outbound terminal corrections, except for these two basic differences:

(1) The terminal-guidance system for return flights is assumed to consist of earth-based, radar-tracking facilities which provide command data to the vehicle; the actual corrective velocity increment is still monitored by a vehicle-borne inertial system.

(2) The lunar return flight lies almost entirely in the terminal-phase flight regime. No midcourse corrections were assumed. However, adequate time is available to make more than one terminal correction during the return flight.

Two types of earth-based, radar-tracking facilities were considered. They are the conventional world-tracking net, using steerable antennas, and a long-baseline, phase-lock system with high angular resolution.

(a) Lunar Return Flights

A representative lunar return flight of 2.75 days duration was chosen. The initial miss distance was 5200 n.mi., which corresponds to an angular error at burnout, in the vicinity of the moon, of about 25 millirads, which is considered to be a very inaccurate launch. Achieving earth impact on the return flight will be easy, particularly for low-energy trajectories, since the vehicle effectively "drops" to earth.

Two techniques for making terminal corrections were considered: (1) corrections are made at radii from the earth ranging from 18,000 to 57,000 n.mi., depending upon the radar system used, and resulting in ideal ΔV requirements of between 395 and 125 ft/sec, respectively; and (2) two corrections are made, one at a radius of 200,000 n.mi. from the earth and a final correction at 18,000 n.mi. The ΔV requirement was found to be 36 ft/sec for the first correction and 29 ft/sec for the second. The total ΔV is thus only 65 ft/sec, using this approach.

The propulsion requirements for trajectory corrections on lunar return flights may be summarized as follows:

1 ΔV: Ideal corrective velocity increments between 50 and 500 ft/sec may be expected. Specific ΔV requirements will depend primarily upon the energy of the return trajectory, the initial miss-distance, and terminal guidance accuracies.

2 Thrust Level: The upper thrust-level limit will be set either by the acceleration tolerance of the payload or by the requirement for 2% total impulse cutoff accuracy. The maximum burning time is restricted to about 20 minutes, due to the increase in ΔV requirement with flight time, which indicates a thrust-to-mass ratio of at least 0.015 lbf/lbm.

3 Controllability: No requirement for thrust variability is apparent; however, capability for several restarts is desirable. Thrust-vector control will be necessary, unless an adequate attitude-control system is available.

4 Storability: Storage times may range from days to months, since the outbound flight time plus stay time must be considered.

(b) Mars Return Flights

The propulsion requirements study for terminal corrections on Mars return flights considered a nominal mission of 150-days flight time, with a 100 n.mi. perigee altitude at earth. The initial miss-distance was 1900 n.mi., which presumes previous midcourse corrections. Only one terminal correction was considered, since the approach velocity is high, resulting in a considerable decrease in the smoothing time which is available for the tracking and guidance functions.

The propulsion requirements for terminal corrections of Mars return missions can be generally characterized as follows:

1 Velocity increments ranging from 200 to 1500 ft/sec may be expected, with specific values dependent largely on the approach-trajectory energy and initial miss-distance. The accuracy of several consecutive corrections will reduce the required ΔV considerably, if adequate smoothing time is available for regaining guidance accuracy.

2 Thrust Level: The upper limit is set by payload-acceleration tolerance or by the 1.0% maximum error in total impulse delivered. The maximum burning time appears to be restricted to approximately 15 minutes for some cases, which indicates a F/m_0 value of at least 0.03 lbm/lbf.

3 Controllability: Total impulse should be accurate to at least 1.0%. No requirement for thrust variability is apparent. Restart capability is desirable if available smoothing time and tracking accuracies will allow for several consecutive corrections, reducing the

total ideal ΔV. Thrust-vector control is required, and may be provided by an auxiliary attitude-control system.

4 Storability: Considering the outbound and return trip durations and probable stay time on Mars, storage times of up to 2 or 3 years may be required. The time between restarts will be on the order of several hours.

D. LUNAR AND PLANETARY ORBITING MANEUVERS

Analyses in the preceding section established the propulsion requirements to achieve a desired perigee altitude at the target. In this section, that perigee altitude will form one apsis point for orbiting maneuvers about the target body.

1. Lunar Orbiting Maneuvers

Approach trajectories of 1.5, 2.0, and 2.75 days were considered, corresponding to hyperbolic approach velocities of 7120, 4950, and about 4000 fps, respectively.

The propulsion requirements for the lunar orbiting maneuvers considered are summarized as follows:

a. ΔV: Impulsive velocity increments ranging between 2000 and 5500 fps will be required, with specific requirements primarily dependent on the energy of the approach trajectory and on the eccentricity of the desired orbit.

b. Thrust level: Initial thrust-to-mass ratios ranging from 1.0 to 2.0 will be desirable.

c. Controllability: Delivered total impulse accuracy should be at least 0.2%. No requirement for thrust modulation is apparent, if the total impulse accuracy is satisfactory. Thrust-vector control will be required to maintain vehicle orientation and to control the trajectory, since thrust-to-weight ratios will be relatively high, and the conventional altitude-control system will usually be inadequate. Restart requirements for the main orbiting propulsion-system are not indicated.

d. Storability: Storage times of several days are indicated by the typical lunar flight times.

2. Mars Orbiting Maneuvers

Propulsion requirements were investigated for maneuvers in orbiting Mars, both with and without the use of atmospheric deceleration. Approach trajectories of 100, 150, and 259 days were considered, corresponding to hyperbolic velocities of 27,150, 17,200 and 8200 fps, respectively.

Without atmospheric deceleration, approach perigee radii from 200 to 2000 n.mi. were considered, with apsidal radius ratios ranging from 0.70 to 3.8.

Atmospheric braking can be utilized to reduce considerably the propulsion requirements for Mars orbiting maneuvers. Several consecutive grazing passes, reducing the apogee altitude, could be made until the desired final apogee altitude is attained. At this time, a final velocity increment would be added to raise the perigee altitude and to establish the desired orbit.

Orbiting maneuvers about the planet Mars can be carried out primarily through atmospheric deceleration for the flights considered here. Allowances on the order of 1000 fps should be included for perigee variation and orbit-control requirements. However, no primary rocket deceleration is required, providing that any aerodynamic heating problems which may arise are solved by vehicle-design techniques.

The propulsion requirements for orbiting maneuvers about the planet Mars may therefore be summarized as follows:

a. ΔV: Impulsive velocity increments for orbiting without atmospheric deceleration range from 4000 to 13,000 fps for 259- and 150-day approach trajectories, with a maximum of 23,000 fps for a 100-day flight, with injection into a high circular orbit. With atmospheric deceleration, desirable orbits can be achieved from all approach trajectories with velocity increments on the order of 1000 fps. The advantage in utilizing atmospheric deceleration is readily apparent.

b. Thrust level: Initial thrust-to-mass ratios on the order of 1.0 to 3.0 lbf/lbm are indicated.

c. Controllability: Total impulse accuracy of 0.1% will be necessary for critical orbits without atmospheric deceleration. Thrust variation, or staging, may be required to maintain tolerable acceleration on maneuvers requiring large ΔVs. If atmospheric braking is used, the fractional total-impulse accuracy is relaxed to 0.3%, since the required ΔVs are small. Multiple restart capability is definitely required for orbiting with atmospheric deceleration, but thrust variation is not indicated, except if required for emergency situations. Thrust-vector control may be required for both cases.

d. Storability: Space storage requirements, before ignition, on the order of 100 days to 1 year are indicated, with times between restart on the order of several hours for the atmospheric deceleration case.

E. LUNAR AND PLANETARY LANDINGS

Three methods for landing on the moon were considered: (a) direct radial approach and landing, (b) injection into circular orbit and a gravity turn from orbit to landing, and (c) injection into circular orbit, transfer to lower orbit, deceleration to zero velocity at low orbit altitude, and vertical descent to the surface.

In calculating the propulsion requirements, errors in measured quantities and operational parameters were considered. The presence of errors indicates that the vehicle should be brought to effectively zero velocity at some safe distance above the surface to avoid destructive impact. Retrothrust with constant deceleration was considered, and final letdown from the zero velocity point was analyzed. Both operations will require variable thrust.

The initial circular orbit altitudes considered were 50 n.mi., and 200 n.mi. The lower circular orbit altitude, to which transfer is made from the 50- or 200-n.mi. orbits, had an altitude of 5 n.mi. A 66-hr trajectory was considered for all cases.

The propulsion requirements for lunar landings using the three methods outlined are as follows:

1. Direct Radial Landing

A direct, radial landing on the moon from a 66-hr trajectory, with no error in ignition altitude, requires an ideal velocity increment of 9000 ft/sec. When error in ignition altitude is included, provision must be made for a letdown from an error altitude of 4 n.mi., assuming 0.33% error in measured quantities. The additional requirement, due to error, is 800 to 900 ft/sec, bringing the total velocity requirement to between 9800 and 9900 ft/sec.

2. Gravity Turn From Circular Orbit

The velocity increment required to land from a 50-n.mi. circular orbit is 5700 ft/sec; from a 200-n.mi. circular orbit it is 6060 ft/sec. The velocity increment required for injection into circular orbits from a 66-hr trajectory is 3200 ft/sec for a 50-n.mi. orbit and 3140 ft/sec for a 200-n.mi. orbit. An error in zero velocity altitude of 5 n.mi. was assumed for the orbit landing, in order to compare the propulsion requirements of the two landing procedures. The velocity increment required to let down from 5 n.mi. is 1100 ft/sec. Thus, the total velocity requirements to land by a gravity turn from orbit, with earth-moon travel time of 66 hr are 10,000 ft/sec from a 50-n.mi. orbit, and 10,300 ft/sec from a 200-n.mi. orbit.

3. Transfer to Low Circular Orbit

The third method considered consists of injection into circular moon orbit, coplanar transfer to a low altitude (5 n.mi.) circular orbit, deceleration at constant altitude to zero velocity, and, letdown vertically to the surface of the moon. Velocity requirements for injection into the 200- and 50-n.mi. circular orbits, and for descent from 5 n.mi. are 3140 ft/sec, 3200 ft/sec and 1100 ft/sec, respectively. The velocity increment requirement for transfer from a 50-n.mi. circular orbit to a 5-n.mi. circular orbit is 110 ft/sec; for a transfer from a 200-n.mi. orbit to a 5-n.mi. orbit the requirement is 465 ft/sec. Constant altitude deceleration at 5 n.mi., with a thrust-to-mass ratio of 1.0 requires ΔV = 5600 fps; this requirement does not vary rapidly with changes in either thrust-to-mass ratio or altitude. An effective error of 1^{o} in thrust-vector angle was assumed, resulting in an additional velocity requirement of 350 ft/sec, at a thrust-to-mass ratio of 1.0. Therefore, the total propulsion requirements for landing on the moon, using this method are: ΔV of 10,360 ft/sec for a 50-n.mi. orbit and ΔV = 10,655 ft/sec for the 200-n.mi. orbit.

From the foregoing totals of propulsion requirements, it is evident that landing from orbit, on a gravity turn, has the lowest propellant requirement of the orbit-landing techniques considered. However, the constant altitude deceleration method is only 2 to 4% more costly, and this margin can be reduced if more accuracy than that which was assumed can be achieved in the thrust-vector control. A direct, radial landing requires 2 to 8% less velocity increment than the orbital cases and may be desirable in order to reduce navigational and operational complexities. All of the maneuvers require thrust variability; a thrust variability of 6 to 8 appears to be adequate for most cases.

2. Mars Landing

The propulsion requirements for landing on Mars were investigated for a direct, radial landing and a landing from orbit.

a. Direct, Radial Landing

Direct, radial landings on Mars considered three approach velocities: 18,100 ft/sec, 22,850 ft/sec and 31,300 ft/sec, corresponding to Earth-Mars transit times of 259, 150, and 100 days, respectively. The analysis indicated that these velocities are too high for a direct entry to the atmosphere using only a single application of retrothrust before landing. Not only is an excessive aerodynamic heating rate expected, but the deceleration required, due to thrust alone, is excessive. Consequently, the deceleration due to thrust was limited to 8g (earth), to simulate the landing restrictions on a manned vehicle. With this limitation, the minimum achievable velocity requirements are on the order of 11,000 ft/sec, for an approach velocity of 18,100 ft/sec; 14,500 ft/sec, for an approach velocity of 22,850 ft/sec; and 19,500 ft/sec, for an approach velocity of 31,300 ft/sec. These high approach velocities require that ignition of the main retro-rockets take place hundreds of miles from the surface of Mars, resulting in large errors in ignition and additional ΔV requirements. The total propulsion requirements (including error effects), for a direct landing from an approach velocity of 18,100 ft/sec with I_{sp} = 430 lbf-sec/lbm, are then found to be approximately 13,000 ft/sec.

b. Landing from Mars Orbit

Two initial circular orbits were considered, one at 1000 n.mi., and one at 350 n.mi. It was determined that the velocity requirements (including injection into orbit), to land on Mars on a gravity turn from the 1000-n.mi., and 350-n.mi. orbits were considerably greater than those for the direct approach when atmospheric braking was not considered. However, if the technique of entering an elliptical orbit, employing atmospheric braking, is used, the orbit-landing requirements are reduced to the following extent: for the 18,100 ft/sec approach velocity (259-day trajectory) the requirements are comparable to the direct approach, and for the fast approach velocities (100- and 150-day approach trajectories), the requirements are 10,000 ft/sec and 15,000 ft/sec, which are considerably lower than those for the direct approach.

The initial thrust-to-mass ratio for the landing maneuvers should not exceed 2.0, to avoid excessive g loads, and the engine should have the capability of throttling to 10% of full thrust.

F. LUNAR AND PLANETARY TAKEOFFS

The propulsion requirements for lunar and planetary takeoffs were analyzed through the use of a conventional gravity-turn ascent-trajectory program on the IBM 7090 computer. The calculations assumed constant thrust and specific impulse, and included an appropriate gravitational constant and atmosphere (as applicable). However, the rotation of the moon or planet was not considered. Trajectories were run for various values of the thrust-to-mass ratio, and the "kick-angle" was optimized, when possible, to achieve minimum propellant expenditure for each of the initial thrust-to-mass ratio (F/m_o) values.

1. Lunar Takeoffs

The propulsion requirements were determined for lunar takeoffs with ascent trajectories into lunar orbits of 50- and 200-n.mi. altitude, and with direct injection on return flights to earth of 1.5, 2.0, and 2.75 days duration. The lunar ascent trajectories were computed for F/m_o values ranging from 0.5 to 9.0 lbf/lbm.

The general propulsion requirements for lunar takeoff missions may be summarized as follows:

a. ΔV: Ideal velocity requirements on the order of 6000 to 6500 ft/sec may be expected for takeoffs to lunar orbits. For direct trajectories to earth, the takeoff ΔV requirements are on the order of 9200 to 11,500 ft/sec.

b. Thrust level: Values of the initial thrust-to-mass ratio in a range from 0.8 to 1.6 lbf/lbm appear to be representative of the optimum F/m_o for most missions.

c. Controllability: Since the burnout velocity should be held to an accuracy of about 5 ft/sec for takeoffs to low orbits, the required percentage of accuracy in delivered total impulse will be about 0.1%. No requirement for thrust variability is indicated. Thrust-vector control will normally be required, due to the relatively high thrust-to-weight ratios at burnout. For lunar orbiting maneuvers, the need for restart at injection is indicated.

d. Storability: The system will undergo storage in the environment of space for 1 to 3 days in the outbound trip, and may subsequently be stored for periods of weeks to months on the lunar surface before the takeoff.

2. Mars Takeoffs

The propulsion requirements were determined for takeoffs to Mars orbits of 350- and 1000-n.mi. altitudes, and for injection on Mars-Earth flights of 100, 150, and 275 days duration. The characteristics of the ascent trajectories differed somewhat from those of the lunar flights, due to the presence of the Martian atmosphere.

a. Takeoffs to Orbit

The takeoff maneuvers for injection into circular orbits of Mars were analyzed for initial thrust-to-mass ratios, ranging from 0.75 to 6.0 lbf/lbm.

b. Takeoffs for Mars-Earth Flights

The Mars ascent trajectories, for direct flights to earth, have been based on the use of a two-stage vehicle, since the velocity increments required appear to be excessive for a single-stage system. The addition of a second stage introduces a number of additional parameters and degrees of freedom, such as the relative mass of each stage, the relative thrust levels between stages, etc. To reduce the number of possible combinations to a reasonable selection, it was assumed that the first stage would consume 53.3% of the gross vehicle weight as propellant, in every case.

The propulsion requirements for Mars takeoffs may be summarized as follows:

(1) ΔV: The ideal velocity increments for Mars takeoffs into circular orbit are on the order of 15,000 to 17,000 ft/sec. For takeoff on direct trajectories for return to earth, the requirements vary from 20,000 to 36,000 ft/sec, dependent largely on the flight duration for the Mars-Earth trajectory.

(2) Thrust level: The initial thrust-to-mass ratios for Mars takeoffs to orbit are on the order of 0.7 to 1.0 lbf/lbm, while the values for both stages of the two-stage vehicle are about 1.5 to 2.0 lbf/lbm.

(3) Controllability: The desired accuracies for orbiting maneuvers and injection on return flights to earth indicate burnout accuracies on the order of 10 ft/sec. This requirement indicates total impulse control to .03% for critical cases. No requirement for thrust modulation is apparent, other than a possible use for vernier cutoff. A single restart is indicated for the takeoffs to Mars orbits. Thrust vector control will be required due to the large thrust-to-mass ratios involved.

(4) Storability: The system will be stored for periods up to 260 days in transit on the outbound flight, and storage times on the surface of Mars may range from months to years. The time between restarts will be on the order of minutes for the orbiting maneuvers.

III. SYSTEM CONCEPTS

This portion of the study reviews the applicable propellants and engine concepts, which could satisfy the mission requirements under consideration. The purpose of this review was to provide the necessary background to develop a discrete family of propulsion systems. The results are combined with the findings of the Mission Analysis section to provide a basis for propulsion-capability classifications.

The propulsion system concepts were examined on the basis of performance, control versatility, and adaptability to meet operational environment factors. The scope of the systems study encompassed: (a) chemical systems including liquid, solid, and hybrid systems; (b) nuclear systems, and (c) electrical systems.

A. CHEMICAL SYSTEMS

1. Liquid Propellants

a. Cryogenic Bipropellants

The cryogenic bipropellants which offer the highest specific impulse are the combinations of liquid hydrogen with liquid fluorine, and liquid hydrogen with liquid oxygen. Since fewer handling difficulties are associated with liquid oxygen, the LO_2/LH_2 combination was considered most currently representative of this class and was therefore chosen as the propellant combination for consideration. After selecting the propellant combination, the propellant fraction, as a function of total impulse for several thrust values, was computed.

b. Storable Bipropellants

The storable bipropellant selected, on the basis of experience and data amassed at Aerojet, was N_2O_4/Aerozine-50. The propellant fractions were calculated for systems using this combination.

c. Storable Monopropellants

The Cavea-B monopropellant was chosen to represent this class because of its development history, performance, and also logistic considerations. The propellant fractions were determined for monopropellant systems using this propellant.

2. Solid Propellants

For space operations, solid propellants should have the highest available specific impulse, consistent with such common operational requirements as good physical properties, long-term stability under severe environmental conditions, reliability, etc.

Increased performance can be predicted with the use of more energetic oxidizers, fuels, or binders.

A beryllium-containing propellant would have a delivered specific impulse, at a 40:1 expansion ratio in vacuum, of 305, while a formulation containing hydrazine perchlorate has a vacuum specific impulse of 300.

Further advances in performance include the use of encapsulated metal hydrides, the development of binders containing hydrazine or difluoramino groups, and new, high-energy oxidizers. The reduction of inert parts weight, by the use of new materials, was seen as another method of improving solid-rocket performance.

Propellant fractions as a function of total impulse were developed for solid-propellant systems, assuming a vacuum specific impulse of 305.

Graphs based upon some early advancement in propellant development were also prepared, showing future propellant fractions to be expected as a function of total impulse (to 10^6 lbf-sec) for various thrust levels.

3. Hybrid Systems

The hybrid system having a solid-fuel grain and a liquid oxidizer was compared with the all-solid and all-liquid chemical systems.

Compared to a solid-rocket propulsion system, the hybrid system should have improved thermal cycling characteristics over a wider temperature range, greater vibrational endurance, and, in general, better physical properties. On the other hand, the necessity of dealing with two physical states may limit either the choice of propellants or the usable temperature range of the system.

Compared to a liquid bipropellant system, the hybrids are potentially simpler in design and capable of greater reliability.

Vacuum specific impulse values of well over 340 lbf-sec/lbm appear to be achievable with well-known oxidizers and fuels.

B. NUCLEAR HEAT-TRANSFER SYSTEMS

A general review of merit, potential, and characteristics of nuclear propulsion systems was undertaken. Two of the most feasible concepts for the utilization of fission were considered. The direct transfer of heat to a working rocket, and the conversion of heat energy into electrical power for ion propulsion.

C. ELECTRIC ENGINE SYSTEMS

Miscellaneous electric engine systems, including the colloidal-particle engine (a variation of the conventional ion engine), were considered in this portion of the study. The arc-jet engine and plasma engine were briefly reviewed.

D. CONTROL CHARACTERISTICS

The various propulsion systems were compared on the basis of the degree of control flexibility and capabilities that they offer. These control characteristics include the ability of the propulsion system to provide thrust-vector control, thrust-level control, and total-impulse control. The study was limited to qualitative capabilities and comparisons, with only occasional use of approximate numerical data to provide some orientation.

E. OPERATIONAL CONSIDERATIONS

The manner in which known environmental conditions of space will affect the various propulsion systems was reviewed. The conditions encountered and the effects considered include zero-g conditions, temperature control, meteroids, ionizing radiation and the effects of the vacuum environment.

IV. MISSION/SYSTEM CLASSIFICATION

The final section of this report is intended to (a) summarize the overall space-propulsion requirements that have been developed by the mission analysis work, (b) provide a condensed summary of the advantages and limitations of various propulsion systems that were considered in the system concepts section, and (c) to indicate any practical categorization of the numerous and diverse mission requirements and system concepts into a limited set of primary propulsion-system capabilities.

Due to the comprehensive nature of both the mission analysis and system-concepts work, a very broad coverage of space-propulsion requirements and applicable propulsion systems has been accomplished. It has, therefore, been found necessary to select "representative" propulsion requirements and "typical" system characteristics in order to present a compact and understandable picture of the overall Phase I results and conclusions. These limitations are necessary only for the sake of clarity; the ranges of basic parameters that were included in Sections II and III of Volume II would allow similar considerations to be developed for other payload/mission/propulsion-system combinations that may be of interest.

A. SUMMARY OF PROPULSION REQUIREMENTS

The space-propulsion requirements that are parametrically presented in Section I, Volume II of this report, have been summarized and condensed in Table 1. This table presents three groups of data for each of the space maneuvers considered, (1) basic mission requirements, (2) representative system characteristics, and (3) requirements peculiar to liquid-propellant systems. The maneuvers listed include all operations considered in Section II of Volume II, except the effects of solar-radiation pressure and of solar and lunar gravities. Such effects are extremely small and can be most efficiently counteracted by use of the vehicle's conventional attitude-control system. Atmospheric drag, while requiring somewhat greater corrective measures, cannot be generalized beyond the indicated limits.

1. Basic Mission Requirements

The basic propulsion requirements for each maneuver, as established by mission analysis, are indicated by the following characteristics:

a. Range of ideal velocity-increment requirements
b. Range of desirable initial-thrust-to-mass ratios
c. Total-impulse accuracy
d. Required thrust variability
e. Restart requirements
f. Thrust-vector control requirements
g. Storability requirements.

The range of ideal velocity increments for each maneuver, typically, is quite broad, since it includes the requirements for all probable variations of the maneuver under consideration. The ideal ΔV requirements, typically, are related to the characteristics of a specific maneuver in a straightforward manner.

The thrust (and acceleration) requirements for the various operations are indicated as a range of "desirable" F/m_o values, where F is the nominal thrust, and m_o is the initial gross mass of the vehicle. The indicated F/m_o ranges are based on requirements and limitations (sometimes qualitative) that could be established through mission analysis, including such considerations as gravity loss, final acceleration, guidance characteristics, maneuver accuracy, and engine weight.

The requirements on delivered total-impulse accuracy that are expected for the various maneuvers have been indicated in two forms, (a) the allowable total-impulse error per unit-mass at cutoff, $\Delta I_t/m_f$, and (b) a typical percentage of total-impulse error, $\Delta I_t/I_t$. Although the typical percentage of impulse error is a representative characteristic for many maneuvers, the impulse error per unit-mass is the primary and definitive requirement, from the propulsion system standpoint.

When control of thrust level (or total impulse per unit-time for pulsing systems) is necessary, due to the basic characteristics of a maneuver, it has been indicated as a ratio of thrust levels under the heading "Required Thrust Variability". The basic requirement for thrust variability arises during operations that specify exact position and velocity conditions, to be achieved simultaneously. For example, rendezvous and landing maneuvers inherently require thrust variability, while takeoffs do not. Secondary requirements, such as thrust reduction at cutoff for impulse accuracy, or staging to limit maximum accelerations, may be necessary for certain systems; however, these are not basic mission-related variability requirements and are therefore excluded from this column.

The requirements for restart capability, thrust-vector control, and system storability are indicated directly in the next three columns of the table. The necessity for thrust-vector control by the propulsion system itself (through engine gimbaling, vernier engines, etc.) may often be eliminated by use of the vehicle's conventional attitude-control system, when acceleration levels are low.

2. Representative System Characteristics

A set of representative system characteristics for the various space maneuvers form the next major section of Table 1. The representative systems are defined by values of initial vehicle mass, thrust level, total impulse, and the resulting payload for the maneuver of interest. Two distinct systems are presented for each maneuver, characterized by (a) a relatively large payload (either manned or unmanned), and (b) a small payload (unmanned).

The initial vehicle-mass values were generally selected to be roughly consistent with the payload capabilities of either the SATURN, NOVA, or CENTAUR launch vehicles. The reference vehicle for each specific case is indicated by the initial-mass superscript and the associated footnote. For manned, planetary missions it was necessary to go beyond the payload capabilities of these boosters in order to achieve adequate space-vehicle sizes. For these cases, the use of orbital-launching techniques, or the future increase of booster capability is assumed. While there is some continuity in relative sizes carried through the various maneuvers involved in probable overall missions, no attempt was made to make this rigorous; each initial mass should be considered individually, as representative of either launch capability or mission necessity. It must be re-emphasized that the selected vehicle sizes were intended only to establish representative values of thrust and total impulse for applicable space-propulsion systems; they are not intended as accurate indications of the capabilities of the referenced launch vehicles for the maneuver under consideration.

3. Liquid-Propellant System Requirements

The final two columns of the table are related basically to liquid-propellant propulsion systems. Since these systems have a cutoff-impulse inaccuracy which is in effect an error in cutoff time, the fractional total-impulse inaccuracy can be reduced by decreasing the thrust-to-mass ratio at cutoff.

The first column of the final section indicates the maximum thrust-to-mass ratio at cutoff for which liquid-propellant systems can achieve the required total-impulse accuracy. These values are based on the allowable $\Delta I_t/m_f$ indicated in a previous column, combined with the cutoff-impulse accuracy characteristics for liquid systems as discussed in Section III of Volume II. If the thrust level at cutoff is expected to exceed this maximum value, then the need for thrust reduction or vernier cutoff for liquid-propellant systems is indicated by the last column of the table.

B. SUMMARY OF SYSTEM CONCEPTS

The many and varied propulsion systems described in Section III, Volume II of this report are listed in Table 2, together with their characteristics and limitations. The system characteristics have been grouped in two general areas, (1) system performance characteristics, and (2) control and operational considerations.

The representative liquid, solid, and hybrid chemical; nuclear-heat-transfer; and electrical-propulsion systems have been tabulated in the first column. Not every conceivable subsystem combination is included, inasmuch as many would represent only minor variations, essentially unevaluated concepts, or detailed component alternatives. Conversely, evaluation of the capabilities of some of these subsystems was necessary to establish the performance of the major systems in the areas indicated in subsequent columns of the table.

1. System Performance Characteristics

System performance for the propulsion systems under consideration has been represented by two basic parameters, specific impulse and system-propellant fraction. The specific-impulse column of the chart is intended to show present (or near-future) achievable performance, and the currently foreseen limits. While it may be acknowledged that future development and evaluation will prove many of these upper performance-limit values to be practically unattainable, it should also be recognized that even better combinations may emerge from future research. The utility of the specific-impulse figures then resides in the general picture they present, indicating where the principal propulsion systems fit into the overall performance spectrum.

The next five columns show approximate propulsion-system propellant fraction, m_p/m_t, for several values of thrust level and total impulse. These data have been abstracted either from rough design studies conducted on this program or from literature values. In general, the numbers represent some advancement from present practice, e.g., use of titanium propellant tanks, and may be several percent higher than for current designs. This is particularly true for maximum values which are intended to reflect considerable future state-of-the-art improvement. In some cases, as for pressure-fed bipropellants and hybrids, insufficient information was found on which to base even hypothetical extrapolations into the future. In addition, the attainment of such "ultimate" figures, which are predicated upon sophisticated materials and fabrication techniques with minimum safety factors, will be significantly influenced by the type of mission (manned or unmanned), reliability considerations, and the availability of large booster systems. It is apparent that most chemical systems, when considered in their most applicable size range, eventually will not differ substantially in propellant fraction. Where such differences exist between any of the propulsion systems listed, other factors, such as specific impulse, may be of greater importance in selection.

2. Control and Operational Considerations

The first column in this area indicates an attempt to correlate cutoff inaccuracy to thrust level; the resulting equations and observations are presented. This is a characteristic that requires further study, since it is so intimately related to mission-error analysis, and some Phase II effort in this study will be devoted to a more thorough assessment of available operational data.

The next nine columns of the table represent system attributes that either cannot be expressed numerically or for which definitive data have not yet been obtained. A three-grade rating system was used to evaluate these parameters. Despite the inherent subjectiveness and possible personal bias inherent in such an approach, it does indicate areas of excellence, difficulty, and ignorance. Considerable opportunity exists in these areas for revising present performance estimates, through system development and a better definition of the space environment. Yet these parameters very frequently guide system selection. While a portion of the Phase II study work will be allocated to additional evaluation of such system characteristics, it remains primarily an area dependent on the pace of future engineering developments.

It should be recognized that Table 2 represents a condensation and at times an over-simplification of many factors, and should be used in conjunction with the more detailed treatment in Section III, of Volume II. Such a compilation indicates the diverse propulsion systems and their performance parameters included in the study, and permits initial system classifications considering the established mission requirements.

C. CATEGORIZATION OF PROPULSION REQUIREMENTS

To conclude the results of the Phase I study effort, an attempt has been made to categorize the diverse space-propulsion requirements into a compact group of desirable propulsion-system capabilities. Table 3 indicates the propulsion requirements for the representative system characteristics (see Table 1), in an abbreviated form. A series of four propulsion-capability classifications are then presented. These are found to satisfy, in general, the majority of the representative space-propulsion requirements. The capability classifications suggest an associated set of fairly distinct propulsion-system types.

1. Abbreviated Space-Propulsion Requirements

The representative propulsion requirements for the maneuvers under consideration are indicated in the first four columns of Table 3. Although considerable detail has been omitted relative to the previous tabulation, these abbreviated requirements are adequate for purposes of this section.

2. Propulsion-Capability Classifications

The four propulsion-capability groups which have been developed to satisfy the majority of the tabulated requirements are indicated on the right of the chart. They include:

a. Class I: Systems in the range of 100 to 1000 lbf nominal thrust providing 0.01 to 0.20 x 10^6 lbf-sec total impulse. The classification does not require thrust-vector control, but must be capable of multiple restarts and provide accurate total-impulse control with variable impulse per unit-time desirable.

This classification suggests a pulse rocket, possibly operating on storable propellants, with a radiation-cooled chamber.

b. Class II: Propulsion systems with nominal thrust in the 2000 to 20,000 lbf range and 0.20 to 2.0 x 10^6 lbf-sec total-impulse capability. Multiple restarts and thrust-vector control are necessary, but no thrust variability and only normal total-impulse accuracy is necessary.

This category suggests conventional storable, or high-performance engines, with regeneratively-cooled or ablative chambers. Several engines are currently available, or are being developed, with satisfactory capabilities in this requirement classification.

c. Class III: Systems with nominal thrust in the 20,000 to 100,000 lbf range, capable of approximately 10:1 thrust variability. Total-impulse requirements range from 2.0 to 20 x 10^6 lbf-sec; multiple restart capability and thrust-vector control is required. Accurate total-impulse control will be available through thrust reduction before cutoff.

The thrust variability requirement implies the use of film, transpiration, and/or radiation-cooling techniques; or the use of ablative chambers, if burning times are not excessive.

d. Class IV: Nominal thrust on the order of 1 to 6 x 10^6 lbf, with total-impulse capability in the range from 100 to 1000 x 10^6 lbf-sec. Normal total-impulse accuracy is adequate, and no thrust variability or thrust-vector control is suggested, since the use of small, highly-controllable auxiliary engines seems desirable for most missions. However, the system should incorporate multiple restart capability. This classification suggests a large, high-performance, propulsion system of otherwise conventional design.

3. Applicability of Requirement Classifications

The coverage of the representative space-propulsion requirements which is provided by the four capability classifications is indicated by the intermediate column of the table. It may be noted that the majority of the requirements have been included, indicating that the four propulsion-capability classifications are fairly indicative of the entire spectrum of space-propulsion requirements which have been considered.

V. OBJECTIVES AND APPROACH

Study efforts during Phase II were directed toward the optimization and conceptual design of the most promising propulsion systems for specific space missions. These include: (1) manned circumlunar missions, (2) manned lunar orbiting and return mission, (3) manned lunar landing and return mission, and (4) unmanned 24-hour satellite mission.

The specification of detailed propulsion requirements and propulsion criteria for these space missions was largely extracted from the results of Phase I (Vol. II), with extensions including (a) selecting and verifying the appropriate three-dimensional nominal trajectories for each lunar mission, (b) verifying the propulsion requirements for trajectory corrections, and (c) further analysis of requirements and criteria for the specific maneuvers at the moon and in a 24-hr earth orbit.

The qualitative selection of applicable propulsion systems to meet the propulsion requirements for the space missions in question entailed the establishment of basic alternate propulsion-system integration concepts for the overall space mission under consideration. This was necessary because of the variations that exist in such primary parameters as total impulse, thrust level, and initial mass, between the individual maneuvers and the probable combination of these requirements in the overall mission. The integrated concepts resulted from appropriate combinations of the requirements for the individual maneuvers comprising the mission within the objectives of the overall mission.

Based on the comparative performance and evaluation data, specific systems were selected which appear superior for the several missions. These systems incorporate the best combination of overall performance capability with minimum compromise of system reliability, ease of development, operational characteristics and flexibility. Injected spacecraft weights consistent with the capabilities of the Nova, Saturn C-3, Saturn C-2, and Centaur launch vehicles were selected as representative for the various missions.

A. LUNAR MISSION

1. Propulsion Requirements and Criteria

In specifying propulsion requirements and criteria for the lunar missions, the following maneuvers were considered: abort at injection; trajectory corrections for the outbound, circumlunar, and return phases; orbiting maneuvers at the moon; perigee variation for the lunar orbit; lunar landings from orbit; return launch from lunar orbit; and return launch from the lunar surface. The combination of these maneuvers provides the basis for all space-propulsion requirements derived, including the following parameters, each independent of the total vehicle mass: ideal velocity increment, thrust-to-mass ratio, impulse accuracy, restarts, thrust variability, and thrust vector control.

2. Applicable Systems

The selection of propulsion systems to best meet the mission requirements was based on consideration of quantitative weight and dimensional criteria, modified by qualitative system attributes and limitations. Propulsion systems were divided into five areas to facilitate the analysis: propellants, tankage, structure, thrust chamber assembly, and pressurization system. Within each of the five areas, operating conditions were chosen, and subsystem weights computed on the basis of fabrication experience, empirical correlations, or analytical equations.

Propellants: Liquid propellant systems were considered primarily for the lunar mission, although solids were compared for specific purposes, e.g., the abort function. Among the cryogenics, the liquid oxygen/liquid hydrogen combination was selected as the most representative of this class. The nitrogen tetroxide/Aerozine-50 combination was selected as a representative and desirable storable-propellant system.

Tankage Weight: All tankage was initially assumed to be spherical, although, in the final evaluation, cylindrical or ellipsoidal tankage was examined for the selected system where it appeared to be more appropriate.

Engine Weights: With regard to engine weights, three basic types of thrust chambers were considered, differentiated by the method of cooling: radiation, ablative, and regenerative. In addition, the ablative and regenerative engine could be either pump- or pressure-fed systems, resulting in the weight difference attributed to the turbopump assembly. The effect of gimballed thrust-vector control capability on engine weight was also included.

Pressurization System Weights: An extensive analysis of the many possible pressurization systems was not warranted by the scope of this study. Two basic pressurization systems were considered, which are typical for cryogenic and storable systems.

Total Propulsion-System Weights: The data generated with regard to the above mentioned subsystem weights were utilized in establishing the total propulsion-system weights for various systems and parameter ranges. An additional allowance of 3% of the propellant weight was made for structure in the initial propulsion systems comparisons. This estimate was subsequently improved by structural weight analyses for the selected configurations.

B. 24-HR SATELLITE MISSION

Throughout the analysis of the 24-hr satellite mission, the satellite was assumed to be an active communications relay with density, configuration position and attitude tolerances typical to this type of satellite. The analysis was divided into two basic parts. First, the propulsion requirements were established and secondly, the competitive systems for each of the payload weights were compared to determine the best system for each payload.

Propulsion requirements were determined for the three basic functions necessary for 24-hr satellite operations: correction, station keeping and attitude control.

VI. LUNAR MISSIONS

A. MANNED CIRCUMLUNAR MISSIONS

The analyses of manned circumlunar missions were based on a spacecraft weight of 150,000 lb or less, with a minimum return capsule weight of 12,500 lb. To cover this wide range of payload weights with some realism, a series of planned launch vehicles were considered, including the Nova, the Saturn C-3 and the Saturn C-2.* The configuration of the Apollo spacecraft was utilized as a typical lunar-mission capsule.**

1. Mission Requirements

a. Trajectory Analysis

The propulsion requirements and criteria for the manned circumlunar mission were established on the basis of a selected nominal trajectory. A trajectory analysis included (1) a translunar outbound trajectory, which carries the vehicle to the vicinity of the moon; (2) a hyperbolic retrograde encounter with the moon, which curves the trajectory back toward the earth; and (3) the trans-earth trajectory from the vicinity of the moon to the re-entry point near the earth's surface.

A computer study of three-dimensional ballistic trajectories for the lunar missions was also undertaken to verify trajectory characteristics and requirements established by two-dimensional computer work and closed-form analytical calculations. From a series of approximately 100 trajectory runs, a sample circum-lunar trajectory was selected for verification of the propulsion requirements previously derived on an analytical basis.

b. Propulsion Requirements for Maneuvers

The parametric propulsion requirements were determined for maneuvers including (1) abort-at-injection capability, and (2) trajectory corrections on both the outbound and return portions of the mission.

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*References 10 and 11, Vol. III
**Reference 12, Vol. III

(1) Abort-at-Injection

Concerning the abort-at-injection maneuver, it was determined that no restart or thrust variability is required for the abort operation; however, thrust-vector control must be provided to maintain stability and orientation. Total-impulse accuracy is not critical for the abort operation.

(2) Trajectory Corrections

The circumlunar mission was considered to include the following individual maneuvers: (1) a midcourse outbound trajectory correction; (2) a terminal outbound trajectory correction; (3) first return-trajectory correction for time of flight and perigee errors, and (4) the final return-perigee correction.

The velocity requirements for trajectory correction were first evaluated by the analytical study, then verified by the computed sample three-dimensional trajectories. An outbound midcourse correction of 100 fps was indicated at approximately 50,000 miles from earth, with a total-impulse accuracy of 0.01 lbf-sec/lbm. This corresponds to an error in implementation of the corrective velocity increment of about 0.3 fps. An initial thrust-to-mass ratio F/m_o between 0.05 and 0.10 lbf/lbm is suggested for the maneuver. The terminal outbound correction of 50 fps will occur at approximately 10,000 miles from the moon, again with a total accuracy of 0.01 lbf-sec/lbm and a thrust-to-mass ratio between 0.05 and 0.10 lbf/lbm.

The first return trajectory correction was postulated to occur at approximately 50,000 n.mi. from earth. A ΔV capability of 500 fps is included for this maneuver. A thrust-to-mass ratio of 0.07 lbf/lbm will result in burning time of about 200 to 250 seconds. Total-impulse accuracy is set at 0.10 lbf-sec/lbm resulting in a velocity error of about 3 fps. The final trajectory correction occurs at a point about 10,000 n.mi. from the earth, and is intended to correct the perigee distance and flight path with sufficient accuracy for the atmospheric entry maneuver. A corrective velocity increment of about 100 fps is indicated since the time-of-flight correction will result in some unavoidable perigee variation. A thrust-to-mass ratio in the range of 0.05 to 0.10 lbf/lbm is satisfactory, and the impulse accuracy for the final correction should be on the order of 0.01 lbf-sec/lbm.

A minimum of four restarts will be required for the trajectory corrections. Thrust variability will not be necessary; however, thrust-vector control must be provided either by the propulsion system itself or by an attitude-control system, to maintain accurate thrust-vector control as well as vehicle orientation and stability.

(3) Summary of Specific Requirements

The specific values for maneuver propulsion requirements associated with various launch-vehicle payload capabilities which were considered are presented in Table 4. These values are based on the requirements established for the separate individual maneuvers; they do not reflect desirable changes which result due to integration of the system to satisfy overall mission requirements.

No extensive space storage will be required for the abort system since operation, if initiated, would begin at or shortly after injection. Since the abort maneuver could be carried out even after injection-stage burnout, the capability for a zero-g start must be included.

The circumlunar trajectory correction system will be subjected to extensive storage durations in space. The final return correction will take place from 5 to 7 days after injection. Since the circumlunar trajectory consists entirely of a corrected ballistic trajectory, the four or more restarts must all be made under zero-g conditions.

2. Selected Concept and System Specification - Nova Circumlunar Mission

The selection and integration of appropriate propulsion systems for each vehicle was achieved by (a) evaluating absolute maneuver requirements based on the general maneuver characteristics outlined in the preceding sections,

(b) consideration and selection of appropriate components and systems to satisfy these requirements, (c) integration and evaluation of complete configurations for the mission, and finally (d) specification of the recommended integrated systems for each mission/vehicle combination.

a. Configuration and Mission Sequence

The overall configuration and conceptual design characteristics of the selected Nova circumlunar vehicle are illustrated in Figure 1. The tankage consists of two N_2O_4 tanks and two Aerozine-50 tanks; the four ablative thrust chambers are grouped near the center of the vehicle to reduce canting losses. The excess payload carried on the circumlunar mission over and above the maximum assumed capsule weight of 20,000 lb is carried between the propulsion system and the capsule. The overall length, including payload, is about 20 ft; the diameter is maintained at the 154-in. manned-capsule diameter. The gimballed engines supply thrust-vector control; no provisions have been included for attitude control in the conceptual design analyses. It is assumed that the zero-g propellant expulsion would be carried out by the use of bladders in the storable propellant tank.

The mission sequence for the Nova circumlunar vehicle, Configuration 5-B_1, can be described as follows: During the final burning of the launch-vehicle stage and through a period shortly after injection, abort capability is assumed to be available through the use of solid-propellant motors for abort of the manned capsule only. The initial trajectory corrections are made on the outbound mission utilizing the four ablative thrust chambers. The engines are fully redundant, supplying very high reliability for the required correction maneuvers. After undergoing the hyperbolic encounter with the moon, the vehicle will require at least two trajectory corrections on the return flight. These corrections are also carried out with the four 2K ablative thrust chambers.

b. Tabular System Specification

Tabular specification of the selected propulsion system characteristics are presented in Table 5.

The selected $5\text{-}B_1$ configuration for the Nova circumlunar mission as specified in this table is capable of meeting all the mission requirements as established in Table 4. It is therefore recommended for further consideration as a desirable concept for trajectory-correction propulsion on circumlunar missions in this payload class.

3. <u>Selected Concept and System Specification - Saturn C-3 Circumlunar Mission</u>

a. Configuration and Mission Sequence

The configuration for the selected alternate $8\text{-}A_1$, for the Saturn C-3 circumlunar mission is shown in Figure 2. A single N_2O_4 tank and a single Aerozine-50 tank are utilized, with two ablative chambers. The overall vehicle length is approximately 18 ft, and the vehicle diameter is maintained equal to the 154-in. manned-capsule diameter.

The mission sequence is identical to that previously described for the Nova circumlunar operation.

b. Tabular System Specification

The required specification of system parameters for the selected Saturn C-3 is presented in Table 5. Since no specific problems or deficiencies are evident, the vehicle is considered entirely capable of carrying out the circumlunar mission. The required total-impulse accuracy of 400 lbf-sec is adequately provided by the selected system as indicated on the specification table.

4. <u>Propulsion System Selection and Integration - Saturn C-2 Circumlunar Mission</u>

a. Configuration and Mission Sequence

The configuration for the Saturn C-2 circumlunar vehicle, alternate $9\text{-}A_1$, is indicated in Figure 3. The general design concept is quite similar to that for the Saturn C-3 system, except for reduction in size. A single N_2O_4 tank and a single Aerozine-50 tank are utilized in the configuration, with two 1K ablative chambers. Two 2K engines with ablative chambers are mounted with canting toward the c.g. of the vehicle; however, their small size permits spacing which avoids significant impulse loss. The single N_2O_4 tank and the single Aerozine-50 tank are located so that the c.g. location is established and maintained on the vehicle centerline during propellant expulsion. The overall spacecraft length is 13-1/2 ft, including payload, and the diameter is maintained at 154 in., equivalent to the manned-capsule diameter.

The mission sequence for the Saturn C-2, 9-A_1 alternate is identical to that for the Saturn C-3 Nova circumlunar operations. All corrections are made with the two 1K engines operated either together or alone. No abort capability is provided.

b. Tabular System Specification

The required detailed specification of the selected Saturn C-2 circumlunar vehicle, alternate 9-A_1, is presented in Table 5. All pertinent mission requirements as established in Table 4 are satisfied by the recommended configuration; the required cutoff accuracy of 150 lbf-sec for the trajectory corrections is easily provided by the selected system. The inherent reliability factor indicates that the storable-propellant combination, pressurized feed system, and redundant engines again provide excellent reliability for the Saturn C-2 circumlunar mission.

B. MANNED LUNAR ORBITING AND RETURN MISSIONS

The manned lunar orbiting and return mission assigned for Phase II study considered spacecraft weights of 150,000 lb or less with a minimum capsule weight of 12,500 lb. The launch vehicles considered included the Nova and Saturn C-3, which provide capabilities at either extreme of the range specified above. The configuration of the Apollo manned capsule was again utilized as a typical return and re-entry module.

1. Mission Requirements

a. Trajectory Analysis

The orbiting and return mission consists of (a) outbound trajectory corrections, (b) a lunar orbiting maneuver, (c) perilune variation, (d) return orbital launch, and (e) return trajectory corrections. The outbound trajectory corrections are intended to provide the correct perilune distance for direct injection into the desired lunar orbit. As the vehicle nears perilune point of the approach trajectory, a velocity increment is added in the retrograde direction for injection into a circular orbit. It is assumed that it will be desirable to reduce the perilune altitude to approximately 50 n.mi. for observation or experimental purposes. After the desired duration in lunar orbit has elapsed, the vehicle will be orbit-launched on the trans-earth trajectory, with injection initiated near the perilune point. Return trajectory corrections are then required to establish the correct perigee altitude for entering the re-entry corridor.

Based on propulsion requirements for the lunar orbiting and return mission, established during the Phase I study and verified during the Phase II computer trajectory analysis, outbound and return trajectories of 65 to 75 hours were selected.

b. Propulsion Requirements for Maneuvers

The maneuvers considered for the manned orbiting and return mission include (1) abort at injection; (2) outbound trajectory corrections; (3) lunar orbit maneuvers, including outbound orbit injection, perigee variation, and return orbital launch; and (4) return trajectory corrections.

(1) Abort at Injection

For abort at injection, a minimum thrust-to-mass ratio of 1.3 lbf/lbm is required to complete the abort maneuver, with a velocity increment of 7000 ft/sec. Restart and thrust variability will not be required during abort, but thrust-vector control must be included. Total-impulse accuracy will not be critical.

(2) Outbound Trajectory Corrections

The use of current booster-guidance capability with conventional radar tracking indicates that an outbound midcourse correction velocity increment of about 100 fps would bring the perilune error to about 40 n.mi. The correction should be made between 50,000 and 75,000 miles from the earth, to provide adequate tracking time for trajectory selection without excessive increase in the corrective velocity increment.

With a vehicle-borne optical terminal-guidance system for lunar approach, the indicated terminal correction velocity of about 20 to 25 fps should be made at a radius of approximately 10,000 n.mi. from the moon. This will ensure a perigee error of ± 5 n.mi. or less.

A thrust-to-mass ratio in a range between 0.025 and 0.25 lbf/lbm is acceptable for the outbound lunar corrections. A total-impulse accuracy, $\Delta I_t/m_o$, of 0.01 would be adequate for the correction maneuvers. Thrust variability is not necessary, but thrust-vector control must be provided, either by the attitude-control system of the vehicle or by gimballing of the correction engines to ensure proper orientation of the corrective velocity vector and to maintain vehicle orientation and stability.

(3) Outbound Orbit Injection

An ideal velocity-increment capability of 3300 fps was specified for a flight duration of 73 hr to accomplish injection into a 200 n.mi. lunar orbit; this requirement was verified by computer analysis. It was determined that increase in ΔV due to finite burning times is quite small for this maneuver - on the order of 20 ft/sec or less for initial thrust-to-mass ratios greater than 0.10 lbf/lbm. A total-impulse accuracy of about

0.20 lb-sec/lbm is indicated, resulting in perigee inaccuracy of 4 to 5 n.mi. A single start is required, and no thrust variability is necessary for the orbit injection maneuver.

(4) Perilune Variation

A velocity increment of 250 fps was calculated for reduction of the perilune altitude to 50 n.mi. from the 200-n.mi. circular altitude, including some allowance for contingencies. A total-impulse accuracy, $\Delta I_t/M_f$, of 0.10 will result in a perilune error on the order of 2 or 3 n.mi. Thrust-to-mass ratios between 0.10 and 0.5 lbf/lbm are indicated for this maneuver. If the cutoff accuracy is adequate, the maneuver can be completed with a single start and no thrust variability.

(5) Return Orbital Launch

A velocity-increment capability of 3200 fps was calculated for the return orbital launch on a 73-hr return trajectory. A return launch impulse accuracy of 0.10 lbf-sec/lbm is suggested, resulting in a velocity error at injection on the trans-earth trajectory of approximately 3 fps. The return orbital launch maneuver can be carried out with a single start and no thrust variability for the propulsion system.

(6) Return Trajectory Corrections

Two trajectory corrections are anticipated on the return flight for the orbiting and return mission. The first would occur at approximately 50,000 n.mi. from the earth and would require a total corrective velocity increment of 150 fps. The final correction would occur at approximately 10,000 n.mi. from the earth and would require a velocity increment of 50 fps. A total-impulse accuracy of 0.01 lbf-sec/lbm is specified for the return corrections, resulting in a velocity error of 0.3 fps. Thrust-to-mass ratios between 0.02 and 0.50 are desirable for the correction maneuvers. Two to three starts will be required for return trajectory corrections. Thrust variability is not required, but thrust-vector control must be provided either by the attitude-control system or by gimballing the correction engine.

(7) Summary of Specific Requirements

The absolute values for maneuver propulsion requirements for this mission, based on the Nova and Saturn C-3 payload capabilities, are indicated in Table 6. These values again indicate the requirements established for the individual maneuvers, with no system integration adjustments for completion of the overall mission.

c. Environmental Considerations

The outbound trajectory for the orbiting and return mission will be of 3 or 4 days duration; 2 or 3 days are anticipated in the lunar orbit; and the return trajectory will also be of 2 or 3 days duration. Thus, the propulsion systems for outbound trajectory correction, outbound orbit injection, and perilune variation will be subject to space storage for 3 to 4 days. The return orbital launch system and return trajectory correction propulsion will be subjected to storage durations in the range from 5 to 10 days. Since the orbiting and return mission consists entirely of ballistic orbital flight, each of the starts for the individual maneuvers must be made under zero-g conditions.

2. <u>Selected Concept and System Specification - Nova Orbiting and Return Mission</u>

a. Configuration and Mission Sequence

The general configuration of the selected Nova lunar orbiting and return system, alternate $3\text{-}C_1$, is presented in Figure 4. This configuration consists of a large, nearly spherical, hydrogen tank with four smaller oxygen tanks nested below. The high-thrust abort engine is centered at the rear of the vehicle, among the four tanks, with the two main engines arranged in a straight line, with some canting. The vehicle plus payload is approximately 34 ft long, with the upper interface with the manned capsule 154 in. in diameter, and the lower interface with the final launch - vehicle injection stage 220 in. in diameter.

The mission sequence for the selected $3\text{-}C_1$ alternate is as follows:

As velocity is gained for injection into the translunar trajectory, the propulsion system is pressurized to begin the abort maneuver, if necessary. If abort is necessary, the vehicle is separated from the booster and re-oriented for the proper abort thrust-vector direction. The large abort engine is fired, with the two main 10K engines operating for thrust-vector control.

If the abort maneuver is not initiated, the propulsion system is not utilized until the outbound trajectory corrections are required; at this time, a single 10K engine is used for carrying out the appropriate correction maneuvers. The abort engine could be separated during the outbound flight if desired, with a small increase in return payload capability.

Upon arrival near the perilune point on the approach trajectory, the two 10K engines are fired for injection into the lunar orbit. After the circular orbit has been established at the desired altitude, the two 10K engines will again be ignited at the desired time for the perilune reduction maneuver. The return orbital launch is also initiated with both of the 10K main propulsion units after the 2 to 4 days assumed to be spent in the lunar orbit.

Return trajectory corrections are carried out with a single 10K engine; the vehicle must be slightly canted during this maneuver with respect to the desired corrective velocity direction. After the final correction maneuver, and upon approach to the re-entry point, the propulsion system is separated and the manned capsule undergoes re-entry.

b. Tabular System Specification

A tabular specification of the characteristics of the selected propulsion system is presented in Table 5. The only shortcoming of the selected Nova orbit and return propulsion system lies in the total-impulse accuracy for return corrections. The allowable impulse error defined by the summary of maneuver requirements in Table 6 is 250 lbf-sec for the return correction; the 3σ cutoff accuracy of the selected system is 420 lbf-sec with a single engine operating. However, the required total-impulse accuracy is with a 2σ confidence-error interval for the selected system; alternately, the final trim on the return trajectory corrections could be provided by settling jets, or by jets included in the attitude-control system. Since all other characteristics of the selected system satisfy the requirements established by the mission and by the assumed launch vehicle, the specification included in Table 5 defines the recommended space-propulsion system for the Nova manned orbiting and return mission.

3. Selected Concept and System Specification - Saturn C-3 Orbiting and Return Mission

a. Configuration and Mission Sequence

The configuration of the selected 6-C_1 alternate for the Saturn C-3 orbiting and return mission is shown in Figure 5. The tankage consists of a single ellipsoidal hydrogen tank and four spherical oxygen tanks nested below it. Two 30K abort engines were utilized rather than a 60K engine as in the Nova configuration due to the fact that a 60K abort engine could not be placed in the center of the LO_2/LH_2 tank cluster without interference with the 5K main engines. The overall length of the payload plus propulsion system is 24 ft. The interface with the manned capsule is at the 154-in. diameter; this diameter is maintained over the entire vehicle length.

The sequence of operation for the selected 6-C_1 system on the Saturn C-3 orbiting and return mission is quite similar to that previously discussed for the Nova orbiting mission. The system is pressurized during injection on the translunar trajectory to allow quick re-orientation and abort if required. After the period for injection abort-capability is passed, the abort engine could be separated if desirable, with a consequent improvement in payload due to less inert weight being carried through the orbiting and return launch maneuvers at the moon. The trajectory corrections are made with a single 5K engine, with adequate total-impulse accuracy. The total 10K thrust capability of the mission propulsion engine is used for the injection into lunar orbit, perilune variation, and return orbital launch maneuver.

b. Tabular System Specification

The required tabular specification of the characteristics of the selected space propulsion system for the Saturn C-3 orbiting and return mission is presented in Table 5.

The characteristics of the selected system adequately satisfy all mission requirements specified for the Saturn C-3 orbiting and return mission in Table 6.

C. MANNED LUNAR LANDING AND RETURN MISSION

1. Mission Requirements

a. Trajectory Analysis

The manned lunar landing and return mission has been based on the concept of a landing from lunar orbit. Compared with a straight-in approach, the orbit landing procedure allows more precise determination of initial conditions for the final landing maneuvers, reconnaissance of the landing site before initiation of the landing, reduced accuracy requirements in certain parts of the landing guidance and control systems, and increased flexibility in selection of the landing area. The lunar landing and return mission based on the orbit landing approach includes the following maneuvers.

(1) Outbound trajectory corrections

(2) Lunar orbiting maneuver

(3) Perilune variation to the desired altitude for landing initiation

(4) A gravity turn continuous-thrust landing maneuver

(5) Hovering and transverse maneuvering at low altitude prior to touchdown

(6) Takeoff for a direct return to earth flight, after desired stay on the lunar surface

(7) Return trajectory corrections for trans-earth flight.

The outbound trajectory for the manned lunar landing and return mission will be identical to that for the lunar orbiting and return operation previously discussed. The perilune variation for the landing and return mission will require a decrease to about 10 n.mi. lunar altitude; this maneuver does not require a large velocity increment, but it does require high total impulse accuracy. The actual lunar landing from orbit is initiated near the perilune point; it is postulated to consist of a continuous-thrust ballistic-turn to the landing site. Although the maneuver can be completed with a nominally constant thrust, error effects will result in a requirement for some thrust level modulation. The effects of gravity loss, errors, and perilune altitude have been investigated by the use of a two-dimensional computer program for the lunar landing maneuver.

The requirement for takeoff from the lunar surface for injection on the trans-earth trajectory was based on the results of the Phase I study. The return trajectory correction requirements were based on results of analytical work similar to the work carried out to determine requirements for the circumlunar trajectory, which established the requirements to achieve a desired perigee altitude and re-entry point on return to earth. The return trajectory correction requirements were found to be quite similar to those which were verified in three-dimensions for trans-earth trajectories on the circumlunar mission.

b. Propulsion Requirements for Maneuvers

(1) Outbound Trajectory Correction and Orbit Injection

The propulsion requirements for the outbound trajectory corrections and the outbound orbit injection maneuvers are identical to those for the manned orbiting and return mission, since identical trajectories and lunar orbit altitude are postulated.

(2) Perilune Variation

The perilune variation maneuver consists of a reduction of perilune altitude to 10 n.mi. from the 200 n.mi. circular orbit. The ideal velocity increment for the perilune variation maneuver remains approximately 250 fps including contingencies. The total-impulse accuracy for the maneuver $\Delta I_t/m_f$, must be held to approximately 0.03 lbf-sec/lbm to maintain a perilune altitude tolerance of about ± 1 n.mi. Values of F/m_0 ranging from 0.1 to 0.5 lbf/lbm appear to be satisfactory for this maneuver. The maneuver will be carried out with a single start after 3 to 5 days space storage. Thrust-vector control must be provided; however, thrust variability is not required, provided that the initial thrust-to-mass ratio, considered from a total impulse accuracy standpoint, is not excessively high.

(3) Lunar Landing

Nominal propulsion requirements for landing from orbit have been based on the results achieved in Phase I of the study. A gravity-turn landing from the orbital altitude of 10 n.mi. is postulated. The required ideal velocity increment capability of 6000 fps includes some allowance for contingencies. It was determined from a landing error analysis that a thrust variability of at least 1.2:1 will be necessary. The landing will require a single start after 3 to 5 days of space storage. Thrust-vector control will be necessary to maintain vehicle attitude and stability.

(4) Hovering and Transverse Maneuvering

The hovering and transverse maneuvering requirement, during the final phase of the lunar landing, will require 1600 fps of ideal velocity increment, assuming a 3 to 5 min hovering capability with about 2000 ft of transverse maneuvering range. A thrust variability of at least 1.5 to 1 will be required for this maneuver. A single start is anticipated after 3 to 5 days of in-flight storage in space. Thrust-vector control will be necessary to maintain vehicle orientation during the hovering and transverse maneuvering functions.

(5) Lunar Takeoff

An ideal velocity increment capability of 9000 fps has been specified for the lunar takeoff maneuver. Thrust level in the range of 0.8 lbf/lbm is expected to be near optimum. A total impulse accuracy of 0.10 lbf-sec will provide a burnout accuracy of ± 3 fps. Thrust variability is not required for the takeoff maneuver, although thrust-vector control through engine gimballing will be necessary due to the relatively high thrust level involved. The takeoff maneuver will be completed with one start after 3 to 5 days of in-flight space storage and 3 to 10 days of storage on the lunar surface.

(6) Return Trajectory Corrections

The return trajectory corrections on the lunar landing and return missions are similar in all respects to those required for the orbiting and return flight. A velocity increment capability of 200 fps is required with a thrust-to-mass ratio from 0.02 to 0.25 lbf/lbm. No thrust variability is necessary, but thrust vector control must be provided either through the use of an available attitude control system or engine gimballing. The trajectory corrections will require 2 to 3 starts after 6 to 8 days of in-flight space storage and 3 to 10 days of storage on the lunar surface.

(7) Summary of Specific Requirements

Desirable specific values for maneuver propulsion requirements on the manned landing mission are presented in Table 7. The characteristics are again based on the individual maneuver requirements, with no adjustment for system integration based on the complete mission.

c. Environmental Considerations

As indicated in the previous section, in-flight space storage durations for the various maneuver propulsion systems may range from 3 to 8 days. Storage duration on the surface of the moon for the lunar takeoff and return trajectory correction systems will be from 3 to 10 days. All maneuvers will require zero-g start, except the lunar takeoff operation which will be initiated under one lunar gravity.

2. <u>Selected Concept and System Specification - Nova Single-Stage</u>

a. Configuration and Mission Sequence

The configuration for the selected Nova single-stage landing and return vehicle is presented in Figure 6. The basic configuration consists of a spherical liquid hydrogen tank with six cylindrical hydrogen and oxygen tanks forming a ring at the rear of the vehicle. The engine projects down through the center of the ring of cylindrical tanks, with thrust structure carrying loads into the vehicle skin at about station 400. Two of the cylinders contain all of the outbound liquid oxygen requirement, and a large fraction of the outbound liquid hydrogen requirement. The 60K, regeneratively-cooled main engine will just pass through the center of the ring formed by the tanks upon takeoff at the lunar surface. The "expended" tankage will form a support structure and launch pad after landing on the moon. The liquid oxygen for the return trip is contained in four spheres located above the spherical liquid hydrogen tank. The zero-g start requirement will be satisfied by the use of small settling jets or by the attitude control system to provide a small acceleration for locating the propellant. The vehicle has an overall length of 48 ft including the payload capsule, and the interface diameter with the launch vehicle is 260 in. Interface with the capsule is again located at the 154-in. diameter.

The operational sequence for the single-stage Nova lunar landing and return vehicle is as follows: after injection on the translunar trajectory, abort capability is provided by the nominal total mission propulsion capability of about 20,000 ft/sec. The outbound trajectory corrections are made with the 60K engine throttled to its minimum thrust level of 10 K.

Upon nearing the perilune point of the approach trajectory, the vehicle is injected into lunar orbit using the full 60K nominal thrust level. The landing from orbit is initiated from the perilune point, the trajectory consisting of a gravity-turn powered deceleration that reaches zero velocity several hundred feet above the lunar surface. During the landing maneuver, the variable-thrust main engine will be under closed loop control, to provide closure on zero velocity and the desired hovering altitude. Upon reaching this altitude, the engine will be throttled to its minimum thrust level, and hovering can be sustained for 3 to 5 min with thrust levels in a range from 10 to 12 K.

Upon completion of the desired lunar stay time, the vehicle is launched from the lunar surface, using the empty tanks as a launching structure. The full 60K thrust level will be utilized at this point to minimize gravity losses during the lunar takeoff. As the vehicle velocity nears the trans-earth injection velocity during lunar launch, the main engine will be throttled to its 10 K thrust level to achieve the required cutoff accuracy for the return flight. The return trajectory corrections for the Nova landing and return mission with a single stage vehicle will be made utilizing the 60K engine throttled to its minimum thrust level of 10K. Before arrival at the re-entry point near the earth, the payload capsule will be separated and will re-enter alone.

b. Tabular System Separation

The required tabular specification for the selected single-stage Nova manned landing and return vehicle is presented in Table 8. The system meets all propulsion requirements indicated in Table 7 with the exception of the return trajectory correction cutoff accuracy of 300 lb-sec. This correction can be made at approximately the 2σ confidence level using the main engine throttled to its 10K thrust level, or it can be trimmed through the use of settling jets or the attitude control system. Since the system meets all other maneuver requirements with high payload capability, it is recommended for further consideration for the manned lunar landing and return mission.

3. Selected Concept and System Specification - Nova Two-Stage

In addition to the single-stage Nova manned landing and return configuration, a two-stage Nova configuration was considered as well.

a. Configuration and Mission Sequence

The configuration for the first stage which was selected for the two-stage Nova lunar landing and return mission is presented in Figure 7. The general characteristics of this configuration include a large single liquid hydrogen tank with a short cylindrical section for better utilization of the vehicle envelope. Four cylindrical LO_2 tanks are nested beneath the hydrogen tank, with the 50K regeneratively-cooled, pump-fed main engine centered among them, and the four 3.5K ablation-cooled vernier engines between their bottom extremities. The overall stage length is approximately 35 ft between separation planes. The nominal diameter of the stage is 220 in., with a tapered second stage interface at 198 in. diameter.

The selected LO_2/LH_2 second stage alternate, $2\text{-}A_{10}$, is illustrated in Figure 8. The pressurized tankage for this configuration also consists of a single, nearly-spherical, LH_2 tank, with four cylindrical LO_2 tanks nested beneath it. The two fixed 25K ablative main engines protrude upward between the cylindrical tanks, and the two gimballed 2K engines are located below the cylindrical tanks with adequate room for gimbal operation. The second stage, alternate $2\text{-}A_{10}$, is 35 ft long, measured from the top to the separation plane. This stage, with the first stage $2\text{-}A_{11}$, results in a combined length, for the lunar landing and return stages, of approximately 70 ft.

The configuration of the storable second stage alternate, $2\text{-}A_{12}$, is illustrated in Figure 9. This configuration includes an ellipsoidal oxidizer tank, with two near-spherical fuel tanks. The two 25K main engines are again projected beside the tanks. The 2K vernier engines are located beneath the tanks and 90° out of the main engine plane. All engines are ablatively cooled and pressure fed. The stage length to the separation plane, including the payload, is approximately 28 ft. The overall length for the combined first and second stage is approximately 63 ft. The nominal stage diameter is maintained at 154 in., except for the flared skirt which completes the interface with the first stage alternate $2\text{-}A_{11}$ at a diameter of 198 in.

The operational sequence for the two-stage lunar landing and return mission is described below. This sequence applies to two-stage vehicles made using either the LO_2/LH_2 upper stage or the storable second-stage configuration.

The abort maneuver capability for the two-stage Nova manned landing and return vehicle is provided by the 20,000 ft/sec of velocity capability which is necessary for the nominal mission. The outbound trajectory corrections are completed using 3.5K ablative verniers in the first stage. The outbound orbit injection maneuver is powered by the single 50K engine; the four ablative verniers providing thrust vector control. The perilune variation, prior to the soft lunar landing, can be accomplished with the four 3.5K engines, or with two used alone, to provide the required total impulse accuracy of better than 3330 lbf-sec. Landing from orbit is initiated with the 50K main engine and the 3.5K verniers providing the required variability to correct for initial perilune errors, and guidance and control variations. The gravity-turn landing maneuver is completed at an altitude of several hundred feet above the lunar surface, with effectively zero vertical and transverse velocities. At this point, the thrust level is reduced by cutting the main 50K engine. The four 3.5K variable-thrust verniers then provide control thrust, in the 10 to 15 K total range, to allow hovering and transverse maneuvering.

For the lunar takeoff maneuver, the second-stage vehicle is launched directly by separation from the first stage. Since the two 25K main engines may not be able to provide the necessary total impulse accuracy, for the lunar takeoff maneuver, this velocity can be trimmed by use of the 2K ablative vernier chambers alone. The total impulse accuracy of a single 2K engine will be adequate to complete the return trajectory corrections within the specified total impulse accuracy of 300 lbf-sec. Upon approaching the re-entry point, after the final re-entry trajectory correction has been made, the manned capsule separates and re-enters alone.

b. Tabular System Specification

The tabular specification of the characteristics of the selected propulsion systems for the two-stage manned lunar landing and return missions are presented in Table 8. Since each of the stages can adequately satisfy the appropriate requirements specified by the maneuver summary, Table 7, the selected systems are recommended, since they are entirely capable of carrying out the landing and return mission.

VII. UNMANNED 24-HOUR SATELLITE MISSION

The analysis of the 24-hour satellite mission was divided into two basic parts. First, the propulsion requirements were established; second, the competitive systems were compared for three payloads and the best system for each payload was specified.

A. MISSION REQUIREMENTS

Propulsion requirements were determined for three basic operations which the satellite propulsion system will be required to perform. These operations are: (1) orbit correction for the elimination of injection errors and for the achievement of the desired longitudinal position, (2) station keeping, and (3) attitude control. Table 9 presents the summary of propulsion requirements for the 24-hour satellite, based on the propulsion requirements for these areas.

1. Correction of Injection Errors

The propulsion requirements for correction of injection errors and for achievement of a desired longitudinal position are as follows:

a. The velocity increment will vary between approximately 100 ft/sec and 450 ft/sec.

b. The minimum thrust-to-mass ratio will be about 4×10^{-4} lbf/lbm; this will increase if maneuver times to achieve the correct longitude are required to be less than one month. The maximum expected value can be as large as 0.2 lbf/lbm.

c. The total impulse-to-mass ratio will vary between 2.0 lbf-sec/lbm and 15 lbf-sec/lbm.

d. Accurate control of total impulse for each correction will be required, since any errors remaining after cut-off of the orbit correction system will have to be corrected by the station-keeping system.

e. The system must have restart capabilities for correction of the in-plane errors.

f. Thrust modulation will not be required if variable burning times are employed.

g. An attitude control system must be available for the correction maneuvers, to provide thrust vector control and to correct any thrust misalignment which may be present.

h. The operational duration of the system used for orbit corrections will range from approximately one week to about one month.

2. Station Keeping

The propulsion requirements for performance of station keeping were determined, and these may be summarized as follows:

a. The velocity increment required will generally range from 15 ft/sec. to 70 ft/sec. for one year operation. An additional maximum increment of 8 ft/sec will be required for each additional year of operation. If angular position tolerances of less than $\pm 0.25^{\circ}$ for a one-year life or $\pm 0.50^{\circ}$ for a two-year life are required for the orbit plane inclination, there will be an additional requirement of approximately 26 ft/sec per year. Therefore, the total requirement could be as high as 95 ft/sec for a one-year life or 130 ft/sec for a two-year satellite life.

b. Thrust-to-mass rotation can vary within a wide range, but for typical propulsion parameters will probably lie in the range from 2×10^{-4} to 10^{-2} lbf/lbm.

c. The required total impulse-to-mass ratio will be in the range between 0.3 and 3.0 lbf-sec/lbm for a one-year life with an additional maximum requirement of 0.25 lbf-sec/lbm for each additional year. If the out-of-plane correction is necessary, then an additional 1.0 lbf-sec/lbm per year will be required. Therefore, the maximum requirement for a two-year life is about 3.25 lbf-sec/lbm.

d. If a convergent correction scheme is specified, the allowable tolerance must be greater than ± 0.5° for out-of-plane motion and about ± 10 to ± 20 for in-plane motion for a two-year life.

e. Maximum possible accuracy should be achieved in total impulse control since the total impulse requirement is a function of the propulsion system accuracy.

f. Multiple restart capability will be arranged.

g. Thrust modulation will not be required.

h. An attitude control system must be available for thrust vector control and for offsetting any thrust misalignment which may be present.

i. The operational duration of the station-keeping correction system will range from a minimum of two months to a maximum of about two years.

3. Attitude Control Requirements

The following table summarizes the total impulse and thrust requirements for the attitude control system.

a. Total Impulse (2-Year Life)

	Total Impulse (lbf-sec)		
	Centaur	Saturn C-2	Saturn C-3
Solar pressure	300	2,000	3,000
Gravity gradient	---	---	---
Thrust Misalignment	180	1,300	6,300
Meteorite Impact	---	---	---
Initial Rates	1	15	90
Undisturbed Limit Cycle	660	12,300	25,000
	1,141	15,615	84,390

b. Thrust

	Thrust(lb)		
	Centaur	Saturn C-2	Saturn C-3
Solar Pressure	10^{-5} to 10^{-3}	10^{-2} to 10^{-4}	$5x10^{-3}$ to 0.5
Gravity gradient	---	---	---
Thrust Misalignment	10^{-3} to 0.1	$5x10^{-3}$ to 0.5	0.03 to 3.0
Meteorite Impact	---	---	---
Initial Rates (Optimum Valves)	1.16	21.0	130.0
Undisturbed Limit Cycle	0.015	0.33	2.0

The additional requirements of the attitude control system include: restartability, maximum total impulse accuracy, and an operational duration in the space environment of about two years.

B. SYSTEM SELECTION AND SPECIFICATION

The various propulsion and control systems applicable to the 24-hour orbit satellite operation were reviewed. Some of the systems considered are: cold gas, monopropellant and bipropellant reaction-jet systems, and special systems such as reaction wheels for attitude control and evaporation or sublimation jets. Subsystems such as tankage, positive expulsion methods, and attitude sensors were also reviewed. Based on the propulsion requirements summarized in Table 9 the selected systems include cold gas, liquid bipropellant and reaction-wheel systems.

1. Specification of the Integrated System - Centaur

The selected propulsion system is a dual system utilizing N_2O_4/Aerozine-50 as propellants for the combined orbit correction, station keeping operation with a total impulse of about 5000 lb-sec, and reaction wheels for attitude control augmented by cold-gas jets. The general configuration and arrangement of nozzles for this system is shown in Figure 10.

The mode of operation will be to employ an optional thrust-pulsing system as follows: During the large initial orbit corrections and the correction to achieve the desired longitude, the system will not operate using the pulsing method, but will employ conventional, continuous thrust monitored by low-g accelerometers to provide accurate thrust termination. For the station keeping operation, however, thrust pulsing will be used; i.e., the system will switch to pulsing operation, providing a variable number of specified minimum-total-impulse pulses. With each correction, fewer pulses are required as finer accuracy is achieved, until in the final limit cycle, one or a very small number of pulses are required to reverse the drift of the satellite at each edge of the allowable error cone.

A final system weight breakdown is given in Table 10 and a summary of the tabular propulsion system specifications is presented in Table 11.

2. Specifications of the Integrated System - Saturn C-2

The selected system utilizes N_2O_4/Aerozine-50 pressure-fed propellants for combined orbit correction, station keeping, and jet augmentation of the reaction wheels used for attitude control. The system has a total impulse of approximately 107,000 lbf-sec. The mode of operation is essentially the same as the one described for the Centaur payload. That is, an optional pulse system is specified - one in which conventional thrusting is employed during orbit correction and attitude control, and pulsing operation is used for station keeping to obtain increased accuracy. The selected system, as for the Centaur case, employs redundant engines for in-plane orbit correction, station keeping, and attitude control functions. The number and arrangement of the engines are the same as for the Centaur payload as shown in Figure 10.

3. Specification of the Integrated System - Saturn C-3

The mode of operation and system characteristics for the Saturn C-3 selected propulsion system are essentially the same as that for the Saturn C-2 selected system, except for size effects. The final system weight breakdown is presented in Table 10, and the complete system specification is summarized in Table 11.

TABLE 1

SUMMARY OF SPACE-PROPULSION REQUIR

Maneuver	BASIC MISSION REQUIREMENTS: Range of Ideal ΔV Requirements ft/sec	Range of Desirable F/M_o, lbf/lbm	Cutoff Impulse Accuracy: Typical $\Delta I_t/I_t$	Cutoff Impulse Accuracy: Allowable $\Delta I_t/M_f$ lbf-sec/lbm	Required Thrust Variability	Restart Requirements	Thrust Vector Control	Storability Requirements	REPRES… Large Pa…: M_{p1}, lbm	M_o,
A. ORBITAL CORRECTION										
1. Orbital Perturbations										
a. Atmospheric Drag	--	--	--	--	None	Multiple	Use attitude control	1 day - weeks	--	--
b. Earth Oblateness Effects	100-4000	0.05-1.0	0.015	1.5	None	Multiple	At higher acceleration	Days - months	36,000	40,000
2. Eccentricity Control	100-5000	0.05-1.0	0.02	2.5	None	None	At higher acceleration	Hours - months	36,000	40,000
3. Orbital Plane Change	200-19,000	0.15-2.0	0.002	1.0	None	None	At higher acceleration	Hours - months	36,000	40,000
4. Orbital Altitude Variation	200-14,000	0.1-2.0	0.002	1.0	None	1-2	At higher acceleration	Hours - months	36,000	40,000
5. Orbital Epoch Change	100-20,000	0.1-2.0	0.002	0.5	None	1-2	At higher acceleration	Hours - months	36,000	40,000
6. Correction of Injection Errors	50-1000 (max)	0.01-0.5	0.001	0.03	None	1-2	Use attitude control	1-5 days	37,600	40,000
B. ORBITAL RENDEZVOUS										
1. Nominal Injection Errors	50-1000	0.01-1.5	--	.15-.5	System Variability to 100 max	0-2	Use attitude control	0-5 days	42,000	45,000
2. Dog-Leg Maneuver (coarse) / (fine)	500-18,000	1.0-3.0 / .01-1.5	-- / --	-- / .15-.5	System Variability to 100, max	1-2	At higher acceleration	0-5 days	20,700	45,000
3. Emergency Rendezvous	1000-25,000	.01-3.0	--	.15-.5	System Variability to 100	1-2	At higher acceleration	0-1 day	10,000	25,500
C. TRAJECTORY CORRECTIONS										
1. Midcourse Corrections										
a. Lunar Flights	25-250	.025-.25	.003	.02	None	3 max	Use attitude control	70 hours max	98,500	100,000
b. Planetary Flights (Mars - Venus)	50-1000	.025-.50	.003	.05	None	6 max	Use attitude control	250 days max	[illegible]x10^6	3,000,0
c. Planetary Return Flights (Mars - Venus)	50-1000	.025-.50	.003	.05	None	6 max	Use attitude control	2-3 years max	125,000	130,000
2. Terminal Corrections										
a. Lunar Flights	25-500	.020-0.50	.005	.05	None	2 max	Use attitude control	2-3 days max	98,500	100,000
b. Planetary Flights (Mars)	100-1000	.020-1.0	.010	0.3	None	3 max	Use attitude control at higher accelerations	250 day max	2.97x10^6	3,000,0
c. Return Flights:										
(1) Moon	50-500	.015-1.0	.020	.15-.30	None	3 max	At higher accelerations	1 day - several mo	19,300	20,000(4
(2) Mars	200-1500	.03-1.0	.01	.1-.25	None	3 max	At higher accelerations	2-3 years max	100,000	130,000
D. ORBITING MANEUVERS										
1. Moon Orbits	2000-5500	1.0-2.0	.002	.15-1.0	None	None	Yes	2-3 days max	72,500	100,000
2. Mars Orbits (1) No atmos Dec.	5000-20,000	1.0-3.0	.002	.15-2.0	None	None	Yes	250 days max	1.5x10^6	3,000,00
2. Mars Orbits (2) With atmos Dec.	≈1000	0.5-1.0	.004	.03	None	5-10	Use attitude control	250 days max	2.8x10^6	3,000,00
E. LANDINGS										
1. Lunar Landings										
a. Direct	9600-9900	1.0	--	--	[illegible]:1	None	Yes	1-2 days	43,000	95,000(2
b. From Orbit	5700-6000	1.0	--	--	[illegible]:1	1	Yes	1-2 days	45,000	72,000(2
2. Mars Landing										
a. Direct	13,000-21,000	1.0-2.0	--	--	1:1	0-1	Yes	250 days	900,000	3,000,00
b. From Orbit	11,000-15,000	2.0-4.0	--	--	10:1	1	Yes	250 days	25,000	125,000(
F. TAKEOFFS										
1. Lunar Takeoffs										
a. To Orbit	6000-6500	1.0-1.6	.001	.15	None	1	Yes	Several weeks	21,200	40,000(2
b. Direct to Earth	9000-11,000	1.0-1.5	.001	.30	None	None	Yes	Several weeks	1[illegible],600	40,000(2
2. Mars Takeoffs										
a. To Orbit	15,000-17,000	0.7-1.0	.0005	.30	None	1	Yes	Several years	4000	25,000(5
b. Direct to Earth	20,000-35,000	--	.0005	.30	None	None	Yes	Several years	--	--
1st Stage / 2nd Stage	--	1.5-2.0 / 1.0-2.0	--	--	--	--	--	--	440,000 / 150,000	1,000,00 / 440,000(

NOTE: (1),(2),(3) Initial mass for maneuver roughly approximates mission based on Saturn, Nova, and Centaur launch capabilities, respectively.

(4) Initial mass for maneuver assumes vehicle roughly sized by 50,000-lb capsule weight returned to earth.

(5) Initial mass for maneuver assumes vehicle roughly sized by 50,000-lb craft landed on Mars from parking orbit.

TABLE 1

SUMMARY OF SPACE-PROPULSION REQUIREMENTS

MISSION REQUIREMENTS				REPRESENTATIVE SYSTEM CHARACTERISTICS								LIQUID-PROPELLANT SYSTEM REQUIREMENTS	
				Large Payload (Manned or Unmanned)				Small Payload (Unmanned)					
Required Thrust Variability	Restart Requirements	Thrust Vector Control	Storability Requirements	M_{pl}, lbm	M_o, lbm	F, lbf	I_t, lbf/sec	M_{pl}, lbm	M_o, lbm	F, lbf	I_t, lbf/sec	Maximum Cutoff F/m, lbf/lbm	Vernier Cutoff Required
None	Multiple	Use attitude control	1 day - weeks	--	--	--	--	--	--		--		
None	Multiple	At higher acceleration	Days - months	36,000	40,000(1)	5000	$1.2x10^6$	7100	8000(3)	1000	$.24x10^6$	Not Restrictive	No
None	None	At higher acceleration	Hours - months	36,000	40,000(1)	20,000	$1.2x10^6$	7100	8000(3)	4000	$.24x10^6$	Not Restrictive	No
None	None	At higher acceleration	Hours - months	36,000	40,000(1)	20,000	$1.2x10^6$	7100	8000(3)	4000	$.24x10^6$	Not Restrictive	No
None	1-2	At higher acceleration	Hours - months	36,000	40,000(1)	20,000	$1.2x10^6$	7100	8000(3)	4000	$.24x10^6$	Not Restrictive	No
None	1-2	At higher acceleration	Hours - months	36,000	40,000(1)	20,000	$1.2x10^6$	7100	8000(3)	4000	$.24x10^6$	Not Restrictive	No
None	1-2	Use attitude control	1-5 days	37,600	40,000(1)	5000	$.62x10^6$	7500	8000(3)	6000	$.12x10^6$	1.0	No
System Variability to 100 max	1-2	Use attitude control	0-5 days	42,000	45,000(1)	22,500	$.84x10^6$	7900	8500(3)	4250	$.16x10^6$	--	--
System Variability to 1000 max	1-2	At higher acceleration	0-5 days	20,700	45,000(1)	13,500	$7x10^5$	3700	8500(3)	25,500	$1.3x10^6$	--	--
System Variability to 1000	1-2	At higher acceleration	0-1 day	10,000	25,500(1)	70,000	$5.1x10^6$	2000	4650(3)	14,000	$1x10^6$	--	--
None	1 max	Use attitude control	70 hours max	98,500	100,000(2)	5000	$.31x10^6$	2470	2500(3)	125	7750	.50	No
None	6 max	Use attitude control	250 days max	$2.55x10^6$	3,000,000(4)	150,000	$46x10^6$	9470	10,000(1)	500	$.15x10^6$	1.0	No
None	6 max	Use attitude control	2-3 years max	125,000	130,000(4)	6500	$2.1x10^6$	2840	3000(1)	150	45,600	1.0	No
None	2 max	Use attitude control	2-3 days max	98,500	100,000(2)	5000	$.31x10^6$	2470	2500(3)	125	7750	.75	No
None	3 max	Use attitude control at higher accelerations	250 day max	$2.97x10^6$	3,000,000(4)	150,000	$9.3x10^6$	9880	10,000(1)	500	31,000	5.0	No
None	3 max	At higher accelerations	1 day - several mo	19,500	20,000(2)	1000	62,000	480	500(3)	25	1550	5.0	No
None	4 max	At higher accelerations	2-3 years max	130,000	130,000(4)	20,000	$3.0x10^6$	2860	3000(1)	150	9300	2.0	No
None	None	Yes	2-3 days max	72,500	100,000(2)	100,000	$11x10^6$	1550	2500(3)	2500	$.26x10^6$	3.0	Yes (critical cases)
None	None	Yes	250 days max	$1.5x10^6$	3,000,000(4)	6,000,000	$620x10^6$	4200	9000(1)	18,000	$1.6x10^6$	8.0	No
None	0-1	Use attitude control	250 days max	$2.8x10^6$	3,000,000(4)	3,000,000	90×10^6	7900	9000(1)	9000	$.28x10^6$	0.5	No
6:1	None	Yes	1-2 days	43,000	95,000(2)	95,000	$21x10^6$	700	2300(3)	2300	$.45x10^6$	--	--
6:1	1	Yes	1-2 days	45,000	72,000(2)	72,000	$11x10^6$	740	1500(3)	1500	$.21x10^6$	--	--
10:1	0-1	Yes	250 days	900,000	3,000,000(4)	6,000,000	$350x10^6$	1450	9000(1)	18,000	$2.2x10^6$	--	--
10:1	1	Yes	250 days	25,000	125,000(5)	500,000	$25x10^6$	630	3500(1)	14,000	$.79x10^6$	--	--
None	1	Yes	Several weeks	21,200	40,000(2)	40,000	$5.6x10^6$	1500	3000(1)	3000	$.42x10^6$	3.0	Yes (critical cases)
None	None	Yes	Several weeks	17,600	40,000(2)	60,000	$7.3x10^6$	930	3000(1)	4500	$.59x10^6$	6.0	No
None	1	Yes	Several years	4000	25,000(5)	37,500	$6.2x10^6$	1600	10,000	15,000	$2.4x10^6$	6.0	Yes (critical cases)
None	None	Yes	Several years	--	--	--	--	--	--	--	--	--	--
--	--	--	--	440,000 130,000	1,300,000(4) 440,000(4)	2,000,000 440,000	$220x10^6$ $130x10^6$	11,000 2000	33,000 11,000	66,000 22,000	$6.5x10^6$ $2.6x10^6$	-- 6.0	-- Yes (critical cases)

s, and

returned to earth.

Mars from parking orbit.

Table 1

TABLE 2

PROPULSION SYSTEM COMPAR

SYSTEM PERFORMANCE CHARACTERISTICS

Propulsion System	Theo. Vacuum[1] I_{sp}, lbf-sec/lbm	m_p/m_t max.	m_p/m_t t = 100 F = 1000	m_p/m_t t = 100 F = 50,000	m_p/m_t t = 500 F = 500,000	m_p/m_t Misc. Pts.	ΔI_t/F Shutdown	Thrust-V Contr
I. LIQUID BIPROPELLANTS	Higher I_{sp} for liquid propellants if solid additives used						OVER WIDE RANGE OF THRUST LEVELS AND SYSTEMS, 1 σ VARIATION IS: $\frac{\Delta It}{F} = \log_{10}\frac{(F/.0707)}{367}$; 3 σ VARIATION IS: $\frac{\Delta It}{F} = \log_{10}\frac{(F/.144)}{116}$	
A. CRYOGENICS	Present - 430 (LO_2/LH_2)							
	Future - 475 (F_2/H_2)							
1. Pump-Fed	Loss in gas generator							
a. Regenerative (some film-cooling)	Loss in film cooling	.965	.65	.86	.96			A[6]
b. Ablative			.66					A
c. Radiation								A
d. Film	Some I_{sp} loss							A
e. Transpiration	Some I_{sp} loss							A
2. Pressure-Fed								
a. Regenerative (some film-cooling)	Loss in film cooling		.58	.88				A
b. Ablative			.59	.91				A
c. Radiation			.73					A
d. Film	Some I_{sp} loss							A
e. Transpiration	Some I_{sp} loss							A
B. STORABLE BIPROPELLANTS	Present - 310 (N_2O_4/Aerozine-50)							
	Future - 360-380 (tripropellants)							
1. Pump-Fed	Loss in gas generator							
a. Regenerative (some film-cooling)	Loss in film cooling	.98	.80	.91	.975			A
b. Ablative			.81					A
c. Radiation								A
d. Film	Some I_{sp} loss							A
e. Transpiration	Some I_{sp} loss							A
2. Pressure-Fed								
a. Regenerative (some film-cooling)	Loss in film cooling		.84	.90				A
b. Ablative			.85	.94				A
c. Radiation			.80					A
d. Film	Some I_{sp} loss							A
e. Transpiration	Some I_{sp} loss							A
C. STORABLE MONOPROPELLANTS	Present - 269 (Cavea-B)							
	Future - 330							
1. Pump-Fed	Loss if used in gas generator							
a. Ablative		>.96	.85					A
b. Radiation								A
2. Pressure-Fed								
a. Ablative		>.92	.88					A
b. Radiation		.98	.85					A
II. PULSE ENGINE (Storable Prop. Assumed)	Present - 310			-	-		5-6% Impulse Bit Variation Between Cycles	A
	Future - 360-380	.976	.957 (I_t=10^5)					
III. SOLID PROPELLANTS	Present - 290 ($NH_4ClO_4/Al,-CH_2$)				Not possible now Eventually .975		.001-.003	Moveable B fluid inje plug nozz
	Future - 330 ($NO_2ClO_4/LiAlH_4,-CH_2NF_2$)	.975	.90	.952				
IV. HYBRID (Storable Liquid Oxidizer)	Present - 310 ($N_2O_4/Al,-CH_2$)					.85 - .90 for I_t~260,000	Should be Similar to Liquid System	B Same as so
	Future - 380 ($-NF_2/Li,LiH$)	-	-	-	-			
V. NUCLEAR-HEAT-TRANSFER (H_2 Propellant)	800 - 1200						No data. Probably main Shutdown poor but can use coolant for Vernier	
A. HIGH PRESSURE, PUMP-FED		.925	-	-	.87			B Same as so
B. LOW PRESSURE								
1. Pump-Fed						.77 (I_t = 11.4 x 10^6		B Same as so
2. Pressure-Fed								B Same as so
VI. ELECTRIC								
A. ION	2000 - 100,000							A
				All-Electric System Propellant Fractions are Low			Should be about the same as liquids.	
B. COLLOID	About same as Ion							A
C. ARC-JET	Up to 2000							B May be sm small siz
D. PLASMA	1000 - 20,000							A

NOTES: (1) Based on 50:1 expansion ratio. "Future" performance values represent presently foreseen limiting values. In many cases, one or more of the propellant constituents have only been hypothesized.

(2) Rating assumes no settling acceleration. All liquid systems may be uprated to B by use of an auxiliary settling rocket.

(3) Rating primarily considers penetration of tanks.

(4) Assumes satisfactory lubricant available.

(5) Burning times for steady-state cooling systems are essentially unlimited; however, firing hours are generally undesirable due to system reliability considerations.

ABLE 2

YSTEM COMPARISON

3σ VARIATION IS: $\frac{\Delta It}{F} = \log_{10}\frac{(F/.144)}{116}$

t tdown	Thrust-Vector Control	Thrust-Level Control	Restart	Storability	Zero g Effects(2)	Meteoroids(3)	Vacuum Environment(4)	Thrust Limits	Burning Time Limits(5)	Remarks
		CONTROL AND OPERATING CONSIDERATIONS								
	A(6)	A 2:1 B 5:1 C>5:1	A	C	C	B	A	A	Hours	
	A	A (>15:1)	B May be limited	C	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	C	C	B	A	≈1000 lbf due to size	Hours	
	A	A (>3:1)	A	C	C	B	A	A	Hours	
	A	A	A	C	C	B	A	A	Hours	May be problems in pore plugging
	A	A 2:1 B 5:1 C>5:1	A	C	C	B	A	A	Hours	
	A	A (>15:1)	B May be limited	C	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	C	C	B	A	≈1000 lbf due to size	Hours	
	A	A (3:1)	A	C	C	B	A	A	Hours	
	A	A	A	C	C	B	A	A	Hours	May be problems in pore plugging
	A	A 2:1 B 5:1 C>5:1	A	A	C	B	A	A	Hours	
	A	A (>15:1)	B May be limited	A	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	A	C	B	A	≈1000 lbf due to size	Hours	
	A	A (>3:1)	A	A	C	B	A	A	Hours	
	A	A	A	A	C	B	A	A	Hours	May be problems in pore plugging
	A	A 2:1 B 5:1 C 5:1	A	A	C	B	A	A	Hours	
	A	A (>15:1)	B May be limited	A	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	A	C	B	A	≈1000 lbf due to size	Hours	
	A	A (>3:1)	A	A	C	B	A	A	Hours	
	A	A	A	A	C	B.	A	A	Hours	May be problems in pore plugging
	A	A (>15:1)	B May be limited	A	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	A	C	B	A	≈1000 lbf due to size	Hours	
	A	A (>15:1)	B May be limited	A	C	B	A	May be size problem at low P_c	Minutes	
	A	A (>15:1)	A	A	C	B	A	≈1000 lbf due to size	Hours	
mpulse riation n Cycles	A	A (>15:1)	A	A (Unless cryogenic propellants used)	C	B	A	1000 lbf due to size	Hours	
003	B Moveable nozzle fluid injection plug nozzle	C May be accomplished by plug noz. acoustic energy, cooled tubes, two different grains	C	A May be radiation and vacuum problems	A	B	B May not be prob. if well sealed	A	Seconds (100 sec without nozzle cooling)	
be r to System	B Same as solids	A 100% control with hyperg. propellants	A Good if hypergolic	A Same as solids	C	B	B Same as solids	A	Minutes	
a. ly main en poor n use k for r	B Same as solids	Many control probl. May be limit on variability	Limited by time lag and phys. prop. of reactor material	C	C	B	A	A	Hours	
	B Same as solids	B	B Same as above	C	C	B	A	B Limit may be 1000-10,000	Days	
	B Same as solids	B	B Same as above	C	C	B	A	B	Days	
	A	A	A	A	C	A	A	C	Days	
be the same uids.	A	A	A	A	C	A	A	C	Days	
	B May be easy in small sizes	A	A	C If cryogenic	C	C	A	B	Hours-Days	
	A	A	A	C If cryogenic	C	C	A	B	Days	

nlimited; however, firing durations of over several erations.

(6) A - Good, easy
B - Fair, some difficulty
C - Poor, difficult

2

Table 2

TABLE 3

CATEGORIZATION OF PROPULSION REQUIREMENTS

REPRESENTATIVE REQUIREMENTS

Mission	Thrust Level, Lbf Unmanned/Manned	Thrust Variability	Restarts	Total Impulse Lbf-sec x 10^{-6} Unmanned/Manned	Applicable Classifications
A. ORBITAL CORRECTION					
1. Orbital Perturbations					
a. Atmospheric Drag	0-20	None	Multiple	.001-1/day	--
b. Earth Oblateness Effects	1000/5000		Multiple	0.24/1.2	I/II
c. Eccentricity Control	4000/20,000		None	0.24/1.2	II/II
d. Orbital Plane Change	4000/20,000		None	0.24/1.2	II/II
e. Orbital Altitude Variation	4000/20,000		1-2	0.24/1.2	II/II
f. Orbital Epoch Change	4000/20,000		1-2	0.24/1.2	II/II
g. Correction of Injection Errors	1000/5000	None	1-2	0.12/0.62	I/II
B. ORBITAL RENDEZVOUS					
1. Nominal Injection Errors	4250/22,500	10:1	0-2	0.16/0.84	I/I, III
2. Dogleg Maneuvers	25,500/135,000	10:1	1-2	1.3/7.0	III/III
3. Emergency Rendezvous	14,000/70,000	10:1	1-2	1.0/5.1	III/III
C. TRAJECTORY CORRECTIONS					
1. Midcourse Corrections					
a. Lunar Flights	125/5000	None	3 max	0.008/0.31	I/II(?)
b. Planetary Flights (Mars, Venus)	500/150,000		6 max	0.15/46	I/III
c. Planetary Return Flights (Mars, Venus)	150/6500		6 max	0.05/2.1	I/II(?)
2. Terminal Corrections					
a. Lunar Flights	125/5000		2 max	0.008/0.31	I/II(?)
b. Planetary Flights (Mars)	500/150,000		3 max	0.03/9.3	I/III
c. Return Flights (Moon)	25/1000		3 max	0.002/0.06	I(?)/I
Return Flights (Mars)	150/20,000	None	3 max	0.009/3.0	I/III
D. ORBITING MANEUVERS					
1. Moon Orbits	2500/100,000	None	None	0.26/11	II/III
2. Mars Orbit (No atmos dec.)	18,000/6,000,000		None	1.6/620	III/IV
Mars Orbit (With atmos dec.)	9000/3,000,000	None	5-10	.28/90	II/IV
E. LANDINGS					
1. Lunar Landings					
a. Direct	2300/95,000	6:1	None	0.45/21	I/(?)/III
b. From Orbit	1500/72,000	6:1	1	0.21/11	I(?)/III
2. Mars Landing					
a. Direct	18,000/6,000,000	10:1	0-1	2.2/850	III/III, IV
b. From Orbit	14,000/500,000	10:1	1	0.79/28	III/III
F. TAKE-OFFS					
1. Lunar Takeoffs					
a. To Orbit	3000/40,000	None	1	0.42/5.6	II/III
b. Direct to Earth	4500/60,000		None	0.59/7.8	II/III
2. Mars Takeoffs					
a. To Orbit	15,000/37,500		1	2.4/6.2	II/III
b. Direct to Earth					
1st Stage	66,000/2,000,000		None	6.5/220	III/IV
2nd Stage	22,000/880,000	None	None	2.6/130	II/IV

Propulsion-Capability Classifications

Class	Propulsion-Capability Classifications
I	Nominal thrust: 100-1000 lbf Multiple restart capability Variable impulse per unit time No thrust-vector control Accurate total impulse control Total impulse: 0.01-0.20 x 10^6 lbf-sec
II	Nominal thrust: 200-20,000 lbf Total impulse: .20-2.0x10^6 lbf-sec Multiple restart, No thrust variability, Thrust vector control Normal total impulse
III	Nominal Thrust 20-100,000 lbf and variable (10:1) Total impulse 2.0-20 x 10^6 lbf-sec Multiple restart capability Thrust-vector control Accurate total impulse control by variability
IV	Nominal thrust: 1-6 x 10^6 lbf, not variable Total impulse: 100-1000 x 10^6 lbf-sec Multiple restart capability Thrust-vector control Normal total-impulse accuracy

Table 3

MANEUVER PROPULSION REQUIREMENTS SUMMARY

MANNED CIRCUMLUNAR MISSIONS

Vehicle/Maneuver	Ideal ΔV Capability ft/sec	Nominal[1] Initial Mass, m_o lbm	Nominal[1] Thrust, F lbf	Nominal[1] Total Impulse, I_t lbf-sec $\times 10^{-6}$	IMPULSE ACCURACY $\Delta I_t/m_f$ $\frac{\text{lbf-sec}}{\text{lbm}}$	IMPULSE ACCURACY ΔI_t lbf-sec	IMPULSE ACCURACY $\Delta I_t/F$, sec	GENERAL REQUIREMENTS Thrust Variability (Overall Maneuver)	GENERAL REQUIREMENTS No. of Starts	GENERAL REQUIREMENTS Space Storage
NOVA MISSION										
Abort at Injection	5000	20,000	50K	2.4	-	Not critical	-	-	1	-
Outbound Trajectory Corrections	150	150,000	10K	0.7	0.01	1500	.15	-	2-3	3-4 day
Return Trajectory Corrections	600	145,000	10K	2.7	0.01	1500	.15	-	2-3	5-7 day
Totals (Excluding Abort)	750			3.4						
SATURN C-3 MISSION										
Abort at Injection	5000	20,000	50K	2.4	-	Not critical	-	-	1	-
Outbound Trajectory Corrections	150	40,000	3K	0.2	0.01	400	.13	-	2-3	3-4 day
Return Trajectory Corrections	600	37,500	3K	0.75	0.01	400	.13	-	2-3	5-7 day
Totals (Excluding Abort)	750			0.95						
SATURN C-2 MISSION										
Abort at Injection[2]	-	-	-	-	-		-	-	-	-
Outbound Trajectory Corrections	150	15,000	1K	0.07	0.01	150	.15	-	2-3	3-4 day
Return Trajectory Corrections	600	14,000	1K	0.27	0.01	150	.15	-	2-3	5-7 day
Totals (Excluding Abort)	750			0.34						

[1] The nominal initial mass, thrust, and total impulse are approximate preliminary values based on mission requirement and assumed specific impulses for the maneuver. Final values are dependent on configuration, structure weights, actual specific impulse, jettison weights, thrust-level optimizations, etc.

[2] The Saturn C-2 payload capability is not adequate to include abort propulsion for the 12,500-lbm minimum payload.

SELECTED PROPULSION-SYSTEM SPECIFICATIONS

		CIRCUMLUNAR SINGLE STAGE NOVA	CIRCUMLUNAR SINGLE STAGE SATURN C-3	CIRCUMLUNAR SINGLE STAGE SATURN C-2	LUNAR ORBITING AND RETURN SINGLE STAGE NOVA	LUNAR ORBITING AND RETURN SINGLE STAGE SATURN C-3
TOTAL-IMPULSE REQUIREMENT, lbf-sec		3.68×10^6	0.98×10^6	0.37×10^6	18.91×10^6	6.88×10^6
THRUST:	Nominal Thrust Level, lbf	4-2K	2-2K	2-1K	2-10K Main 1-100K Abort Only	2-5K Main 2-30K Abort Only
	Type of Thrust Control	Not required	Not required	Not required	Not required	Not required
	Control Range	-	-	-	-	-
	Accuracy of Thrust Programming Required	-	-	-	-	-
	Restart Requirements	4-6	4-6	4-6	7-9	7-9
	Starting Transients (Ignition/Response) Typical, Sec[1]	.01/0.3	.01/0.3	.01/0.3	0.04 /.3	0.04 /.3
	Cutoff-Impulse Tolerance, 3σ, lbf-sec	150	150	70	420	200
	Startup-Impulse Tolerance[2]	Not critical	Not critical	Not critical	Not critical	Not critical
	Vector Control Requirements	$\pm 6°$	$\pm 6°$	$\pm 6°$	$\pm 6°$	$\pm 6°$
	Vector Control Implementation	Gimbal	Gimbal	Gimbal	Gimbal	Gimbal
PROPELLANTS:	Composition, Mixture Ratios	N_2O_4/Aerozine-50 (2.1:1)	N_2O_4/Aerozine-50 (2.1:1)	N_2O_4/Aerozine-50 (2.1:1)	LO_2/LH_2 (5:1)	LO_2/LH_2 (5:1)
	Nominal Delivered Specific Impulse, lbf-sec/lbm	313	313	313	430	430
	I_{sp} Uncertainty (Computed/Delivered) lbf-sec/lbm	<3/6	<3/6	<3/6	<3/6	<3/6
	Operational Compatibility	Good (Toxic Propellants)	Good (Toxic Propellants)	Good (Toxic Propellant)	Excellent	Excellent
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:	Zero 'g' Propellant Supply	Bladders, or attitude-control system	Bladders, or attitude-control system	Bladders, or attitude-control system	Settling jet or attitude-control system	Settling jet or attitude-control system
	Space Boil-off Losses	None	None	None	<5%	5 to 10%
	Insulation	None required	None Required	None required	.5 in. Linde SI-4	.5 in. Linde SI-4
	Micrometeorite[3] Protection	No special protect. req'd.	No special protect. req'd.	No spec. protect. req'd	No special protect. req'd.	No special protect. req'd.
	Ionizing Radiation Protection	None required	None required	None required	None required	None required
	Target or Payload Contamination	No problems anticipated	No problems anticipated	No problems anticipated	No problems anticipated	No problems anticipated
	Ground Support Considerations	Hazardous propel. handling	Hazardous propel. handling	Hazard. propel. hand.	Cryogenic propel. handling	Cryogenic propel. handling
SYSTEM GEOMETRY AND CHARACTERISTICS	Configuration	$5-B_1$	$8-A_1$	$9-A_1$	$3-C_1$	$6-C_1$
	Vehicle Envelope Restrictions, Maximum	260 in. dia.	220 in. dia. max. 154 in. dia. min.	220 in. dia. max. 154 in. dia. min.	260 in. dia.	220 in. dia.
	Propellant Feed System	Pressure	Pressure	Pressure	Pressure	Pressure
	Thrust-Chamber Cooling Method (Main)	Ablative	Ablative	Ablative	Ablative	Ablative
	Thrust-Chamber Cooling Method (Abort)	-	-	-	Ablative	Ablative
	Thrust-Chamber Characteristics (Main)	$\epsilon = 40$, $P_c = 100$	$\epsilon = 40$, $P_c = 100$	$\epsilon = 40$ $P_c = 100$	$\epsilon = 40$, $P_c = 100$	$\epsilon = 40$, $P_c = 100$
	Thrust-Chamber Characteristics (Abort)	-	-	-	$\epsilon = 5$ $P_c = 100$	$\epsilon = 5$ $P_c = 100$
	System Weight-Payload/Gross Weight[4]	128,650/150,000	34,850/40,000	13,340/15,000	20,000/150,000 (62,140 excess)	18,000/40,000
	Propellant Fraction	0.55	0.610	0.710	0.722	0.759
RELIABILITY FACTOR[5]		0.988[illegible]	0.9902	0.9907	0.9586 (.9977 for abort)	0.9517 (.9970 for abort)

Notes:

1. Assume tanks pressurized prior to start.
2. Guidance system monitors delivered velocity increment in real time for all maneuvers.
3. Micrometeorite environment not well defined. Vehicle skin and tank wall should provide high probability of no disabling damage.
4. The excess payload weight for the $3-C_1$ configuration is the payload that would have to be jettisoned in the lunar orbit to deliver the 20,000 lbm maximum returned payload.
5. "Inherent" system reliability factor based on current reliability data.

MANEUVER PROPULSION REQUIREMENTS SUMMARY

MANNED ORBITING AND RETURN MISSIONS

Vehicle/Maneuver	Ideal ΔV Capability ft/sec	Nominal[1] Initial Mass, m_o lbm	Nominal[1] Thrust, F lbf	Nominal[1] Total Impulse, I_t lbf-sec x 10^{-6}	IMPULSE ACCURACY: $\Delta I_t/m_o$ lbf-sec/lbm	IMPULSE ACCURACY: ΔI_t lbf-sec	IMPULSE ACCURACY: $\Delta I_t/F$, sec	GENERAL REQUIREMENTS: Thrust Variability (Overall Maneuver)	GENERAL REQUIREMENTS: No. of Starts	GENERAL REQUIREMENTS: Space Storage (days)
NOVA MISSION										
Abort at Injection	6400	80,000[2]	120K	11.4	-	Not critical	-	-	1	-
Outbound Trajectory Corrections	125	150,000	5K	0.60	0.01	1500	0.30	-	2-3	3-4
Outbound Orbit Injection	3300	148,500	25K	13.5	0.10	11,000	0.45	-	1	3-4
Perilune Variation	250	115,000	25K	0.90	0.10	11,000	0.45	-	1	3-6
Return Orbital Launch	3200	35,000[3]	10K	3.0	0.10	2800	0.28	-	1	5-8
Return Trajectory Corrections	200	28,000	1K	0.17	0.01	250	0.25	-	2-3	7-10
Totals (Excluding Abort)	7075			18.2						
SATURN C-3 MISSION										
Abort at Injection	5900	40,000	70K	5.4	-	Not critical	-	-	1	-
Outbound Trajectory Correction	125	40,000	2K	0.16	0.01	400	0.20	-	2-3	3-4
Outbound Orbit Injection	3300	39,500	10K	3.6	0.10	3000	0.30	-	1	3-4
Perilune Variation	250	30,000	10K	0.23	0.10	3000	0.30	-	1	3-6
Return Orbital Launch	3200	29,500[4]	10K	2.6	0.10	2500	0.25	-	1	5-8
Return Trajectory Corrections	200	23,500	1K	0.15	0.01	200	0.20	-	2-3	7-10
Totals (Excluding Abort)	7075			6.7						

1 As for Table 5.
2 Assume excess payload jettisoned for abort.
3 Based on 20,000 lbm return payload weight.
4 No payload jettisoned at moon.

MANEUVER PROPULSION REQUIREMENTS SUMMARY

MANNED LANDING AND RETURN MISSION

Vehicle/Maneuver	Ideal ΔV Capability ft/sec	Nominal[1] Initial Mass, m_o lbm	Nominal[1] Thrust, F lbf	Nominal[1] Total Impulse, I_t lbf-sec x 10^{-6}	IMPULSE ACCURACY $\Delta I_t/m_f$ lbf-sec/lbm	IMPULSE ACCURACY ΔI_t lbf-sec	IMPULSE ACCURACY $\Delta I_t/F$, sec	GENERAL REQUIREMENTS Thrust Variability (Overall Maneuver)	GENERAL REQUIREMENTS No. of Starts	Space Storage In-flight	Space Storage Lunar Surface
NOVA MISSION											
Abort[2]	-	-	-	-	-		-	-	-	-	-
Outbound Trajectory Correction	125	150,000	5K	0.60	0.01	1500	0.30	-	2-3	3-4 day	-
Outbound Orbit Injection	3300	148,500	25K	13.5	0.10	11,000	0.45	-	1	3-4 day	-
Perilune Variation	250	115,000	25K	0.90	0.03	3300	0.13	(?)[4]	1	3-4 day	-
Landing from Orbit	6000	113,000	60K max.	17.2	-		-	1.2:1	1	3-5 day	-
						(Closed-loop control)					
Hovering and Transverse Maneuvering (3-5 min)	1600	73,000 (m_f=65,000)	15K max.	3.4	-		-	1.5:1	1	3-5 day	-
Lunar Take-off	9000	55,000[3]	50K	11.3	0.10	3000	0.06	-	1	3-5 day	3-10 day
Return Trajectory Corrections	200	29,000	1K	.18	0.01	300	0.30	-	2-3	6-8 day	3-10 day
Totals	20,475			47.1							

1 As for Table 5.
2 Adequate on-board ΔV capability allows successful abort during injection or translunar flight using mission propulsion.
3 Some inerts jettisoned at moon.
4 Thrust reduction may be desirable for I_t accuracy.

SELECTED PROPULSION-SYSTEM SPECIFICATIONS

NOVA LUNAR LANDING AND RETURN

		SINGLE STAGE	TWO STAGE 1st Stage	TWO STAGE Cryogenic 2nd Stage	TWO STAGE Storable 2nd Stage
TOTAL-IMPULSE REQUIREMENT, lbf-sec		47.67×10^6	35.75×10^6	11.65×10^6	10.31×10^6
THRUST:	Nominal Thrust Level, lbf	60K Main	50K Main 4-3.5K Vernier	2-25K Main 2-2K Verniers	2-25K Main 2-2K Verniers
	Type of Thrust Control	Continuous Modulation	Continuously modulated Verniers	Not Required	Not Required
	Control Range, lbf	10 to 60K	5 to 14K (Verniers)	-	-
	Accuracy of Thrust-Programming Required	Not critical	Not critical	-	-
	Restart Requirements	8 - 10	3 Main, 6-7 Verniers	1 Main, 3-4 Verniers	1 Main, 3-4 Verniers
	Starting Transients(Ignition/Response)Typical, sec[1]	0.04 /1	0.04 / 1(Main) .3(Verniers)	0.04 / 1(Main) 0.3(Verniers)	.014/ 1(Main) 0.3(Verniers)
	Cutoff-Impulse Tolerance, 3σ, lbf-sec	420	200	150	150
	Startup Impulse Tolerance	Not critical	Not critical	Not critical	Not critical
	Vector-Control Requirements	$\pm 6°$	$\pm 6°$	$\pm 6°$	$\pm 6°$
	Vector-Control Implementation	Gimbal	Gimballed Verniers	Gimballed Verniers	Gimballed Verniers
PROPELLANTS:	Composition, Mixture Ratio	LO_2/LH_2 (4.8:1)	LO_2/LH_2 (4.8:1)	LO_2/LH_2 (5:1)	N_2O_4/Aerozine-50 (2.1:1)
	Nominal Delivered Specific Impulse, lbf-sec/lbm	424	424	430	313
	I_{sp} Uncertainty(Computed/Delivered)lbf-sec/lbm	<3/6	<3/6	<3/6	<3/6
	Operational Compatibility	Excellent	Excellent	Excellent	Good (Toxic Propellants)
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:	Zero 'g' Propellant Supply	Settling jet or attitude control system	Settling jet or attitude control system	Settling jet or attitude-control system	Settling jet or attitude control system
	Space Boiloff Losses	< 3%	< 3%	5 to 10%	None
	Insulation	.5 in. Linde SI-4	.5 in. Linde SI-4	.5 in. Linde SI-4	None required
	Micrometeorite Protection[3]	No special protect. req'd.	No special protect. req'd.	No special protect. req'd.	No special protect. req'd.
	Ionizing Radiation Protection	None required	None required	None required	None required
	Target or Payload Contamination	No problems anticipated	No problems anticipated	No problems anticipated	No problems anticipated
	Ground-Support Considerations	Cryogenic propel. handling	Cryogenic propel. handling	Cryogenic propel. handling	Hazardous propel. handling
SYSTEM GEOMETRY AND CHARACTERISTICS	Configuration	$1\text{-}A_{12}$	$2\text{-}A_{11}$	$2\text{-}A^{10}$	$2\text{-}A^{12}$
	Vehicle-Envelope Restrictions	260 in. dia.	260 in. dia.	260 in. dia.	260 in. dia.
	Propellant-Feed System	Turbo-Pump	Turbo-Pump	Pressure	Pressure
	Thrust-Chamber Cooling Method (Main)	Regenerative	Regenerative	Ablative	Ablative
	(Vernier)	-	Ablative	Ablative	Ablative
	Thrust-Chamber Characteristics (Main)	ϵ = 40:1 P_c = 300	ϵ = 40 P_c = 300	ϵ = 40 P_c = 100	ϵ = 40 P_c = 100
	(Vernier)	-	ϵ = 40 P_c = 40	ϵ = 40 P_c = 100	ϵ = 40 P_c = 100
	System Weight - Payload/Gross Weight	19,148/150,000	51,770/150,000	17,790/51,770	13,470/51,770
	Stage Propellant Fraction[+]	0.860	0.855	0.797	0.860
RELIABILITY FACTOR[4]		0.9444	0.9156	0.9714	0.9798

Notes:
1. Assume tanks pressurized prior to start.
2. Guidance system monitors delivered velocity increment in real time for all maneuvers.
3. Micrometeorite environment not well defined. Vehicle skin and tank wall should provide high probability of no disabling damage.
4. "Inherent" system reliability factor based on current reliability data.

TABLE 9

SUMMARY OF PROPULSION REQUIREMENTS FOR THE 24-HOUR SATELLITE (2-YEAR SATELLITE LIFE)

REQUIREMENTS

Function	Velocity (ft/sec)	Total Impulse (lbf-sec)	Thrust (lb)	Duration	Other
CENTAUR (800-lb payload)					
Orbit correction	100 to 450	1600 to 12,000	0.32 (Min.); 10 + Desired	1 mo	(1) Restartability required (2) Attitude control required
Station keeping	50 to 130	480 to 2600	0.2 to 8	2 mo to 2 yr	(1) Restartability required (2) Attitude control required
Attitude control	-	1140 nominal	0.01 to 0.1	2 yr	(1) Restartability required
SATURN C-2 (6000-lb payload)					
Orbit correction	100 to 450	12,000 to 90,000	2.5 (Min.); 75 + Desired	1 mo	(1) Restartability required (2) Attitude control required
Station Keeping	50 to 130	3600 to 19,500	1 to 50	2 mo to 2 yr	(1) Restartability required (2) Attitude control required
Attitude control	-	15,600 nominal	0.1 to 1.0	2 yr	(1) Restartability required
SATURN C-3 (28,000-lb payload					
Orbit correction	100 to 450	56,000 to 420,000	11 (min.); 350 + Desired	1 mo	(1) Restartability required (2) Attitude control required
Station keeping	50 to 130	16,800 to 91,000	5 to 300	2 mo to 2 yr	(1) Restartability required (2) Attitude control required
Attitude control	-	84,400 nominal	1 to 10	2 yr	(1) Restartability required

TABLE 10

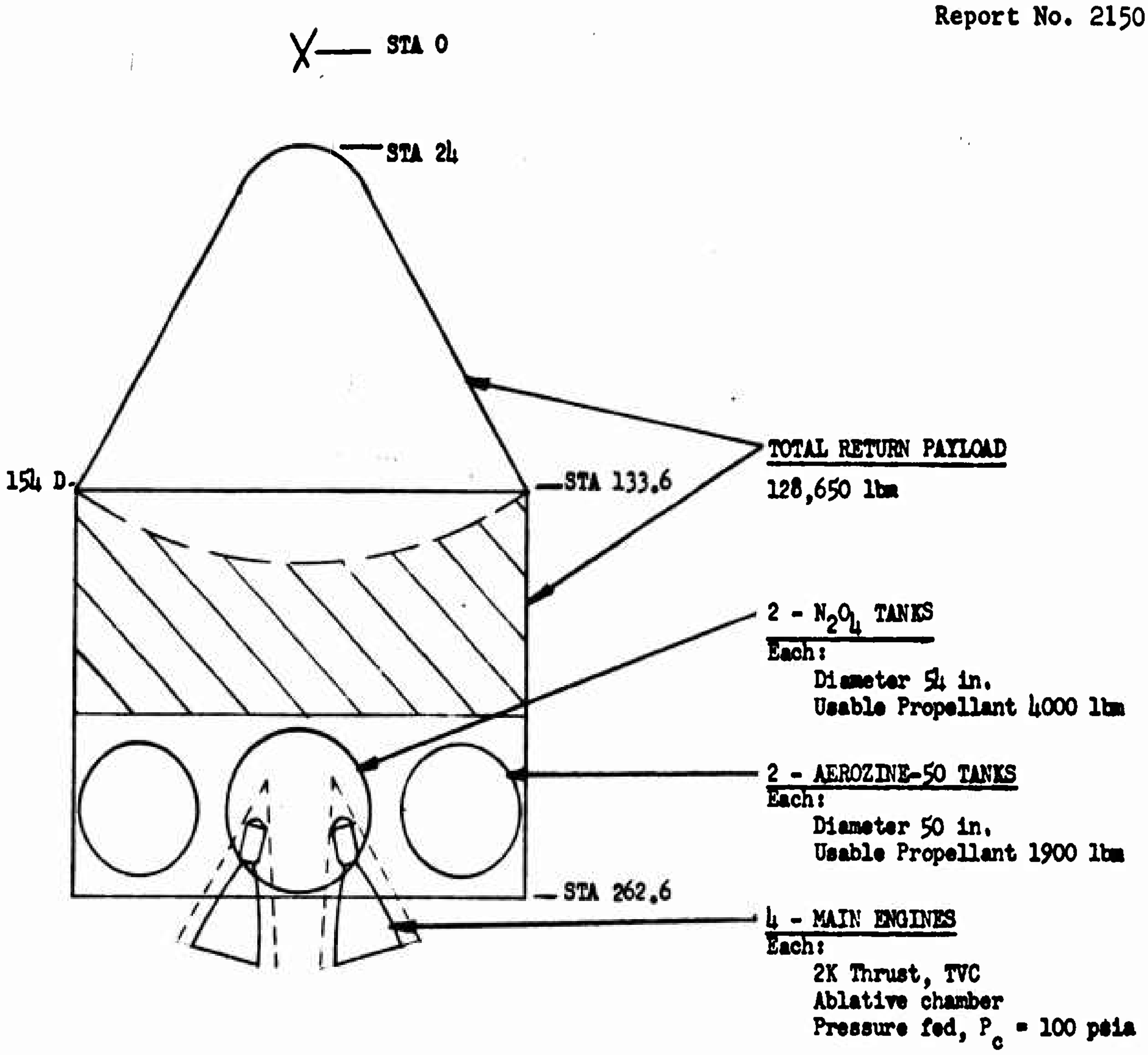
SELECTED SYSTEMS-WEIGHT BREAKDOWN
VEHICLE AND PAYLOAD

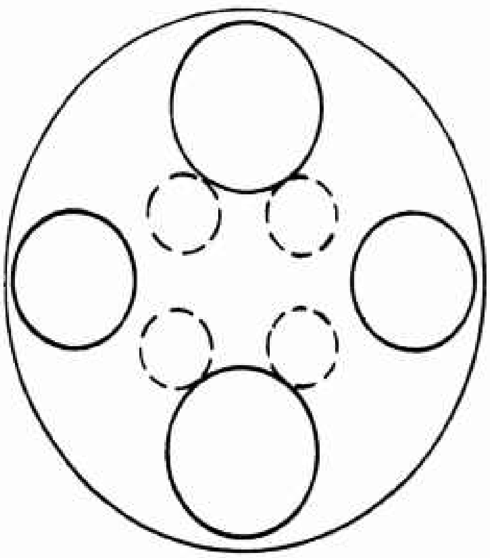
Component (wt in lb)	Centaur (800 lb)	Saturn C-2 (6000 lb)	Saturn C-3 (28,000 lb)
	Liquid noncryogenic bi-propellant for orbit correction and station keeping; reaction wheels with cold-gas augmentation for attitude control.	Liquid noncryogenic bi-propellant for orbit correction and station keeping and attitude-control augmentation; reaction wheels for primary attitude control.	Same as Saturn C-2
Propellants	N_2O_4/Aerozine-50	N_2O_4/Aerozine-50	N_2O_4/Aerozine-50
Propellant	52	390	1813
Propellant tank	3.5	15	60
Pressurization gas	3.0	4	18
Pressurizing tank assembly	5.5	7	32
Regulators and valves	10	17	24
Rocket assembly	5	9	18
Reaction-wheel assembly	18	35	75
Fittings, etc.	6	8	12
Total	103-	485	2052

SELECTED PROPULSION-SYSTEM SPECIFICATIONS

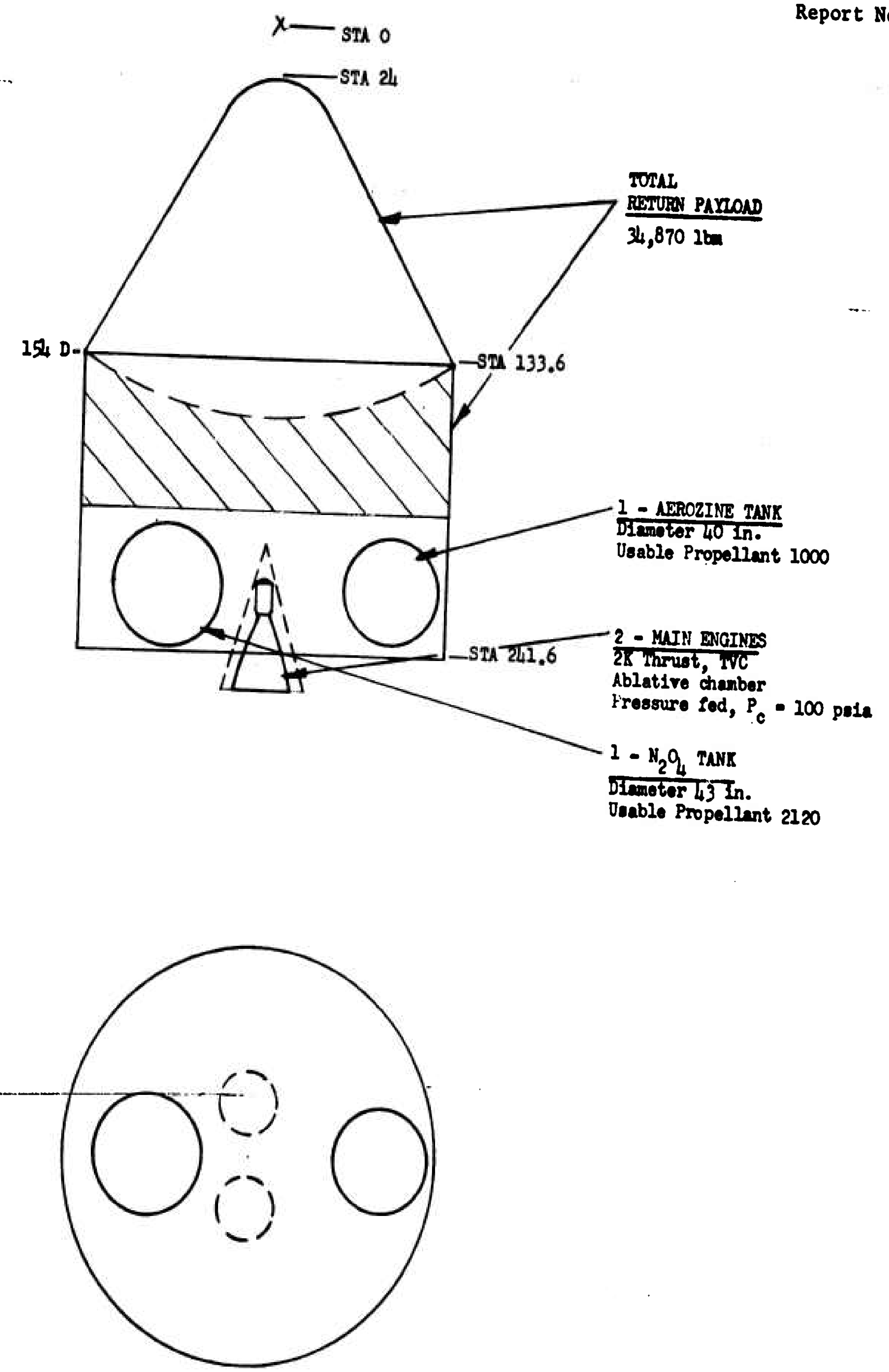
		CENTAUR 800-lb PAYLOAD	SATURN C-2 6000-lb PAYLOAD	SATURN C-3 28,000-lb PAYLOAD
TOTAL IMPULSE REQUIREMENT, lbf-sec		15,000	111,000	518,000
THRUST:	Nominal Thrust Level (Station-keeping/Attitude control), lbf	5/0.1	40/1	150/10
	Type of Thrust Control (Station-keeping)	Pulsing operation	Pulsing operation	Pulsing operation
	Control Range, lbf	-	-	-
	Accuracy of Thrust-Programming Required	-	-	-
	Restart Requirement (Orbit Correction/Station-keeping/Attitude control)	10/50/50	10/50/50	10/50/50
	Starting Transients (Ignition/Response) Typical, sec[1]	0.005/0.025	0.005/0.025	0.005/0.025
	Cutoff-Impulse Tolerance (Orbit correction/Station-keeping/Attitude control), 3σ, lbf-sec	0.065/0.025/0.0005	0.84/0.20/0.0075	2.5/0.75/0.16
	Startup-Impulse Tolerance (Pulsing operation), lbf-sec	0.025	0.20	0.75
	Vector-Control Requirements	-	-	-
	Vector-Control Implementation	Attitude-Control System	Attitude-Control System	Attitude-Control System
PROPELLANTS:	Composition, Mixture Ratio	N_2O_4/Aerozine-50 (2.1:1)	N_2O_4/Aerozine-50 (2.1:1)	N_2O_4/Aerozine-50 (2.1:1)
	Nominal Delivered Specific Impulse, lbf-sec/lbm	300	300	300
	I_{sp} Uncertainty (Computed/delivered), lbf-sec/lbm	3/6	3/6	3/6
	Operational Compatibility	Excellent	Excellent	Excellent
ENVIRONMENTAL AND OPERATIONAL RESTRICTIONS:	Zero 'g' Propellant Supply	Redundant Teflon Bladders	Redundant Teflon Bladders	Redundant Teflon Bladders
	Space Boiloff Losses	None	None	None
	Insulation	None Required	None Required	None Required
	Micrometeorite Protection[2]	No Special Protect. req'd.	No Special Protect. req'd.	No Special Protect. req'd.
	Ionizing-Radiation Protection[3]	None Required	None Required	None Required
	Target or Payload Contamination	No Problems Anticipated	No Problems Anticipated	No Problems Anticipated
	Ground-Support Considerations	Hazardous Propel. Handling	Hazardous Propel. Handling	Hazardous Propel. Handling
SYSTEM GEOMETRY AND CHARACTERISTICS	Configuration	Figure 48	Figure 48	Figure 48
	Vehicle-Envelope Restrictions	120-in. diameter	220-in. diameter	220-in. diameter
	Propellant-Feed System	Pressurized	Pressurized	Pressurized
	Thrust-Chamber Cooling Method	Radiation	Radiation	Radiation
	Thrust-Chamber Characteristics	ϵ = 40:1, P_c = 25	ϵ = 40:1, P_c = 25	ϵ = 40:1, P_c = 25
	System Weight, lbf	103		2050
RELIABILITY CONSIDERATIONS:	Inherent System-Reliability Factor[4]	0.79	0.75	0.75

1 Assumes tanks pressurized prior to start.
2 Micrometeorite environment not well defined. Vehicle skin and tank wall should provide high probability of no disabling damage.
3 Protection for the propellant bladders should be provided by the tanks and vehicle skin.
4 Based on critical-component analysis with current failure-rate data.

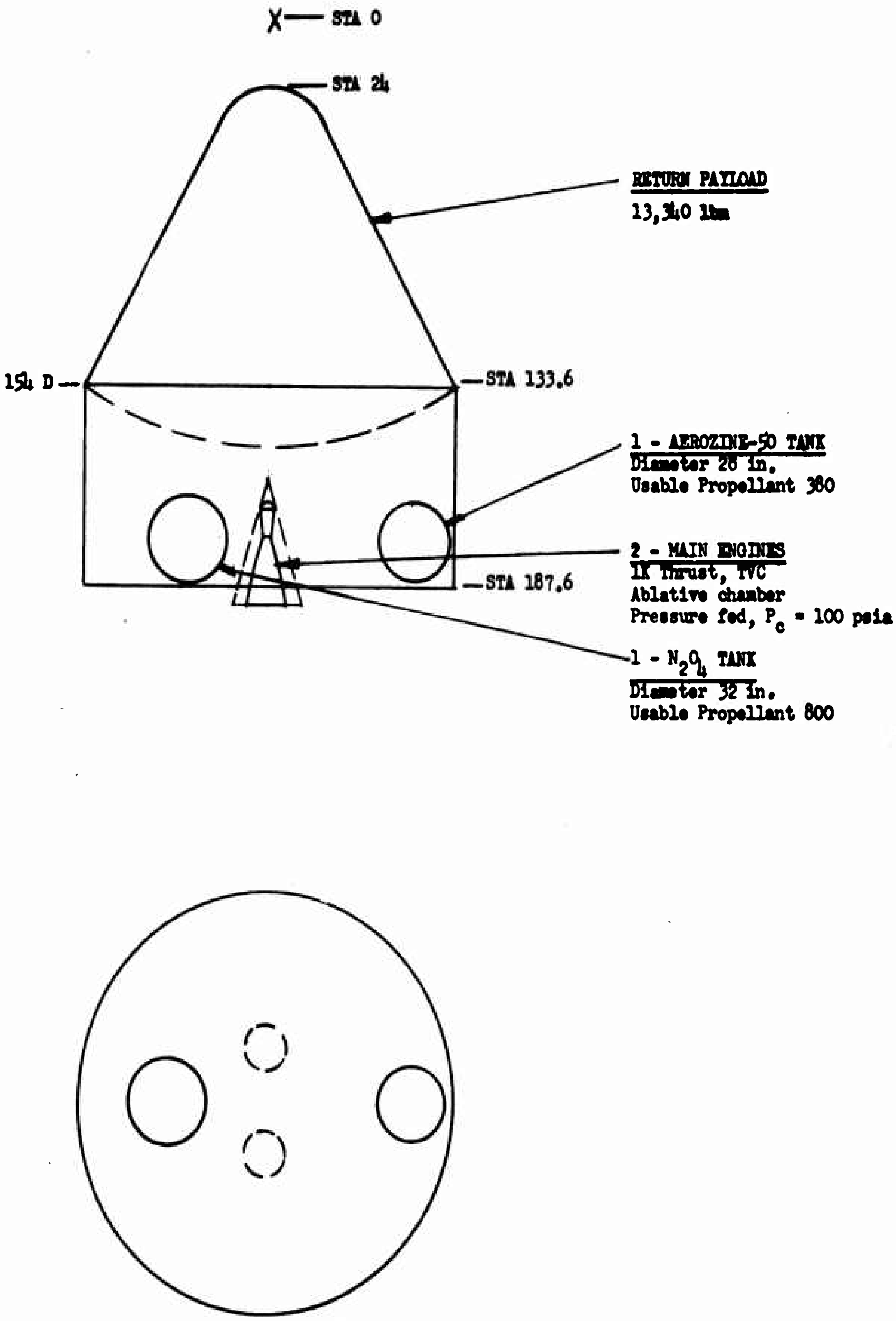




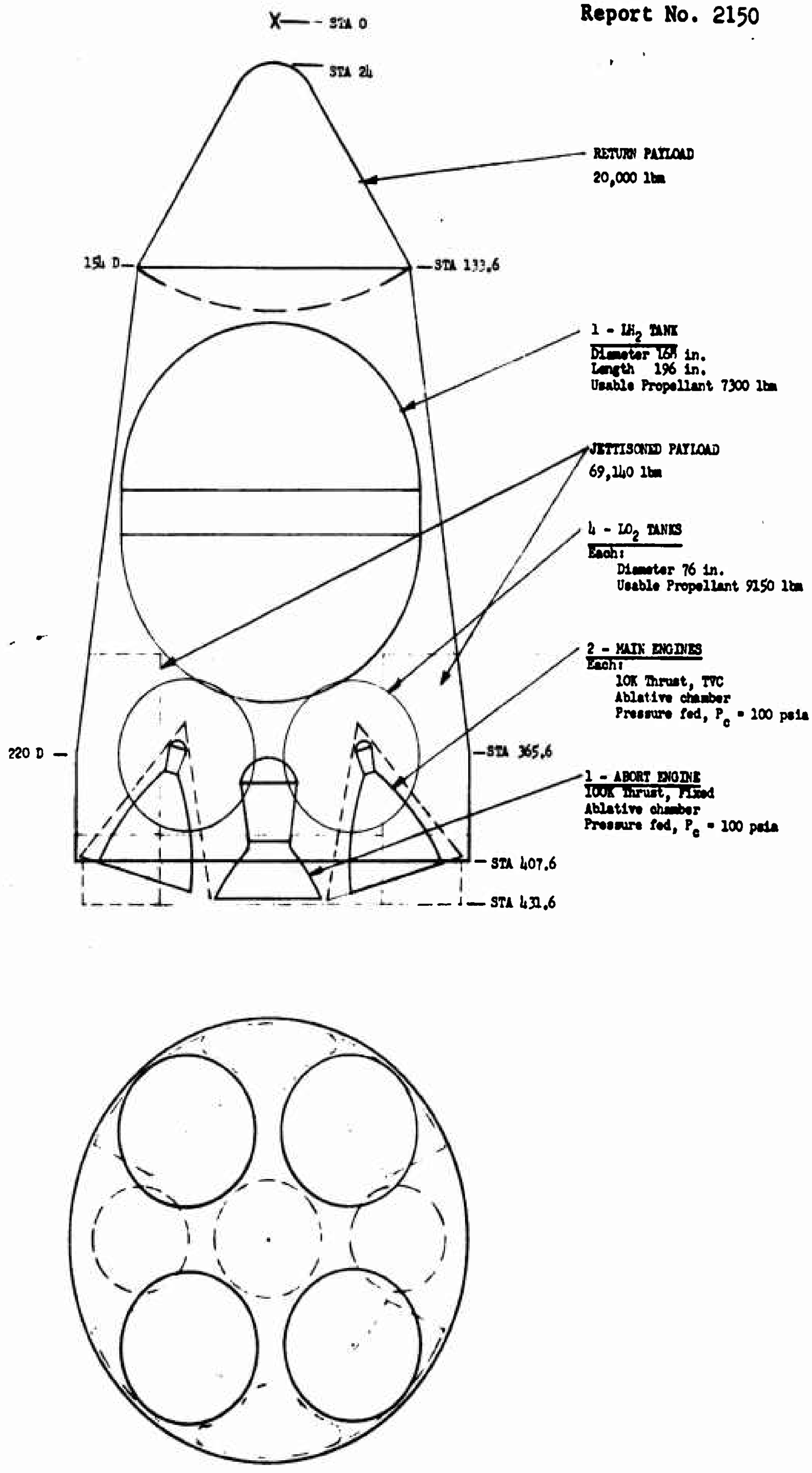
Selected Alternate $5\text{-}B_1$,
Nova Circumlunar Mission



Selected Alternate 8-A_1,
Saturn C-3 Circumlunar Mission

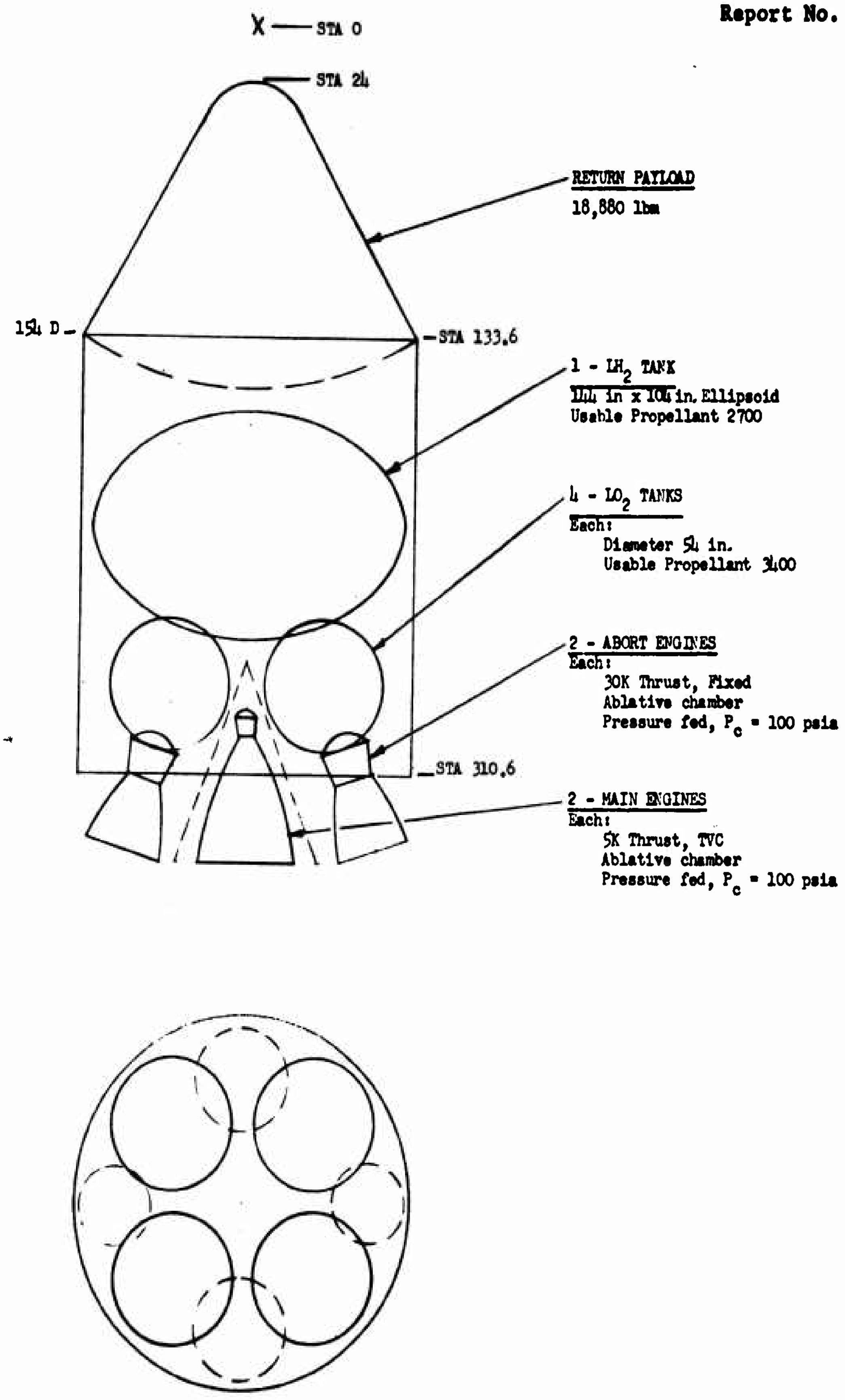


Selected Alternate 9-A_1,
Saturn C-2 Circumlunar Mission

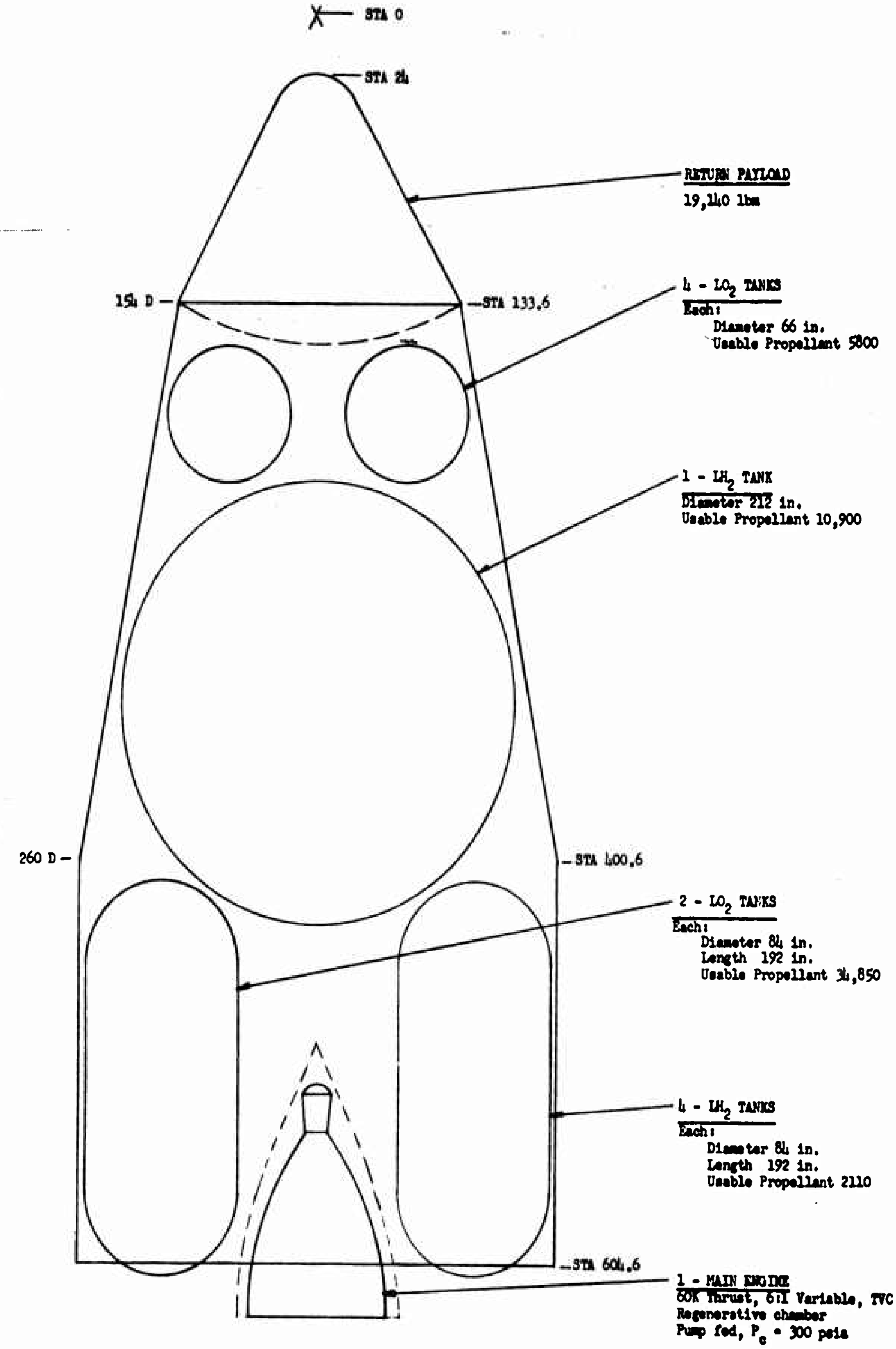


Selected Alternate 3-C_1,
Nova Lunar Orbiting and Return Mission

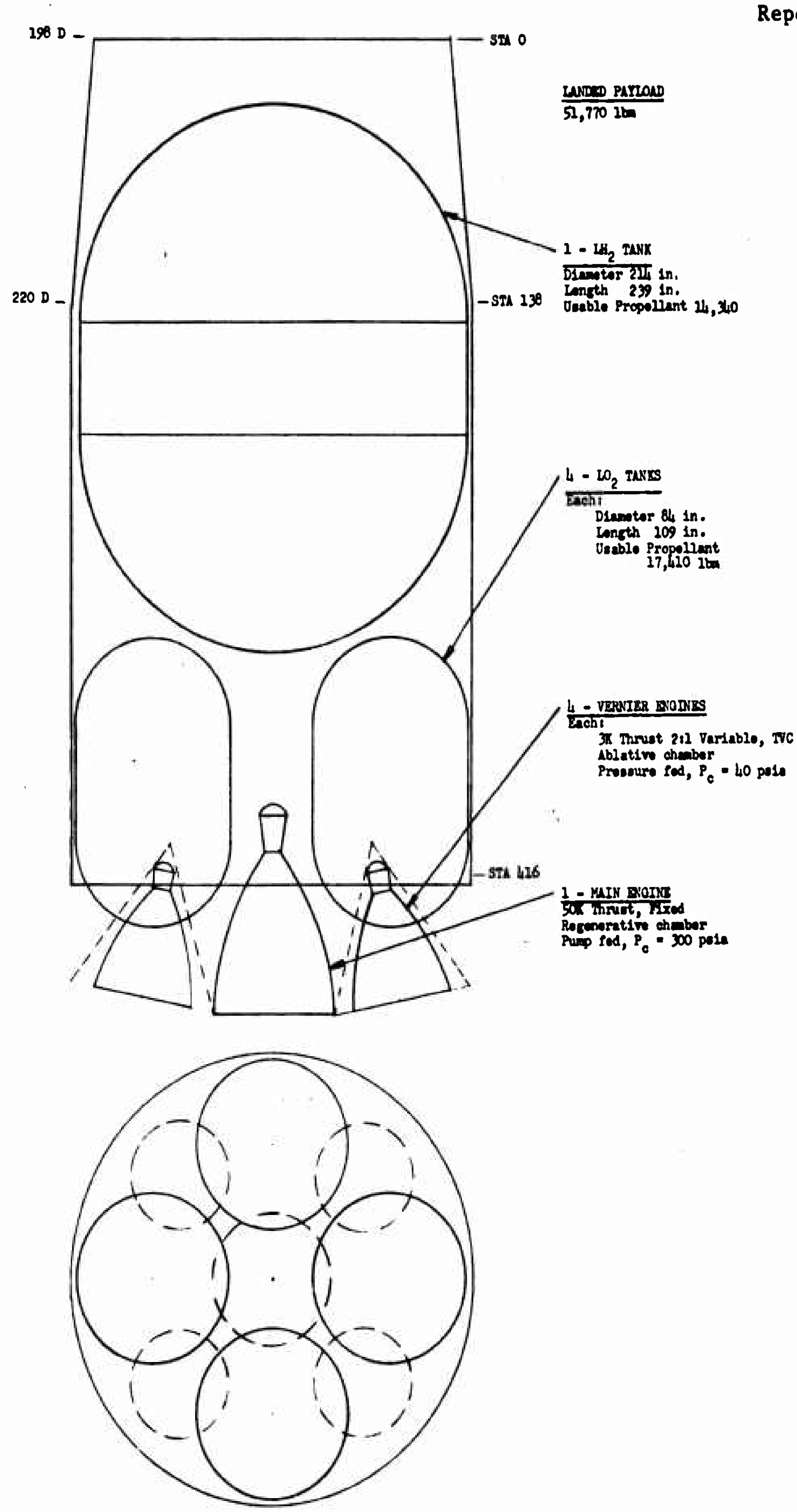
Figure 4



Selected Alternate 6-C_1,
Saturn C-3 Lunar Orbiting and Return Mission

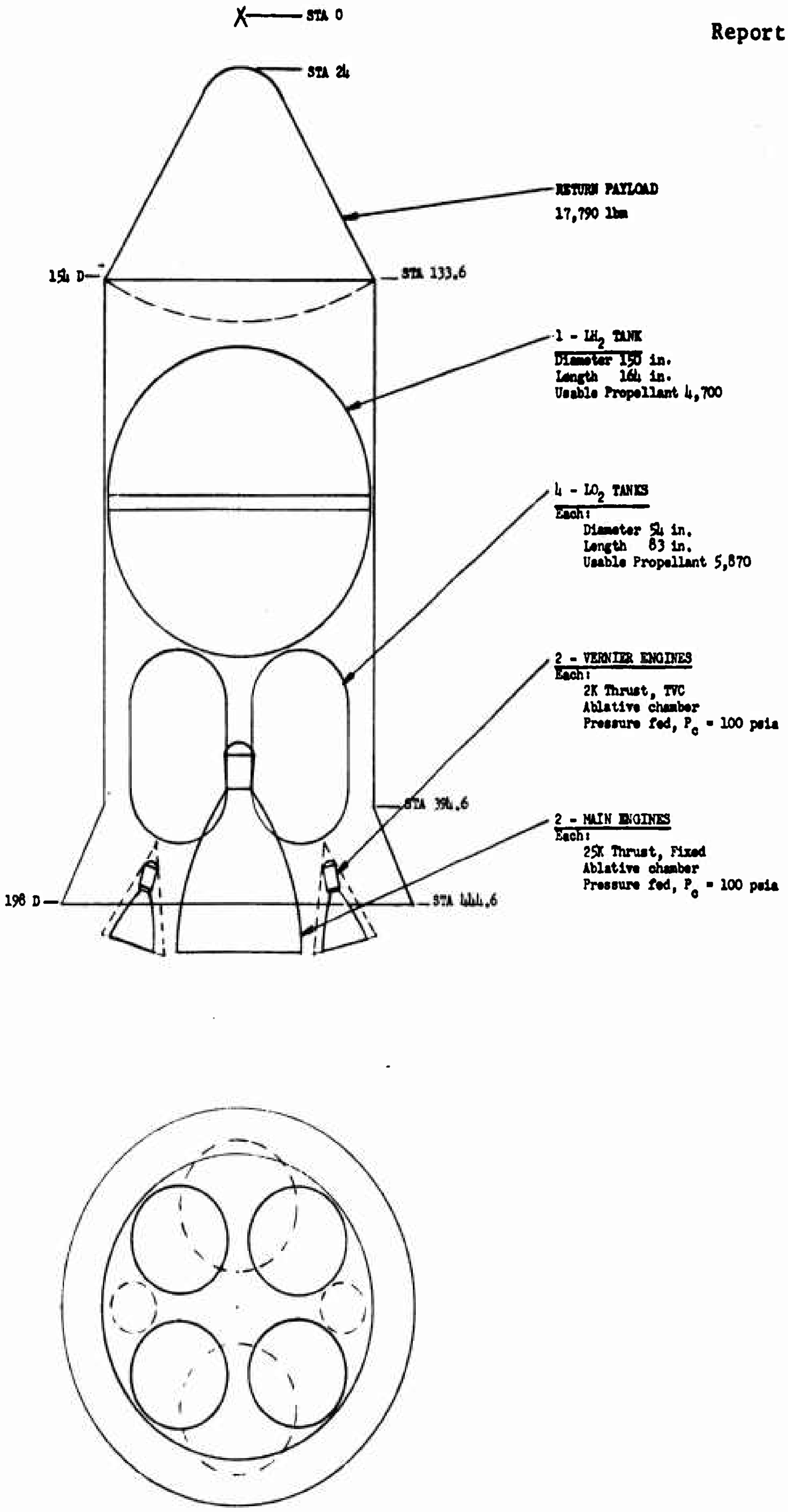


Selected Alternate 1-A_{12},
Nova Lunar Landing and Return Mission

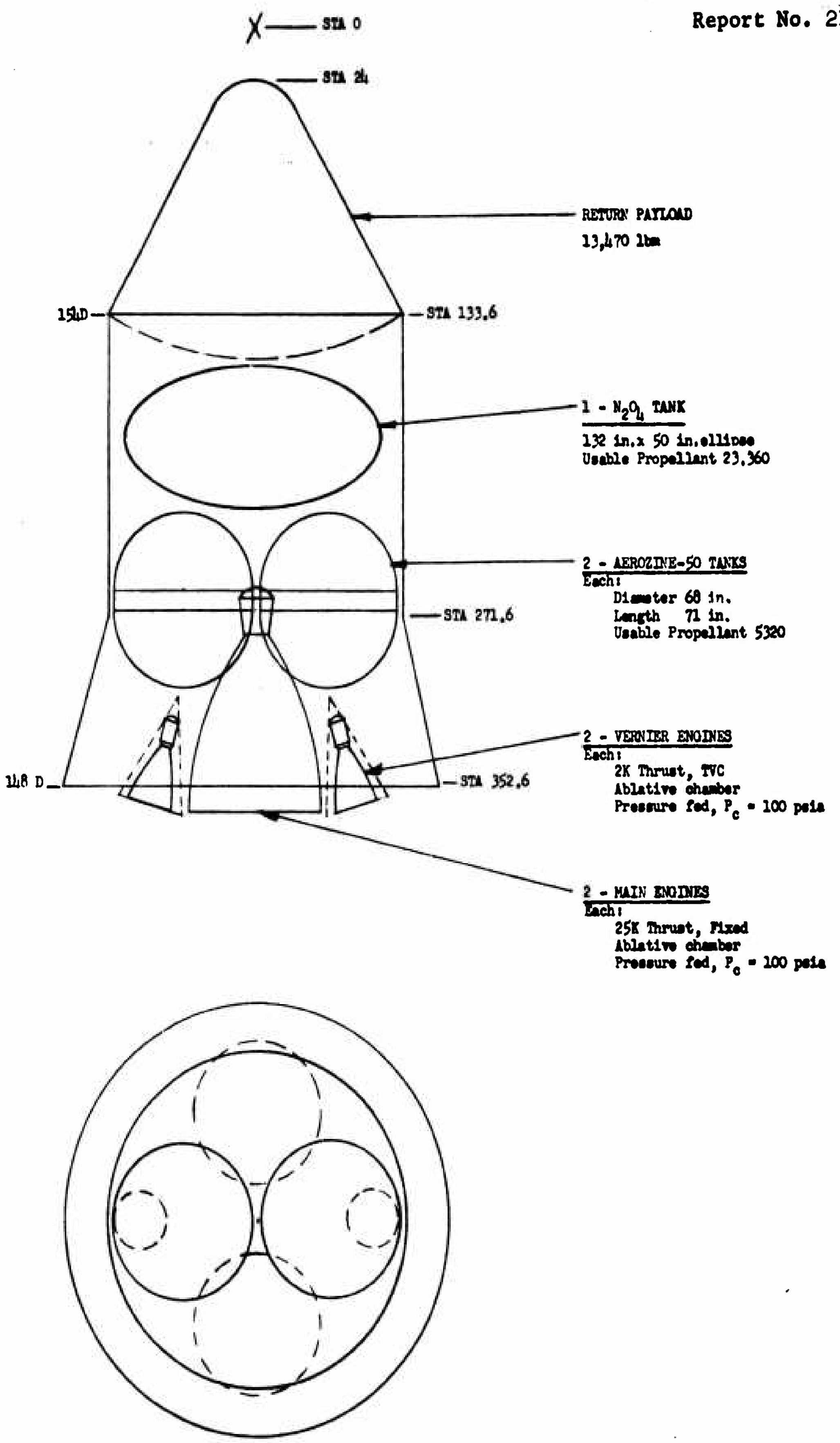


Selected Alternate 2-A_{11},

Nova Lunar Landing and Return Mission, First Stage

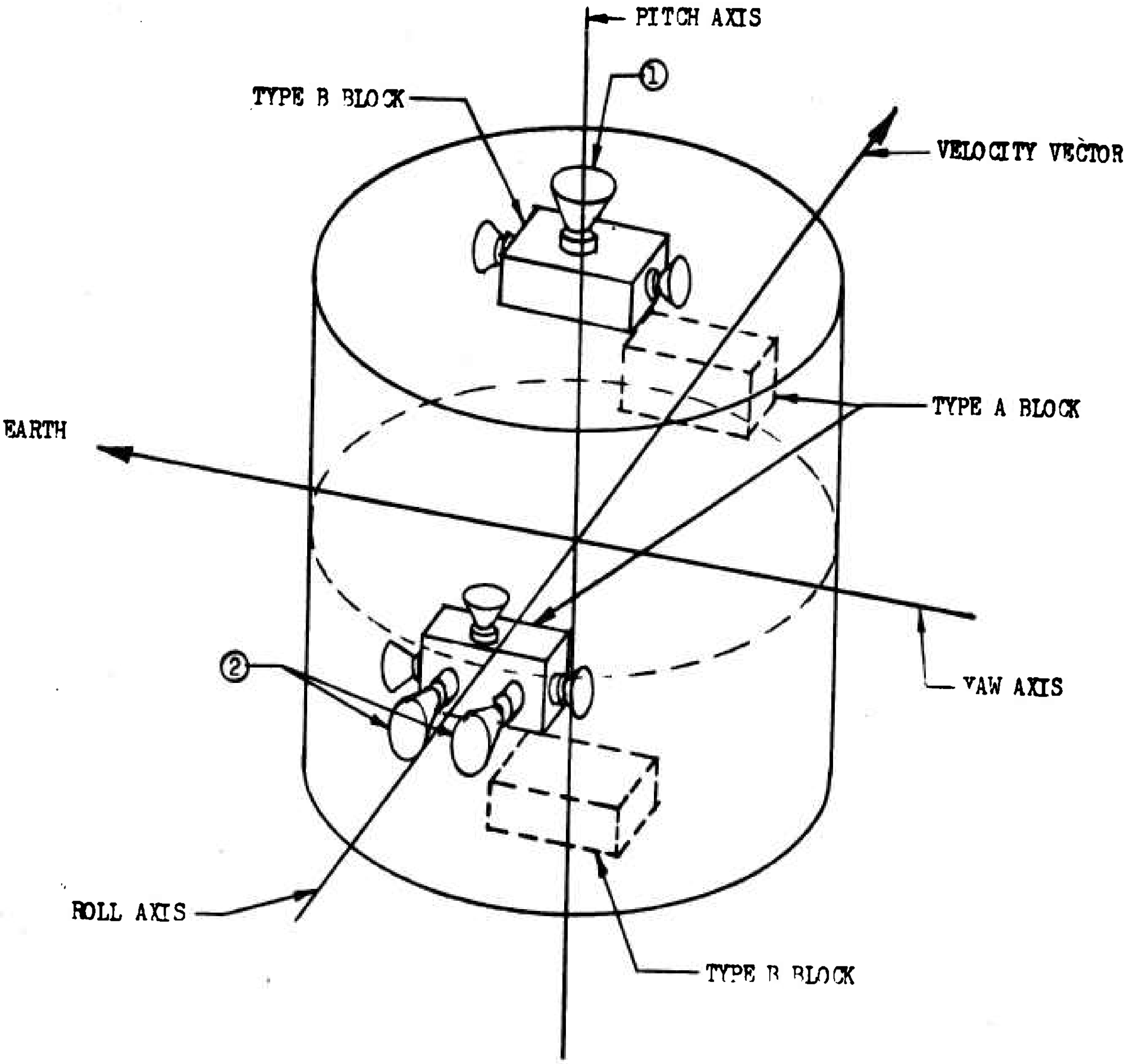


Selected Alternate 2-A[10],
Nova Lunar Landing and Return Mission, Second Stage



Selected Alternate 2-A[12],
Nova Lunar Landing and Return Mission, Second Stage

Figure 9



1. Out-of-Plane Orbit Correction Rockets
2. In-Plane Orbit Correction Rockets

All Other Nozzles are for Attitude Control

Rocket Nozzle Orientation

Figure 10

UNCLASSIFIED

UNCLASSIFIED