NASA-S-66-6585 JUN

APOLLO SPACECRAFT WEIGHT DATA

	CURRENT (LBS)	PREDICTED (LBS)	MAXIMUM (LBS)
COMMAND MODULE (INCL CREW)	11,755	12,050	12,250
SERVICE MODULE	10,300	11,250	11,300
SM BAY PAYLOAD	0	0	1000
SPS PROPELLANT	37,075	38,800	38,964
LUNAR MODULE (NOT INCL CREW)	30,755	32,000	32,486
SC/LM ADAPTER	3755	3900	4000
TOTAL	93,640	98,000	100,000

Fig. 65

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LUNAR MODULE WEIGHT BREAKDOWN

	CURRENT (LBS)	PREDICTED (LBS)	MAXIMUM (LBS)
ASCENT STAGE	4450	4620	4645
RCS PROPELLANT	507	540	540
APS PROPELLANT	4538	4810	4921
DESCENT STAGE	4685	4795	4795
DPS PROPELLANT	16,575	17,235	17,585
TOTAL	30,755	32,000	32,486

Fig. 66

APOLLO NAVIGATION, GUIDANCE AND CONTROL

by

Robert C. Duncan

INTRODUCTION

The purpose of the guidance system is to control the position and velocity of the vehicle. The navigation process involves the determination and indication of position and velocity, and the guidance process involves controlling these quantities in a closed-loop fashion. Fig. 1 shows a generalized functional diagram of the guidance and control system. In order to minimize guidance errors the system must reduce the effect of interferring quantities, and it must respond quickly to command signals. An inertial guidance system is fundamentally mechanized as a specific force measuring system using single axis accelerometers which operate in coordinates that are determined by gyros.

The guidance system operates as a force-vector control system, i.e., the system must change the direction and magnitude of controllable forces (lift, drag, and thrust) in such a way that the vehicle reaches its desired point in space and time. It is usual in the theory of dynamics of rigid bodies in three dimensions to separate the motion of the center of mass from the motion of the body around the center of mass. Guidance is the process of moving the center of mass of the vehicle along some desired path. Stability and control are associated with motions about the center of mass.

The guidance and control systems for all manned spacecraft have involved a mix of spacecraft systems and ground systems. Fig. 2 shows the guidelines used in the Apollo program for this mix of spacecraft and ground systems:

(1) It is mandatory that there be a ground navigation capability provided in earth orbit, cislunar space, lunar orbit, during the lunar landing phases, and during the lunar rendezvous phases.

(2) It is mandatory that the spacecraft contain onboard a completely self-contained navigation, guidance, and control capability to be used in the event that the data link with the ground is lost.

(3) The onboard system is designed in such a way to take maximum advantage of the ground system and to include all necessary interfaces.

Fig. 3 shows the navigation, guidance, and control system which evolved for the command module. The LEM system is very similar and will be discussed later. The primary navigation system in cislunar space is the ground system. This consists of the manned space flight network (MSFN) comprised of a number of tracking

stations around the world operating in conjunction with the Houston Mission Control Center (MCC). This system is connected to the onboard system by way of the updata link and voice communications. The updata link provides the navigation state vector to the Apollo guidance computer (AGC). The primary guidance and control system consists of the AGC, the inertial measurement unit (IMU), the scanning telescope (SCT), sextant (SXT), and the display and keyboard assembly (DSKY).

The primary guidance and control system operates the reaction control system (RCS) which is used primarily for attitude control in space and during reentry. The AGC also activates the gimbal servos to drive the service propulsion (SPS) engines. In the event the primary control system has a failure, the backup system (labeled in Fig. 3 the Stabilization System) can also drive the reaction control system and the SPS gimbals. The SCS (stabilization and control system) provides an attitude reference and also has an accelerometer to measure ΔV. The entry monitor system (EMS) is a simplified backup guidance system to be used during the entry phase of the mission in the event of failure of the primary system. An integral part of both the primary system and the backup stabilization system is the astronaut. He obtains information from the computer by the DSKY and from the display panel. He communicates with the computer through the DSKY and is able to control the system through the use of the engine throttle and attitude hand controller.

The stabilization system is shown in block diagram form in Fig. 4. The basic function of this system is:

(1) Drive the jet drivers to turn on and off the small reaction thrusters.

(2) Direct the gimbals of the service module engine to orientate properly the thrust vector of the main engine.

Attitude information comes either from the G&N system (guidance and navigation system) or the AGAP (attitude gyro accelerometer package). Rate information comes from the rate gyro package (RGP) and is displayed on the display panel. Rate and attitude information is used in conjunction with the manual controller to control the attitude jets and the main engine gimbals. The attitude jets can be controlled through two paths, one path via a deadband limiter, pseudo rate logic, and jet select logic to the jet drivers and the other path direct by manual control to the jet drivers. The term pseudo-rate means that the output of the switching amplifier (an on-off device) commands a vehicle acceleration which neglects reaction jet time delays and dynamics. The short period output of this signal through a lag

filter is indicative of the vehicle rate. The time constant of the lag network determines the interval over which the output is a valid indication of the vehicle rate. The gains and time constant have been selected for the Apollo SCS to provide the desired signal for an average vehicle inertia during the lunar mission. The configuration of the pseudo-rate feedback for the Apollo SCS has been developed for limit cycle operation. During maneuvers the effect of the feedback should be to pulse the jets prior to the commanded maneuver rate being achieved, thus resulting in an over-damped response. To avoid this, the pseudo-rate feedback is switched out during manual maneuvers.

The guidance and navigation system is located in the lower equipment bay of the spacecraft, Fig. 5.

The G&N equipment is shown in a handling fixture in Fig. 6. The primary components of this system are the DSKY's, the gimbal position indicators, sextant, scanning telescope, displays and controls, power and servo assembly (PSA), and computer. The inertial measurement unit is behind the panel and is mated with the optical system on the navigation base. A precise angular relation must be maintained between the optical system and the inertial measurement unit; this angular relation is provided through the navigation base.

Fig. 7 is a schematic diagram of the gimbals of the IMU. The stable member houses three single-degree-of-freedom 25 IRIG gyros and three 16 PIPA accelerometers. The gyros maintain a coordinate system with respect to inertial space in accordance with reference directions determined by the optical system and gravity. The accelerometers measure specific forces in the three coordinate directions of this inertial reference system. The acceleration measurements are integrated in the computer to give velocity and integrated again to give position. The platform is isolated from the spacecraft by the three-gimbal system shown in Fig. 7. The three-gimballed platform was chosen instead of a four-gimballed platform because it could be built with smaller size and weight. The only disadvantage of a three gimbal platform is that of gimbal lock in certain orientations. This is readily avoided in Apollo by a simple subroutine in the computer program which torques the platform away from potential gimbal locks as the condition approaches.

Fig. 8 shows the Apollo inertial measurement unit (IMU) with the resolvers on one of the outer gimbals removed. This unit is about the size of a basketball and is very similar to a Polaris platform. The corrugations on the outer portion are coolant lines through which the coolant fluid flows to maintain precision temperature control of the IMU.

Fig. 9 shows the IMU with the top removed. Three gimbals, platform electronics, and the accelerometer and gyro package can be seen in this picture.

Fig. 10 is a photograph of the inertial measurement unit and the optical system (scanning telescope and sextant) mounted on the precision navigation base which maintains accurate angular orientation between the two subsystems. The optical system is used to align the inertial system and for navigation in earth oribt, lunar orbit, and in cislunar space. The inertial measurement unit is used as a primary attitude reference and is used for guidance purposes during all maneuvers and during reentry.

Fig. 11 shows the instrument panel in front of the command pilot of the CSM. The switches in the panel to the right control the CM RCS and SM RCS propellant. The switch and dial at the top right indicate the quantity of RCS propellant. The control panel in the center is the display and keyboard assembly (DSKY). This will be discussed in more detail shortly.

The indicator with curved lines and rays at the top left is the entry monitor system. This system is discussed in greater detail near the end of this paper where the entry phase of the mission is discussed. Directly below the entry monitor system is the FDAI (flight director attitude indicator), commonly called the "8-ball" or the "gyro horizon." The needles above, below, and to the right of the 8-ball itself are error needles. To the left of the FDAI are control switches for the SPS (service module propulsion system). Below the SPS switches are the attitude set indicators and controls.

Directly below the FDAI is the "ΔV Remaining" counter and thrust and direct ullage switches. At the bottom are the control mode select switches for the SCS (Stabilization and Control System). It can be seen that the modes available are:

(1) Monitor

(2) G&N attitude control

(3) G&N ΔV

(4) G&N entry

(5) SCS local vertical mode

(6) SCS attitude control

(7) SCS ΔV

(8) SCS reentry

The throttle control is the T-handle in the lower left-hand corner of the photograph.

Fig. 12 shows the faceplate of the display and keyboard (DSKY). The computer in both the command module and LEM are identical. They are microelectronic computers which are designed by MIT and produced by Raytheon. The Apollo computer is a very powerful lightweight computer with the largest memory of any airborne computer in history. It has a memory of 36,000 words (each of 16 bits) and is approximately equal to an IBM 704 in computational capability.

The DSKY provides the communication link between the astronaut and the computer. Through the DSKY the astronaut can monitor system activity, alter parameters, and dictate system modes. In addition, the DSKY has indicator lights which display system and computer status and alarm. The computer display on the DSKY consists of three two-digit displays labeled "Program", Verb", and "Noun" and three five-digit general word readouts. The two-digit displays are coded for various modes in instruction. The program display indicates the major operating mode of the computer such as "lunar landing maneuver." The "verb" and "noun" displays are used together and coded to give numerous possibilities of meaningful phrases or instructions. Examples of typical "verb" and "noun" displays are:

Verb	Noun
Display value	Velocity
Compute	Abort velocity
Read in	Landmark angle

When the computer wishes to communicate a request for data or signal an alarm to the astronaut, the "verb" and "noun" numbers flash until the astronaut takes action. He enters data to the computer through the keyboard which is on the right hand side of the display as seen here.

A schematic representation of the operation of the manned space flight network tracking system (MSFN) is shown in Fig. 13. The vehicle is illuminated by an 85 ft. antenna which provides range, angles, and velocity. This information is transmitted to the Mission Control Center in Houston from which navigation information is determined. The vehicle can also be tracked by 30 ft. antennae which use three-way doppler information to provide position and velocity data.

Distance is determined by modulating the carrier with random digits (0 and 1). The signal is received by transponders in the CSM or the LEM and retransmitted. The measurement of transit time of the signal is a measure of the distance of the spacecraft. Velocity is determined by measuring the doppler shift in the signal returned by the spacecraft.

Fig. 14 shows the location of MSFN unified S-band station sites. There is overlap of coverage among the three 85 ft. stations. These stations are located in the western United States (Goldstone, California); Madrid, Spain; and Canberra, Australia. Additionally, there are ten 30 ft. antennae spaced strategically around the world (Bermuda, Carnarvon, Guaymas, Hawaii, Cape Kennedy, Corpus Christi, Houston, Guam, Ascension, and Antigua).

Fig. 15 summarizes the characteristics of the tracking stations. The one-sigma tracking accuracies are as follows:

Range: Noise 60 ft. Bias 120 ft.

Angles: Noise 0.8 milliradians Bias 1.6 milliradians

Two-way doppler: Noise 0.1 ft/sec Bias .07 ft/sec

Three-way doppler: Noise 0.1 ft/sec Bias 0.2 ft/sec

The frequency reference is a rubidium frequency standard with a short and long term stability of 5×10^{-11} parts per part. The MSFN stations with either the 30 or 85 ft. antenna can track spacecraft at lunar distances using either the high-gain antenna or omnidirectional antenna.

Fig. 16 summarizes the operating modes of the system during a mission. During the earth orbital phase, the system uses C-band and measures range and two angles. During the cislunar phase, the system uses the unified S-band measuring three-way doppler.

Onboard derived navigation data is telemetered to the MSFN. The radar tracking data and the telemetered data are piped into the Mission Control Center at Houston and the guidance and navigation parameters are then computed in the real-time computation center (RTCC). After the guidance and navigation data is determined, it is telemetered to the Apollo guidance computer onboard the spacecraft.

Fig. 17 shows the measurement uncertainties of the MSFN system. Here we assume that the system is operating in the three-way doppler mode with one 85 ft. station transmitting and two stations receiving. The one sigma measurement uncertainty in moise and bias for two-way doppler and three-way doppler were

indicated in Fig. 15. Additionally, we assume an uncertainty (one sigma) in location of the station in latitude and longitude to be 1 to 6 arc seconds. The altitude error is assumed to be 100 to 200 ft. The uncertainty of the gravitational constant (one sigma) for the earth is assumed to be 106×10^9 ft^3/sec^2. For the Moon, this is assumed to be $6 \times 10^9 ft^3/sec^2$. In the orbit determination program, the parameters estimated are three components of velocity, three components of position, and two three-way doppler biases. Using these assumptions, the accuracy in performance of the system was computer and will be discussed shortly.

Let us look now at the techniques for navigating onboard the spacecraft. One technique for navigating is shown in Fig. 18. Here a star is picked up with the scanning telescope and centered, at which time the astronaut shifts to the sectant (a 28-power narrow field-of-vision instrument). The astronaut uses the sextant to position accurately the star over the landmark. When the star and the landmark are superimposed, the astronaut presses a button and the angles between the two, as well as time, are entered into the computer automatically. This information is entered by the astronaut via the DSKY. Fig. 18 shows a star superimposed on the Golden Gate Bridge.

In earth orbit the astronaut can measure his position by tracking known landmarks which are entered into Keplerian equations in the computer. It is also possible, by knowing the error propogation equations, to track unknown landmarks and to compute his position accurately in earth or lunar orbit. At the same time the computer determines the geographic position of the unidentified landmark. When the Apollo program was initiated, it was planned that known landmarks would be tracked and navigation would be performed in this way. During the many Gemini flights, however, it has been observed that it is quite difficult to plan ahead for those landmarks which will not be obscured by clouds. On most Gemini missions much of the earth has been obscured. If one is limited to known and predetermined landmarks, mission planning becomes quite complex. Therefore, it is likely that greater and greater reliance in Apollo will be placed on unknown landmark tracking for both earth orbital and lunar orbital navigation.

Let us look at the geometry of measuring a navigational fix in cislunar space. Fig. 19 shows the various angles involved using the stars Fomalhaut, Deneb, and Antares. In this geometrical sketch, the lunar horizon is used with Antares and navigational landmarks are used with Fomalhaut and Deneb. The angles measured with the three stars form three cones in space. The

intersection of two of these cones forms a line and the intersection with the third cone forms a point.

Computation is performed automatically in the computer. Apollo does not use the conventional method of computation performed by mariners for many centuries, i.e., computation based upon two or more star sightings and running the earlier sightings forward to the last sighting and computing a fix. This we call "deterministic" techniques. In Apollo, recursive navigation techniques are used (involving Kalman filter theory). Under this concept the accuracy of position and velocity determination is improved as more and more sightings are taken and the uncertainties are reduced with each sighting. This method involves statistical mathematics techniques. Fixes as they are known in maritime navigation are not performed in Apollo.

Another method of navigation originally planned in Apollo is shown in Fig. 20. Here the elevation angle of a star is measured with respect to the earth horizon. It is common knowledge that a precise definition of the horizon is difficult in space due to cloud cover and the uncertainty in the definition of the terrain horizon. Apollo planned to use a horizon at 100,000 ft. to be measured by a horizon photometer operating at a specific narrow frequency in the optical band. This horizon measurement is made automatically and the angular information is used in much the same way that mariners use elevation angle. The computational procedure involves recursive navigation cited earlier. Due to technical problems in the development of the horizon photometer electronics, this system will not be flown in early Apollo spacecraft.

Fig. 21 shows the accuracy of the navigation and guidance system using the optics in the way cited previously. In the earth orbital phase scanning telescope uncertainties are 4 milliradians and landmark position uncertainties are 1000 ft. Nominally, seven landmark sightings are made. During the translunar and transearth phases, the sextant uncertainty is 10 arc seconds and the uncertainties in the horizon are 1 nautical mile for Earth and 0.5 nautical miles for Moon. Additionally, we assume that forty landmark sightings are made enroute to the Moon and enroute back to the Earth. The velocity corrections made enroute to the Moon and returning to the Earth are accurate in magnitude of 1%. The pointing of the thrust vector is accurate to 10 milliradians.

Fig. 22 continues with the accuracies of the onbaord navigation system. In the lunar orbital phase, the scanning telescope uncertainty is 4 milliradians and the landmark position

uncertainties on the Moon are 1000 to 5500 feet in a horizontal direction and 2300 to 3000 feet in a vertical direction. Five landmarks on the earthside are chosen along the lunar orbit track and three sightings per landmarks are made on each of the first two orbits. Periodic updates are made thereafter.

GUIDANCE AND NAVIGATION PERFORMANCE

Let us look now at the performance of the system using the performance of the manned space flight network (MSFN) and the onboard system discussed previously. Fig. 23 summarizes the results of a digital computer simulation program using the accuracies cited previously. The RMS position measured by the onboard system and by the MSFN is given in nautical miles and velocities are measured in feet per second. The accuracies are shown at injection as measured either by land stations or by ship. Also shown are accuracy at the first midcourse, second midcourse, and third midcourse corrections enroute to the Moon and at perilune. In lunar orbit, the accuracies are shown at LEM separation and LEM rendezvous. The return accuracies are shown for injection at the transearth phase and at first, second, and third midcourse corrections and at entry. You will note that at Earth reentry, the MSFN accuracy is 0.5 nautical miles while the onboard system accuracy is about 9 nautical miles. With the MSFN, velocity is accurate to 2.2 ft. per second while the onboard system is inaccurate to the extent of about 45 ft. per second.

Fig. 24 is a comparison of the data as measured onboard and by the MSFN during the translunar phase. The top plot is position and the bottom plot is velocity as a function of time. The MSFN system is more accurate until about 35 hours after the spacecraft is injected toward the Moon. At that time the onboard system is of somewhat greater accuracy. The midcourse corrections are indicated here by the diamonds labeled MCC. These are made at two hours after injection, 49 hours after injection, and 61 hours after injection.

Fig. 25 gives surprising results concerning navigational uncertainties during the lunar orbit phase. The earth-based manned space flight network is more accurate than the onboard system throughout the trajectory. Time is measured from the time of spacecraft insertion into lunar orbit. The solid lines show points of MSFN tracking and the dashed lines show periods during which telescope sightings of lunar landmarks are made. Of course, errors grow during the period between tracking or sightings and they immediately drop to more accurate values as greater data is accumulated.

Fig. 26 shows the midcourse velocity performance during the translunar phase. The bottom plot shows the ft/second uncertainty in velocity correction which would be made as a function of time from injection to the first velocity correction as measured by the MSFN and the onboard system. The upper of the two figures shows the sum of the first two velocity corrections. There is a significant velocity savings by using MSFN data. These figures show that it is worthwhile to delay, from a fuel standpoint, before making velocity corrections. These figures are indicative of the fuel economies which could be made as the uncertainties in velocity corrections are reduced.

Fig. 27 describes in simple form the interface between the onboard system and the manned space flight network. The computations for the MSFN are made by the Houston Mission Control Center. The interface with the computer is made via two routes, the up and down data link directly from the MSFN to the Apollo guidance computer and voice communications to the crew who enter the data through the DSKY to the computer. The computer updata rate is 10 up-link words per second. The total navigation updata information required is a state vector consisting of six components and time. The updata frequency required is once prior to each guidance maneuver. Total transmission time is 60 seconds maximum (for a 99% probability of no errors). The navigational updata information is not time critical.

APOLLO REENTRY

The end result of the lunar mission is a safe reentry into the earth's atmosphere. Fig. 28 represents this problem. The guidance system must hit a corridor approximately 26 miles deep. This accuracy is obtained in three midcourse corrections during the return flight from the Moon. Prior to entering the earth's atmosphere, the service module is jettisoned. If the spacecraft comes in above this entry corridor, too little energy will be transferred from the vehicle to the atmosphere and the spacecraft will enter a highly elliptical earth orbit. The eccentricity of this orbit is a function of how far above the upper entry boundary the spacecraft trajectory carries it. Since there is no significant propulsion onboard the CM other than the 100 pound RCS system, there would be no way for the astronaut to recover from this highly elliptical orbit. It is necessary that this entry corridor be entered. If the astronauts come in too low they will exceed the heat capabilities of the heat shield.

One of the most critical portions of the mission from the guidance and control standpoint is the reentry phase. In Apollo

the earth's atmosphere is entered by making one skip. The total range of the vehicle from the point of entry is nominally 2500 miles. No propulsion system is used to remove energy from the vehicle; all of the energy must be removed by the atmosphere. We expect to land within 30 miles of the chosen landing point.

ENTRY TECHNIQUES

There are many different techniques which may be used to guide a spacecraft during the atmospheric entry maneuver. The selection of a particular technique is influenced by a number of considerations, some of which are shown in Fig. 29. The most important consideration which will influence the entry guidance is the requirement for the safety of the crew. The velocity and angle at which the spacecraft enters the atmosphere affects the entry guidance. The entry guidance must consider the physical properties of the spacecraft and the amount of range control which is required. The entry guidance is limited by the physical quantities which can be measured and by the size and speed of the onboard computer. There must be some technique for monitoring the primary entry guidance in order to insure a safe entry. The primary guidance system and the entry monitoring system must be compatible with each other.

The maximum permissible acceleration is normally considered to be 10 g's although the maximum emergency acceleration may be as high as 20 g's. The protection from the aerodynamic heating is provided by the heat shield, but entry guidance must minimize heating problems.

The Mercury and Gemini spacecraft entered at near circular velocities while the Apollo spacecraft enters at near parabolic velocities. The task of the entry guidance system becomes more difficult as the entry velocity increases. The entry angle must be kept within a value which will allow the spacecraft to be captured by the atmosphere and a value which will not result in excessive acceleration.

The aerodynamic properties of the entry vehicle may be divided into the general categories shown in Fig. 30. The first group consists of vehicles such as the Mercury spacecraft which do not develop any lift and which have a constant ballistic number (W/C_DS). In this discussion the aerodynamic properties are considered to be variable only if the values of these quanties can be controlled by the guidance system. Although the aerodynamics properties are termed constant they may vary as functions of Mach number and Reynolds number. The second group

listed in Fig. 30 are vehicles which do not develop lift but whose drag properties can be varied. Such a vehicle would have a small amount of range control. The third group includes both the Gemini and Apollo spacecraft. The vehicles of this group develop lift but their lift-to-drag ratio (L/D) and their ballistic number are constant. The trajectory which is followed by a spacecraft of this type can be controlled only by rolling the spacecraft. This changes the direction of any lift which is developed. Finally there is the last group of vehicles for which the ratio of lift-to-drag and the ballistic number can be varied.

ENTRY GUIDANCE SCHEMES

An entry vehicle which is designed to develop lift to change the path of the vehicle overcomes many of the problems which are inherent with a ballistic vehicle. The guidance system of such a vehicle must be capable of utilizing this lift properly. Many of the characteristics of the entry guidance will depend upon the amount of lift which can be generated and the manner in which the lift is generated. Both the Gemini and Apollo spacecraft are trimmed at a constant angle of attack. This results in a constant lift-drag ratio and a constant ballistic number. Therefore the only way that the path of the spacecraft can be controlled is by rolling the spacecraft about its longitudinal axis.

The possible entry guidance methods may be divided into two general classifications:

(1) Guidance using predicted capabilities

(2) Guidance using a nominal trajectory

The choice of which type to use depends upon considerations such as the size and speed of the onboard computer and the range of entry conditions which the guidance system must be capable of handling. The dividing line between the two classifications is somewhat obscure. It is possible that an entry guidance logic will use elements of both techniques.

In the method of guidance about a nominal trajectory, the state variables along the nominal path are precomputed and stored onboard the spacecraft. The variations in the measured variables from the stored values are used in the guidance logic either to control the spacecraft back to the nominal trajectory (path controller) or to establish a new trajectory to reach the destination (terminal controller). For this guidance logic, a desirable nominal trajectory must be selected. The desired

nominal trajectory is selected prior to the entry by optimization procedures.

The method of guidance using predicted capabilities is capable of handling a wider variety of entry conditions than the guidance about a nominal trajectory. This guidance technique predicts the path by which the vehicle will reach the desired destination without violating the heating and acceleration limits. The prediction of the future trajectory may be accomplished by a rapid forward integration of the equations of motion for the remainder of the flight, or by using approximate closed-form analytic solutions to the equations of motion. The main advantage of the fast prediction method is that it is able to handle any possible flight condition. The principal disadvantage of this method is the requirement for speed in the computer. The use of closed-form solutions reduces the required computational speed and flexibility of the guidance system.

GEMINI ENTRY GUIDANCE

Projects Gemini and Apollo furnish this country with experience in the entry guidance of lifting manned spacecraft. Two forms of entry guidance logic are used in the Gemini program. The same footprint capability of about 600 nautical miles across range exists for each technique. The two techniques may be termed rolling entry and fixed-bank entry.

In the rolling entry technique the steering logic is based on calculating the difference between the actual range to go and the predicted range based on a continuously rolling entry. During entry, the ratio of the downrange error to the crossrange is used to compute the bank angle required to rotate the axis of symmetry of the footprint the amount required to pass through the desired touchdown point before the downrange goes to zero. When this rotation is completed, the spacecraft flies at zero bank angle until the downrange error is zero. At this point a continuously rolling entry is initiated. Because downward lift is not used, the range calculation is biased slightly to predict a greater range than the time value, thus preventing the target from moving outside the footprint. The continuous rolling will therefore be interrupted occasionally for correction by flying near zero bank angle.

The second technique appears to have certain advantages over rolling entry. The entry trajectory is flown at a fixed series of bank angles. The prediction is based on the range obtained

at a fixed bank angle and the bank angle is adjusted to make the predicted range agree with the desired range to the touchdown point. Crossrange error is allowed to increase until it reaches a fixed percentage of the lateral range capable at that particular time. The bank angle is then reversed. Thus the entry is flown in a series of reversals of an essentially constant bank angle. Theoretically, any point in the footprint is available with only one bank angle reversal, but guidance inaccuracies will generally require additional maneuvering.

The fixed-bank-angle method has the advantage of being more compatible with the crew monitoring function than the rolling entry method. By viewing the horizon out the spacecraft windows the astronauts can compare the maneuvers with those expected for the entry condition and take over control in the event of a malfunction. The manually controlled backup technique is similar to the automatic method, but uses a precomputed program of bank angles. The fixed-bank-angle method will require less attitude control fuel than the rolling method. Figure 31 summarizes the results of reentry during the Gemini program to date. The early flights of Project Gemini dramatically demonstrated the effect of the difference between the estimates of the aerodynamics obtained from wind tunnel tests and the aerodynamics of the actual spacecraft. The unmanned flight of GT-2 indicated that the L/D ratio of the spacecraft was higher than the predicted value over most of the Mach number range. This led to the removal of 58 pounds of non-functional ballast for the first manned flight (GT-3) piloted by Grissom and Young.

The rolling entry technique was used for the entry of GT-3. This required the crossrange error to be eliminated first and then the downrange error was to be eliminated. One purpose of this flight was to check out the onboard guidance system. For this reason the spacecraft was controlled manually to the ground-computed commands and the commands generated by the onboard guidance were observed. The crossrange error was eliminated. Although full position lift was used for the remainder of the flight, the GT-3 spacecraft fell 64 nautical miles short of its target point. If the bank angles computed by the onboard guidance have been used, the spacecraft would have come closer to the target but it would not have reached the target.

The GT-3 spacecraft fell short of the target, chiefly, because it developed a lower L/D than had been estimated before the flight of GT-2. The footprint of the Gemini spacecraft was

reduced to about one-third of its original estimate size because of the reduced L/D. The estimated footprint is about 200 nautical miles long and 34 nautical miles wide. The length is measured from the zero lift point to the maximum lift point.

It was felt that the reduced footprint was not any cause for concern. It was decided to readjust the retrofire times in order to place the reduced footprint over the target point. Also, the crossrange and downrange errors were to be eliminated simultaneously at the beginning of the entry when the effective lift is the greatest.

The GT-4 spacecraft missed the target point by 47 nautical miles. The guidance computer had failed prior to entry. The spacecraft was flown on the basis of information furnished by the ground-based computers.

As in the case of the first two manned Gemini flights, GT-5 fell far short of its target point. In fact, it was the worst miss of the program. The spacecraft fell 97 nautical miles short of the target. Subsequent study disclosed that most of the miss was due to a ground error in failing to provide the onboard computer with the spacecraft's proper inertial coordinates at the time of retrofire. The information computed by the ground system ignored the fact that the Earth rotates approximately 361 degrees in 24 hours instead of 360 degrees. This error was present during the flights of GT-3 and GT-4 but the duration of these flights was not long enough for the effects of this error to become pronouned. By the end of the eight-day GT-5 mission this error was approximately 8 degrees. As a result of this error the onboard guidance attempted to steer the spacecraft to a target which was approximately 480 nautical miles closer to the actual target point. The miss distance would have been greater except that the Gemini entry guidance is prohibited from rolling the spacecraft to develop negative lift. The entry guidance called for a zero-lift entry which is the minimum lift allowed. By the time the astronauts realized that the guidance was giving erroneous commands, the range capability of the spacecraft was insufficient to reach the target point. During the entry the guidance was providing the correct commands to guide the spacecraft to the false target.

The flights of GT-6 and GT-7 indicate that the problems previously experienced with the entry have been solved. The actual landing point of GT-6 was within 7 nautical miles of the target point. The spacecraft was manually controlled

to the bank angles commanded by the onboard computer. The retrograde maneuver was performed with the acceptable tolerance. After the retrofire and the jettisoning of the retro-adapter section, the spacecraft was rolled to the full positive lift position. At 290,000 feet the onboard computer fed bank angle commands to the flight director displays. The nominal bank was first 47 degrees left and then 47 degrees right. These bank angles were held to 80,000 feet. The drogue parachute was deployed at about 50,000 feet and the main chute was deployed at about 10,500 feet. The landing point of GT-7 was also within 7 nautical miles of the target point.

APOLLO ENTRY

The obvious difference between the guidance problems of Gemini and Apollo is the difference between the entry velocities of the two missions. The Gemini spacecraft enters at near circular velocities while the Apollo Spacecraft will enter at near parabolic velocities. The much higher entry velocity of the Apollo spacecraft greatly increases the possibility than the acceleration limits will be exceeded. The range of a spacecraft entering at near parabolic velocities can be increased by allowing the spacecraft to skip out of the atmosphere. The time of flight may exceed the duration of the power supply or the lift support system, if the exit velocity is too high.

The Apollo entry guidance is a combination of the technique of using predicted capabilities and the technique of using a nominal trajectory. The guidance logic during the initial phases uses predicted capabilities while the guidance logic during the final phase uses a nominal trajectory. The logic is divided into portions which reflect the characteristics of the entry trajectory which the Apollo spacecraft will follow.

The typical Apollo entry trajectory can be divided into the general areas which are shown in Figure 32. At some time prior to entry, the entry vehicle is separated from the Service Module and aligned to the entry attitude; the IMU is aligned and the navigation system is updated for the last time. The initial bank angle depends upon whether the vehicle is entering at the top of the corridor or at the bottom. If the spacecraft is entering close to the top of the corridor, the bank angle will be such that the lift is directed downwards in the vertical plane. Otherwise, the lift is directed upwards in the vertical plane.

Upon encountering the sensible atmosphere the remaining phases of the entry are:

(1) An initial phase during which a safe capture is ensured and excessive acceleration is avoided.

(2) A second phase during which the vehicle is steered so that the final phase will be able to guide the spacecraft to the target.

(3) A ballistic lob which may be bypassed if the range to the target is short.

(4) The final phase during which the spacecraft is steered to the target. This final phase usually includes the last six to eight hundred miles of the entry and is similar to the entry from a low orbit about the Earth.

The basic flow of the guidance logic is shown in Figure 33. Certain portions at the beginning and end of the system are entered each time the steering commands are computed. These portions are the NAVIGATION, TARGET DATA, and ROLL COMMAND sections. The LATERAL LOGIC section is also entered each time the steering command is computed except during the initial phase of the entry and during any ballistic lob. If the spacecraft is returning from a low altitude orbit, the PHASE SELECTOR will be set so that only the BALLISTIC PHASE and the FINAL PHASE are used.

The NAVIGATION section calculates the inertial positon and velocity of the vehicle by using simple numerical integration techniques. The measured acceleration is combined with a calculated gravitational acceleration and then is used in the equations of motion.

The TARGET DATA section calculates the distance between the position of the spacecraft obtained by the NAVIGATION section and the inertial position of the target at the estimated time of arrival. The inertial velocity of the spacecraft is used until the velocity becomes less than approximately one-half the circular velocity at an altitude of 300,000 feet altitude.

The PHASE SELECTOR directs the logic to the section which is concerned with the phase of the entry that the spacecraft is in at that time. Initially, the PHASE SELECTOR will direct the logic to the INITIAL ROLL section or to the BALLISTIC PHASE section. The correct path will depend on whether the spacecraft is returning at near parabolic velocity or at near circular velocity.

During the initial phase of entry, the bank angle is held constant until the drag exceeds 2 g's. At this time the vehicle is rolled to a zero-bank angle which is held constant until the second phase is entered. The second phase is entered when the altitude rate is greater than -700 feet per second.

The second phase is the heart of the Apollo entry guidance system. This section inclues the logic which will decide the trajectory which will be flown in order to reach the target. There are three paths which the logic may follow after this section. The three paths are to the CONSTANT DRAG section, to the UP CONTROL section, or to the FINAL PHASE section. If the predicted exit velocity is greater than the circular velocity or the predicted range capability is greater than the actual range plus 25 nautical miles, the CONSTANT DRAG section is used. The second phase will be reentered during the next computation cycle which is two seconds later. The logic will be directed to the FINAL PHASE whenever the predicted exist velocity is less than 18,000 feet per second. The UP CONTROL section is entered only when the difference between the predicted range to the target and the range of the calculated trajectory is less than 25 nautical miles.

The CONSTANT DRAG section attempts to guide the spacecraft along a constant acceleration path. The value of the acceleration is calculated the first time the second section of the guidance logic is entered.

The UP CONTROL section attempts to guide the vehicle to the calculated exit conditions. The guidance system continues to use this section until the drag becomes less than some arbitrary value or the altitude rate becomes negative. If the drag falls below this arbitrary value, the guidance logic enters the BALLISTIC PHASE section. If the altitude rate becomes negative first, the FINAL PHASE section is entered.

During the ballistic lob, the BALLISTIC PHASE checks for the start of the final phase of the trajectory. The attitude of the spacecraft is controlled during the ballistic lob so that the sideslip is zero and the spacecraft is trimmed about its nominal angle of attack.

The FINAL PHASE section uses a stored table of values to attempt to steer the vehicle along a reference trajectory to the target. The steering command of the FINAL PHASE will be modified by the "G" LIMITER whenever the drag is greater than 5 g's. The bank angle will be decreased in an attempt to keep the drag level below 10 g's.

The output command of the CONSTANT DRAG, UP CONTROL, and FINAL PHASE sections is a value of L/D. The commanded roll angle is determined by the ratio of the commanded value of the L/D to the maximum value of the L/D.

The LATERAL LOGIC section is used to decide to which side of the vertical plane the vehicle should be rolled. Basically, the vehicle is rolled toward the target. To avoid a large number of roll reversals, there is a deadband built into the logic. That is, lift may be directed away from the target if the predicted landing point is within limits. This limit has arbitrarily set at on-half the lateral range capability.

The final section of the guidance logic is the ROLL COMMAND. This section selects the direction of roll which will result in the shortest angle to be travelled. The command from this section is then transmitted to the reaction control system.

The entry of a spacecraft at near parabolic velocities presents a number of sources of danger to the crew. The chief dangers are excessive accelerations and exiting along a trajectory which would exceed the lifttime of any of a number of onboard systems such as lift support, power, and attitude control fuel. It is desirable to have an independent and reliable system for monitoring the primary guidance system.

The entry monitoring system (EMS) must be sufficiently accurate to detect impending unacceptable trajectory characteristics such as excessive accelerations or an uncontrolled atmospheric ship in sufficient time to prevent their occurrence. The EMS must not unnecessarily restrict the performance of the primary guidance system and it must be at least an order of magnitude more reliable than the primary guidance system. Obviously something must be relinquished in order to achieve a more reliable system. In this case the capability for precise range control is lost but the system is still capable of gross range control.

The EMS is currently envisioned as consisting of four basic parts:

(1) An entry threshold indicator

(2) A corridor indicator

(3) A bank indicator

(4) A flight monitor

The entry threshold indicator is an on-off signal that is excited when the sensed acceleration is greater than some nominal value. The corridor indicator consists of two signals which are used to indicate whether the entry is at the top or the bottom of the corridor. The signals result from comparing the sensed acceleration to a nominal mid-corridor value at a discrete time interval after the threshold indicator is turned on. The back attitude indicator is a meter which indicates angular rotation about the approximate stability axes.

The flight monitor in the Apollo vehicle is a rectilinear plotter which presents a trace of the variation of total acceleration with the velocity of the vehicle. The astronaut compares this trace with information which is presented on the face of the plotter. Figure 34 shows a simple version of the information which is presented. Two families of curves are presented on the plotter.

One family serves to indicate if the acceleration limit is being exceeded and the second family is used to warn of an uncontrolled skip from the atmosphere. The high g lines are of interest only if the vehicle is not at aero bank angle. Whenever there is a danger of excessive accelerations the vehicle should be rolled to a zero bank angle.

The second family of curves are the most important since these are used to prevent an uncontrollable skip. For every combination of the acceleration and the velocity there exists a limiting rate at which safe atmospheric exits can be made. If the flight trace is compared to a set of rays which emanate from approximately zero g and an exit velocity which is less than the local circular orbital velocity, a safe limiting rate can be defined by tangency of the actual flight trace and the ray. Originally, these rays were straight lines such as shown in Figure 34, but it was found that there are times that the flight monitor would indicate a failure erroneously. This deficiency has been partially overcome by redefining the rays as curved lines and by shaping the entry trajectories so that a violation of the tangency criteria was more readily discernable.

LEM GUIDANCE AND CONTROL

Figure 35 shows the LEM guidance and control system. This is quite similar to the Command Module system, although the nomenclature is different. There are four major elements shown:

(1) Guidance and navigation system

(2) Stabilization and control system

(3) Reaction control system (attitude jets)

(4) Ascent and descent engines

The LEM G&N system has a landing radar and either a rendezvous radar or an optical tracker system for rendezvous purposes. The computer is the central data processor as shown. It receives data from the radar, DKSY, Alignment Optical Telescope (AOT), and the inertial measurement unit. Other inputs to the LEM guidance computer are from the attitude controller (which is the control stick used by the astronaut) and the throttle command. The computer drives either the ascent or the descent engines and the reaction control jets.

If the computer or the G&N system fails, the attitude controller and the throttle operate via the stabilization and control system to drive the ascent and descent engines. Either the stabilization and control system or the G&N system is used to provide attitude information on the FDAI (flight director attitude indicator) which is a gyro horizon. In the event the primary G&N system fails, the system also has an abort guidance system which is used to effect safe recovery into a rendezvous trajectory.

Figure 36 shows a more simplified block diagram of the LEM stabilization and control system showing the flow of attitude data, range data, timing, engine commands, and other information required for control, guidance and navigation.

Figure 37 is the current configuration of the LEM cockpit. The two windows, one for the command pilot and one for the pilot, can be seen to the right and left. The two FDAI's ("8-balls") are in the center of the instrument panel. Immediately above the FDAI's are pointers which indicate translational velocities, the delta V counter, throttle setting indicator, and propellant quantity gages. In the upper portion of the panel are main propulsion system and environmental control system control switches and indicators. Between the FDAI's are various subsystem control switch, vertical velocity indicator and thrust-to-weight ratio indicator.

The lower console in Figure 37 contains control switches and indicators for the stabilization and control system, power generation system, and cryogenic storage. At the bottom center of the lower console is the DSKY which provides a communications link between the astronaut and computer.

Let us look now at some of the gross features of the trajectory from lunar orbit to the lunar surface. This phase of the mission will be discussed in greater detail later in this symposium. Figure 38 shows this trajectory. The Command Module, Service Module, and LEM are initially in an 80-mile circular orbit around the Moon. The velocity of this orbit is approximately 6000 feet per second and the period is about two hours. This means that the spacecraft are hidden from the Earth about one hour out of every two hours. The operation of the onboard guidance and control system is important during these periods.

At such time as the vehicle reaches the proper point for lunar entry, an impulse of approximately 100 feet per second is applied by the descent engine. The descent trajectory is a Hohmann trajectory. After the vehicle has traversed 180° around the Moon, it reaches a perilune of about 50,000 feet. The trajectory is monitored by the astronauts in the LEM tracking the Command Module and by checking out the landing radar. It is also monitored by the astronaut in the Command Module who tracks the LEM through the Command Module optical system. Additionally, the earth-based ground tracking system tracks both vehicles.

The next phase of the landing maneuver is shown in Figure 39. This covers a distance of about 240 miles across the lunar surface. During this phase the vehicle proceeds from an altitude of 50,000 down to an altitude of about 10,000 feet. Here the LEM descent engine has been on for a period of about 450 seconds. After completion of this phase the vehicle enters a point known as "high-gate."

Figure 40 shows the "constant attitude" phase of the mission. This is the phase of the mission from about 10,000 feet down to approximately 500 to 1000 feet. During this phase of the mission the descent engine is throttle back to about 50% and the vehicle is pitched up into an attitude of about 35° to 45° with respect to the horizontal. During this phase of the mission the astronaut is surveying his landing site out the window.

One of the interesting problems associated with the lunar landing is that of visibility. If the sun angle is parallel to the trajectory path, i.e., if the sun is behind the astronaut, it tends to "wash out" the landing area such that he cannot discern characteristics of craters, etc. It is much like the reflection of the sun shining on the ocean. Therefore, the sun will be placed at such an angle as to be somewhat different from the angle of the landing trajectory

or else the astronaut will be required to make a dog-leg into a landing to a site which he can survey carefully off to the side. This problem is discussed in greater detail in another paper in this symposium.

After the astronaut proceeds to the "low-gate" point at 700 to 1000 feet, he takes over control of the vehicle manually. He flies to an altitude of about 100 feet, at which time he nulls out all accumulated drifts in the platform with the aid of the landing radar. He is then in a position to make an instrument landing or a visual landing depending upon the degree of obscuration of his vision by a cloud of dust which might be created by his engine plume.

If the astronaut should land with too great a horizontal velocity or if he should begin to tip over into a crater, he can abort the mission as long as he acts before the vehicle exceeds an angle of about 45° measured with respect to the vertical. Similarly, if the bearing strength of the lunar surface is of such a nature as not to support the vehicle and he notices that the vehicle is sinking, he can press a button can abort the mission. Considerable redundancy and safety has been built into the vehicle characteristics and into the landing trajectory to provide for pilot safety.

RELIABILITY

In Apollo, two reliability criteria are applied in system design and in mission planning. One is associated with crew safety and the other is associated with mission success. Figure 41 shows the reliability diagram of the navigation and guidance system and the control system from a crew standpoint. In the navigation and guidance system, the onboard system and the ground system are in parallel paths. The ground system consists of the Mission Control Center, communications systems, and the tracking network. The onboard system consists of the Apollo Guidance Computer, the optics telescope, and the inertial measurement unit.

In the control system there is even greater redundancy from the crew safety standpoint. The computer and inertial measurement unit are connected in parallel with the body-mounted attitude gyros and accelerometers. Also in parallel are the sextant and scanning telescope operating automatically or manually and dual thrust-vector control electronics and dual reaction control jet electronics.

The mission criteria for crew safety is such that if sufficient equipment malfunctions have occurred that one more malfunction of any kind would endanger the crew, then at that point the mission should be aborted. The realiability numbers are such that we can expect potential danger to one flight crew in 200 missions.

Figure 42 shows the mission success block diagram. In this block diagram, there are series connections of the guidance and control system rather than the parallel paths shown in the crew safety diagram. The probability of mission success, of course, is somewhat less than that for crew safety.

QUESTIONS AND ANSWERS

Speaker: Dr. Robert C. Duncan

No questions

NASA-S-64-6589

GENERALIZED GUIDANCE AND CONTROL SYSTEM

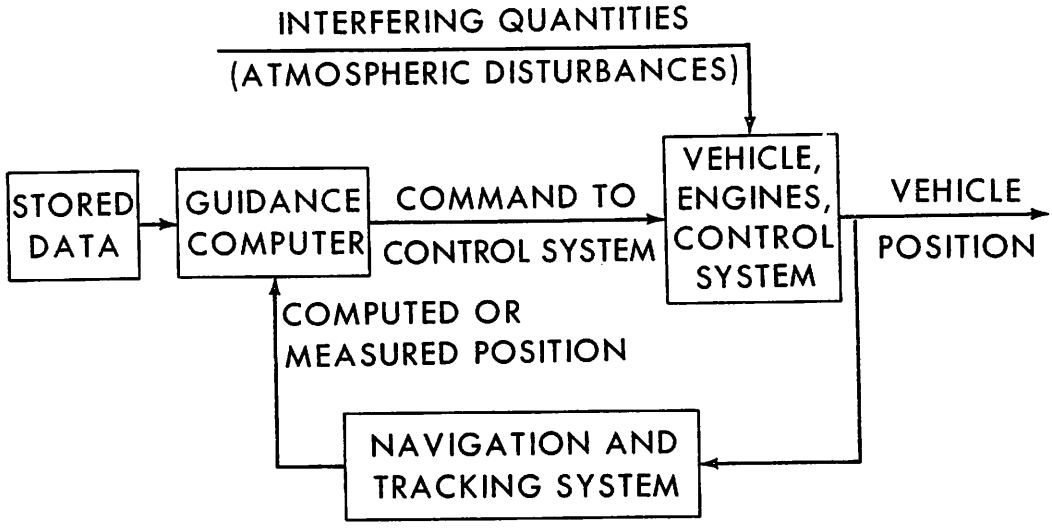


FIGURE 1

NASA-S-65-10001

EARLY PROGRAM GUIDELINES

- GROUND NAVIGATION CAPABILITY MANDATORY REQUIREMENT
- SELF-CONTAINED ON-BOARD NAVIGATION CAPABILITY MANDATORY REQUIREMENT
- ON-BOARD SYSTEM DESIGN CONCEPT TAKE MAXIMUM ADVANTAGE OF GROUND CAPABILITIES AND INCLUDE NECESSARY INTERFACES

FIGURE 2

NASA-S-65-4483

BLOCK II
N, G, AND C CONFIGURATION

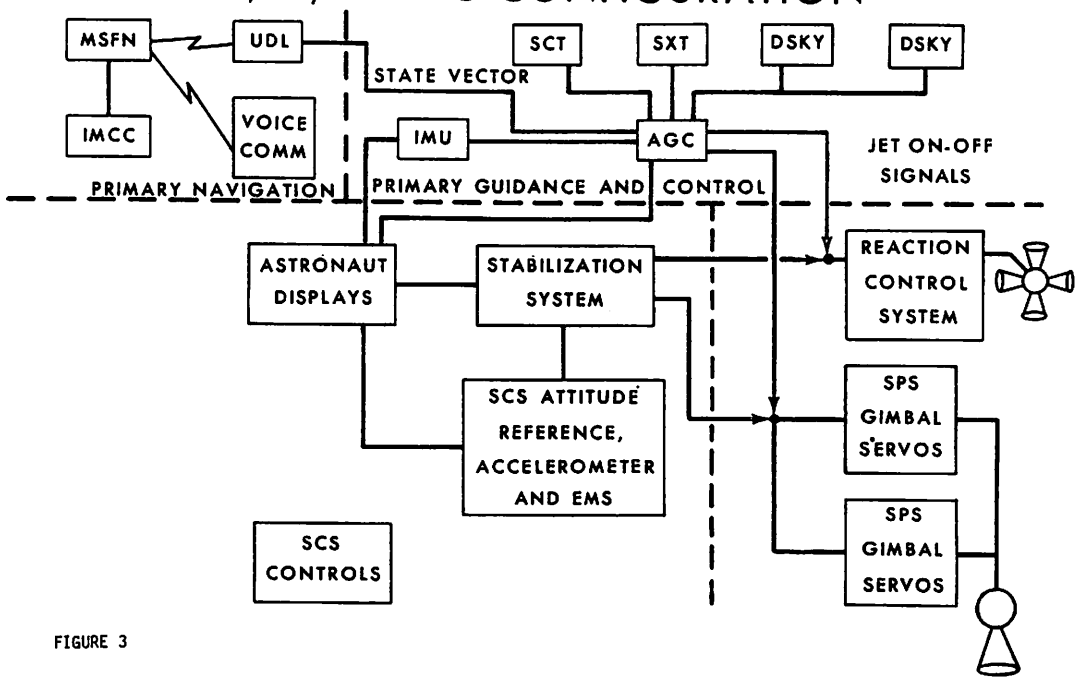


FIGURE 3

NASA S-63-10214

SCS BLOCK DIAGRAM

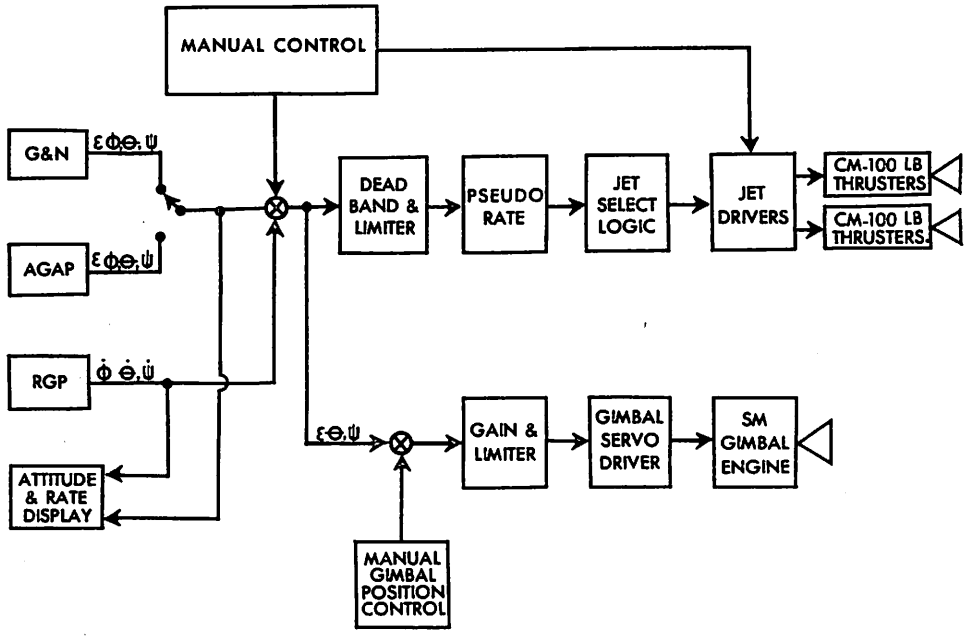


FIGURE 4

NASA-S-64-6653

GUIDANCE & NAVIGATION SPACECRAFT LOCATION

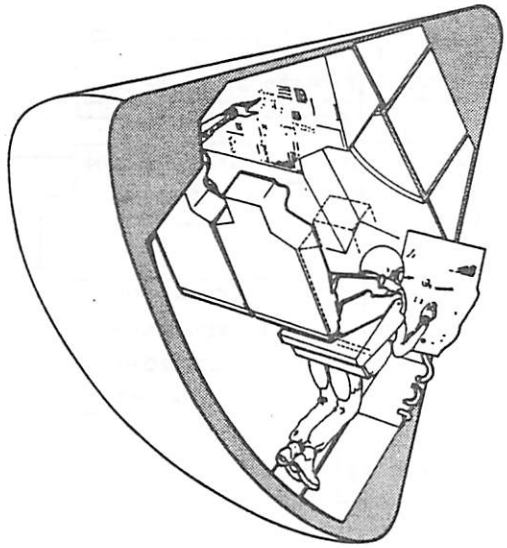


FIGURE 5

NASA-S-64-6638

GUIDANCE AND NAVIGATION SYSTEM IN HOLDING FIXTURE

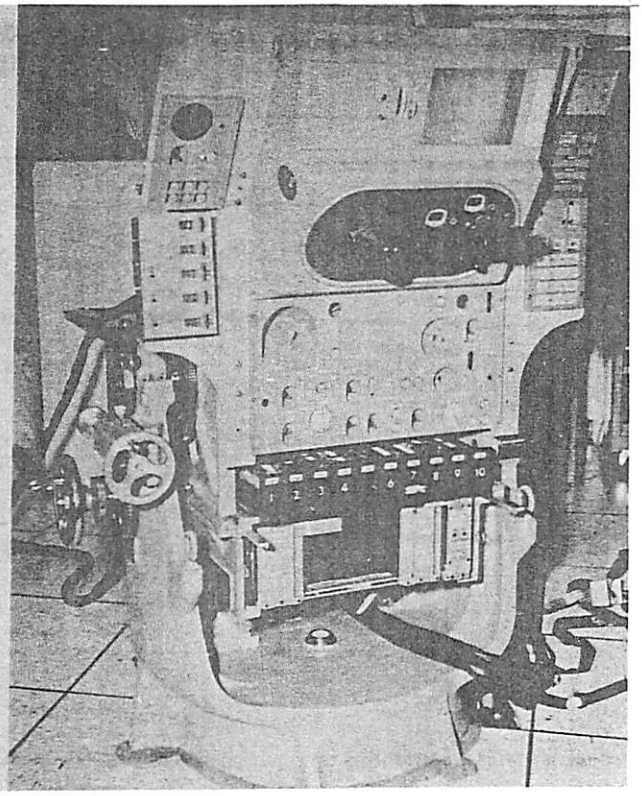


FIGURE 6

NASA S-63-10217

THREE DEGREE OF FREEDOM PLATFORM

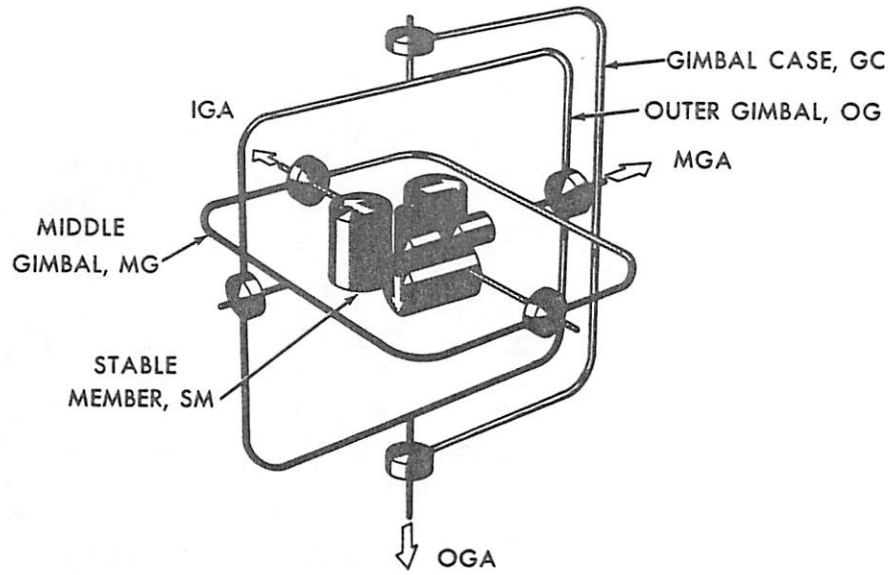


FIGURE 7

NASA-S-65-7496

INERTIAL MEASUREMENT UNIT

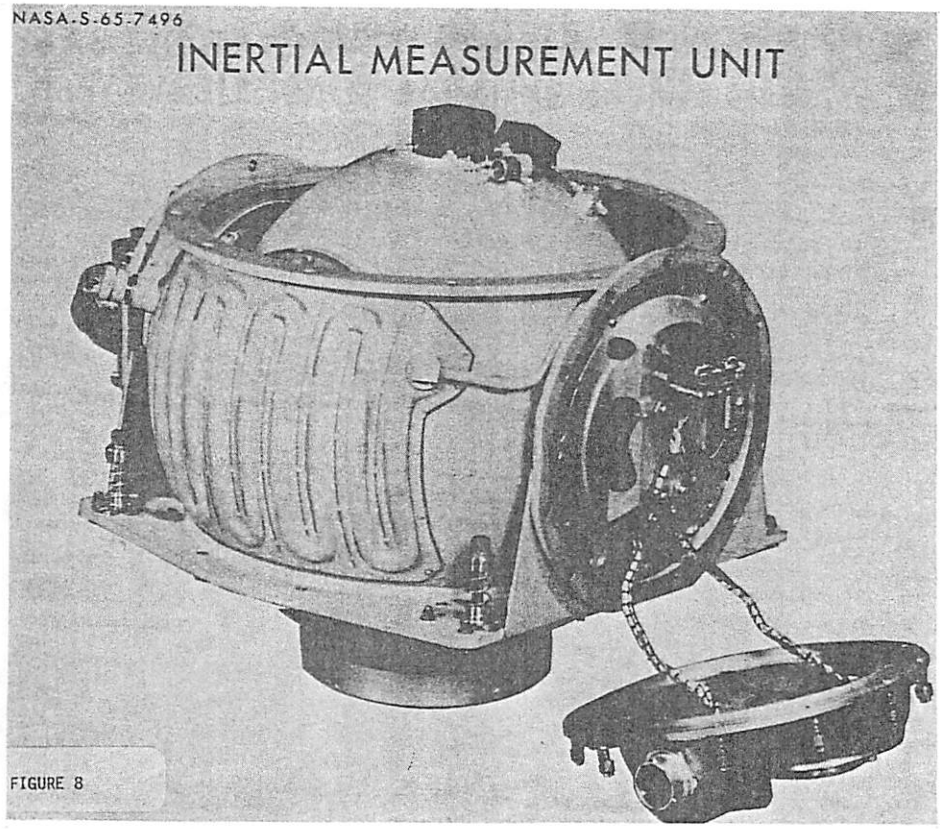


FIGURE 8

NASA-S-65-3973

APOLLO

INNER, MIDDLE & OUTER GIMBAL ASSEMBLIES

IMU-5 FOR APOLLO G&N EQUIPMENT

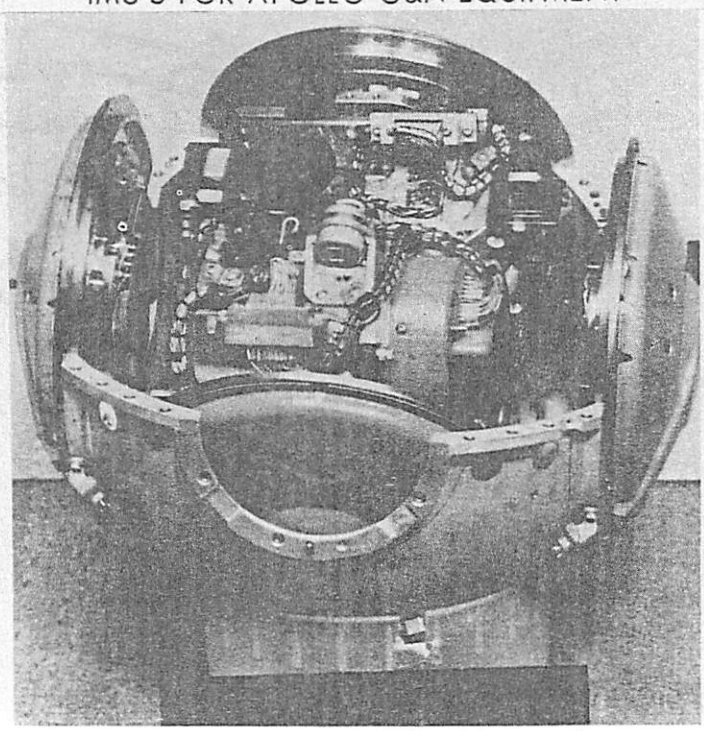


FIGURE 9

NASA-S-65-3978

THE IMU, NAVIGATION BASE AND OPTICAL SUBSYSTEM

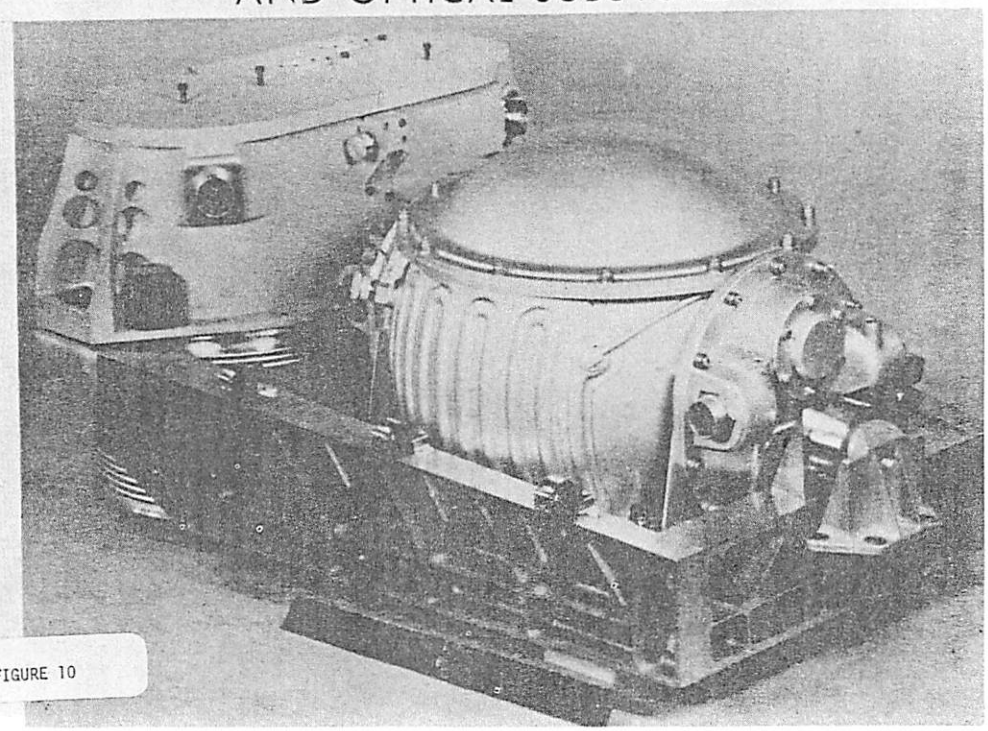


FIGURE 10

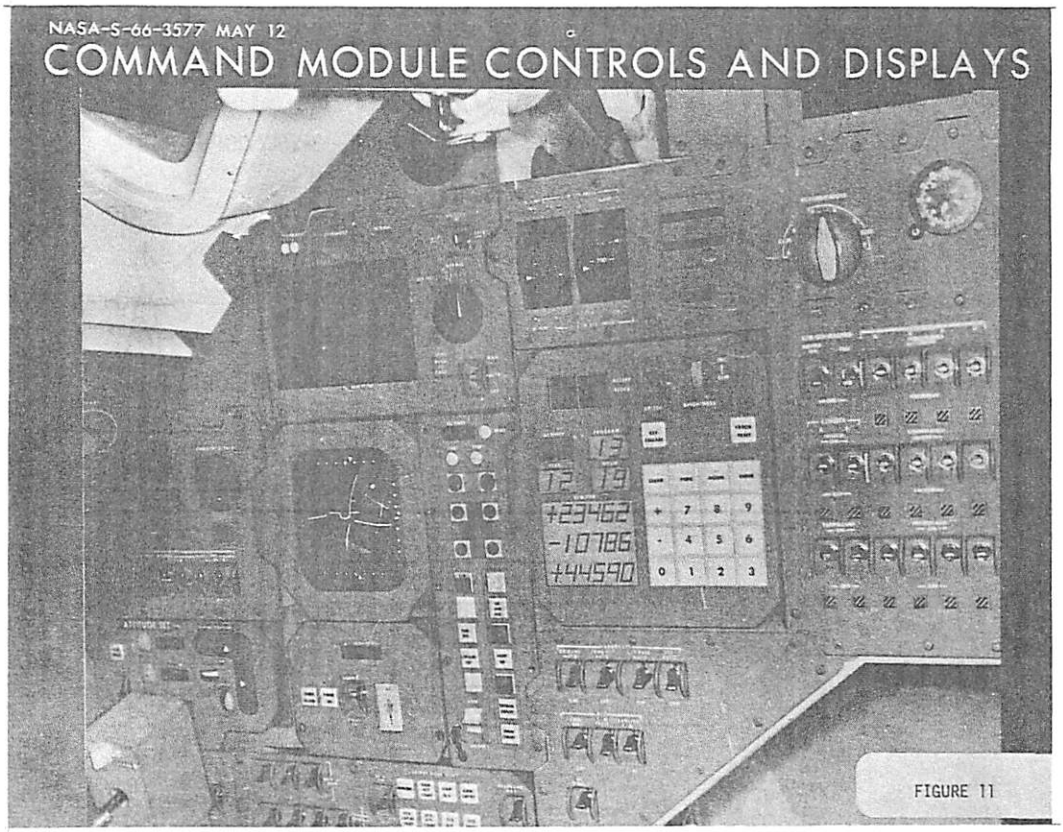


FIGURE 11

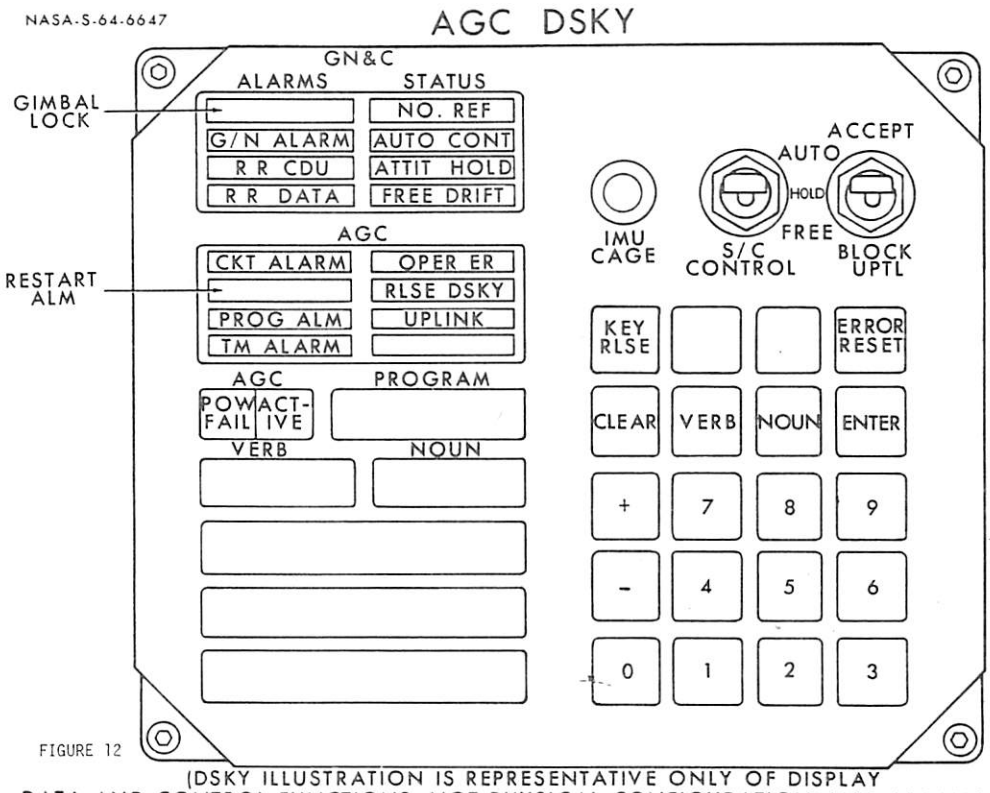


FIGURE 12

(DSKY ILLUSTRATION IS REPRESENTATIVE ONLY OF DISPLAY DATA AND CONTROL FUNCTIONS, NOT PHYSICAL CONFIGURATION AND ARRANGEMENT)

NASA-S-65-10004

MSFN TRACKING SYSTEM

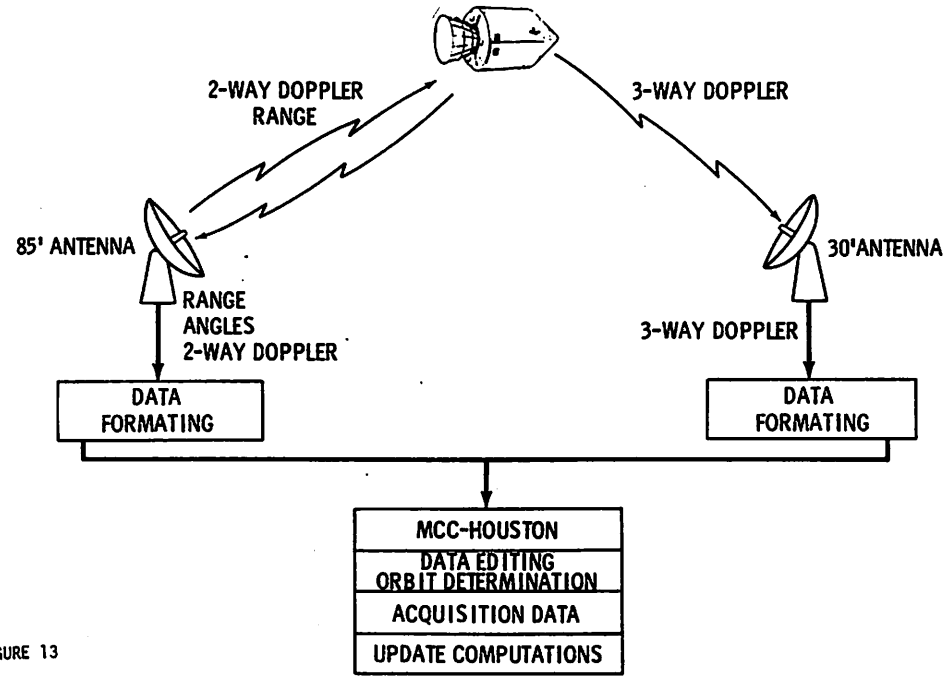
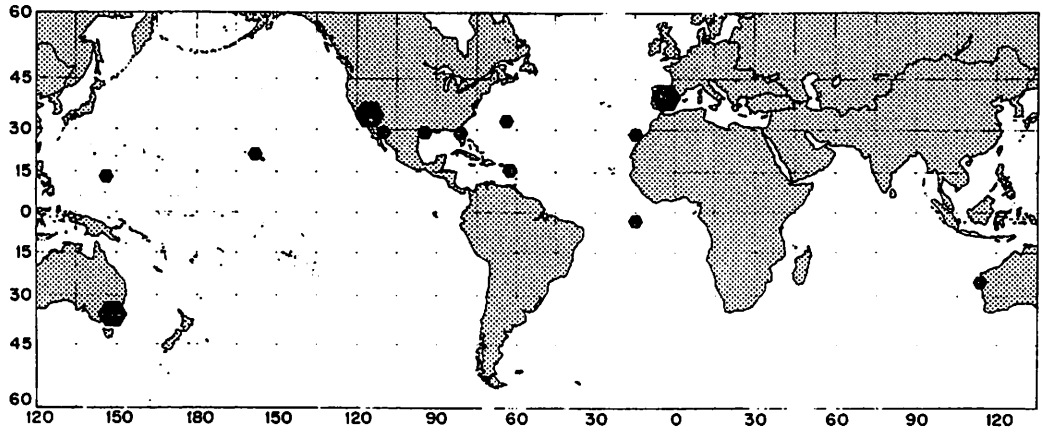


FIGURE 13

NASA-S-65-10011

MSFN UNIFIED S-BAND STATION SITES



DUAL SITES
MESSAGE SHARING
MULTIPLE SITE COVERAGE
OVERLAP OF COVERAGE BY 85 FT ANTENNAS

85 FT ANTENNA
30 FT ANTENNA

FIGURE 14

NASA-S-65-9997

STATION CHARACTERISTICS

- TRACKING ACCURACIES (ONE SIGMA)

MODE	NOISE	BIAS
RANGE	60 FT	120 FT
ANGLES	0.8 MR	1.6 MR
TWO-WAY DOPPLER	0.1 FPS	0.07 FPS
THREE-WAY DOPPLER	0.1 FPS	0.2 FPS

- FREQUENCY REFERENCE
  - RUBIDIUM FREQUENCY STANDARD
  - STABILITY (LONG AND SHORT TERM) - $5 \times 10^{-11} \frac{\text{PARTS}}{\text{PARTS}}$
- OPERATING CAPABILITIES
  - MSFN STATIONS WITH 30 OR 85 FOOT ANTENNAE CAN TWO-WAY DOPPLER TRACK WITH SPACECRAFT OMNIDIRECTIONAL OR HI-GAIN ANTENNAE AT LUNAR DISTANCES

FIGURE 15

NASA-S-65-10003

MISSION OPERATIONS

- MSFN TRACKING MODES

PHASE	SYSTEM	MEASURABLE
EARTH ORBIT	C-BAND	RANGE, 2 ANGLES
CISLUNAR	USBS	3-WAY DOPPLER AND RANGE, 2 ANGLES
LUNAR ORBIT	USBS	3-WAY DOPPLER

- DATA FLOW
  - ON-BOARD NAVIGATION DATA TELEMETERED TO MSFN
  - RADAR TRACKING AND TELEMETERED DATA PIPED INTO IMCC
  - GUIDANCE AND NAVIGATION PARAMETERS COMPUTED IN RTCC
  - MSFN GUIDANCE AND NAVIGATION DATA TELEMETERED TO ON-BOARD COMPUTER

FIGURE 16

NASA-S-65-9996

ASSUMPTIONS
MSFN NAVIGATION ACCURACY

- MSFN OPERATING IN 3-WAY DOPPLER MODE WITH ONE STATION TRANSMITTING AND TWO RECEIVING

- MEASUREMENT UNCERTAINTIES (ONE SIGMA)

	NOISE	BIAS
2-WAY DOPPLER	0.1 FPS	0.07 FPS
3-WAY DOPPLER	0.1 FPS	0.2 FPS

- STATION LOCATION UNCERTAINTIES (ONE SIGMA)
  - LATITUDE 1 - 6 SEC
  - LONGITUDE 1 - 6 SEC
  - ALTITUDE 100 - 200 FT

- GRAVITATIONAL CONSTANT UNCERTAINTIES (ONE SIGMA)
  - EARTH 106×10^9 FT^3 / SEC^2
  - MOON 6×10^9 FT^3 / SEC^2

- PARAMETERS ESTIMATED IN ORBIT DETERMINATION PROGRAM
  - THREE COMPONENTS OF VELOCITY
  - THREE COMPONENTS OF POSITION
  - TWO 3-WAY DOPPLER BIASES

FIGURE 17

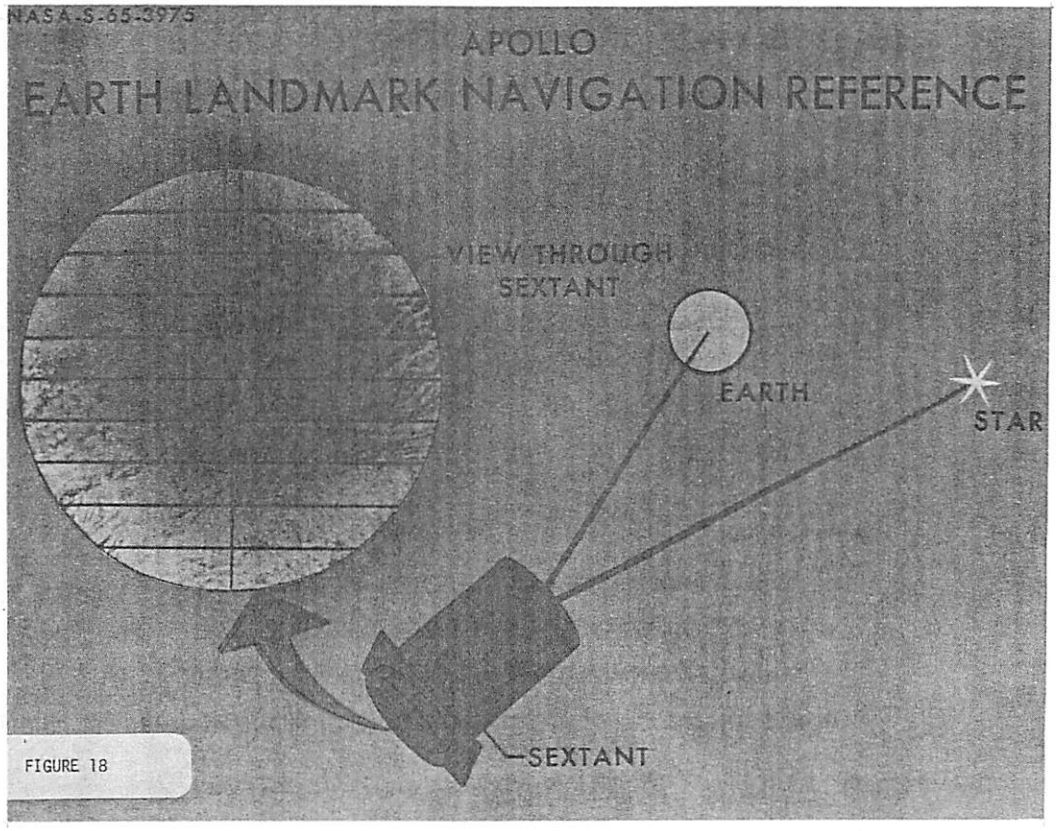


FIGURE 18

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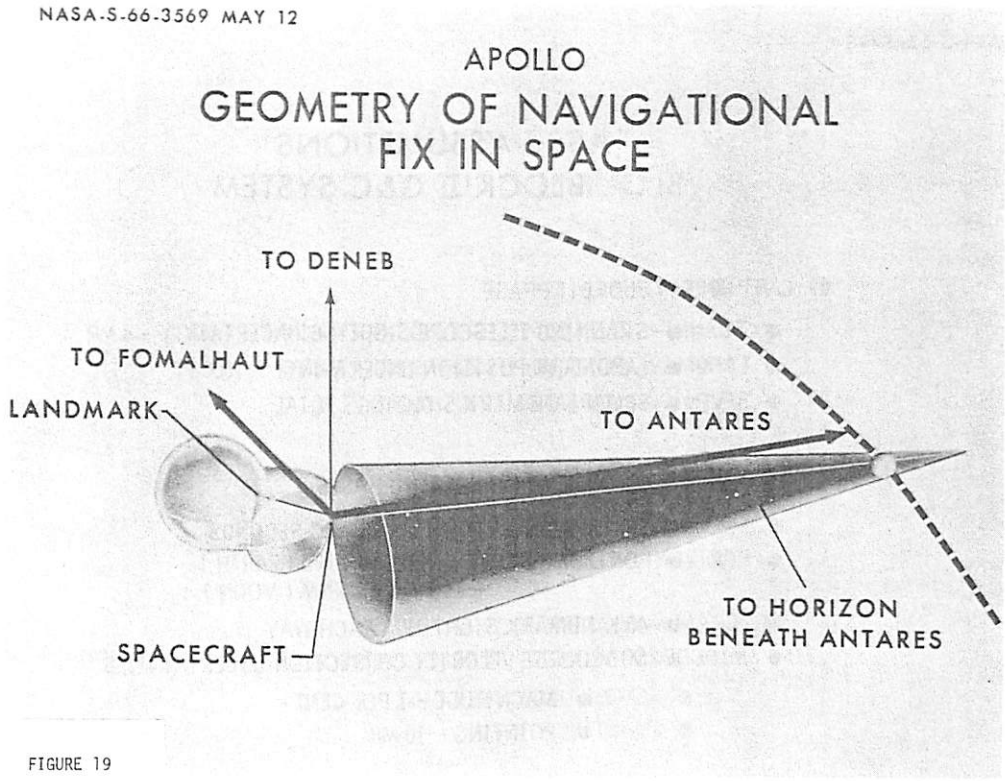

FIGURE 19

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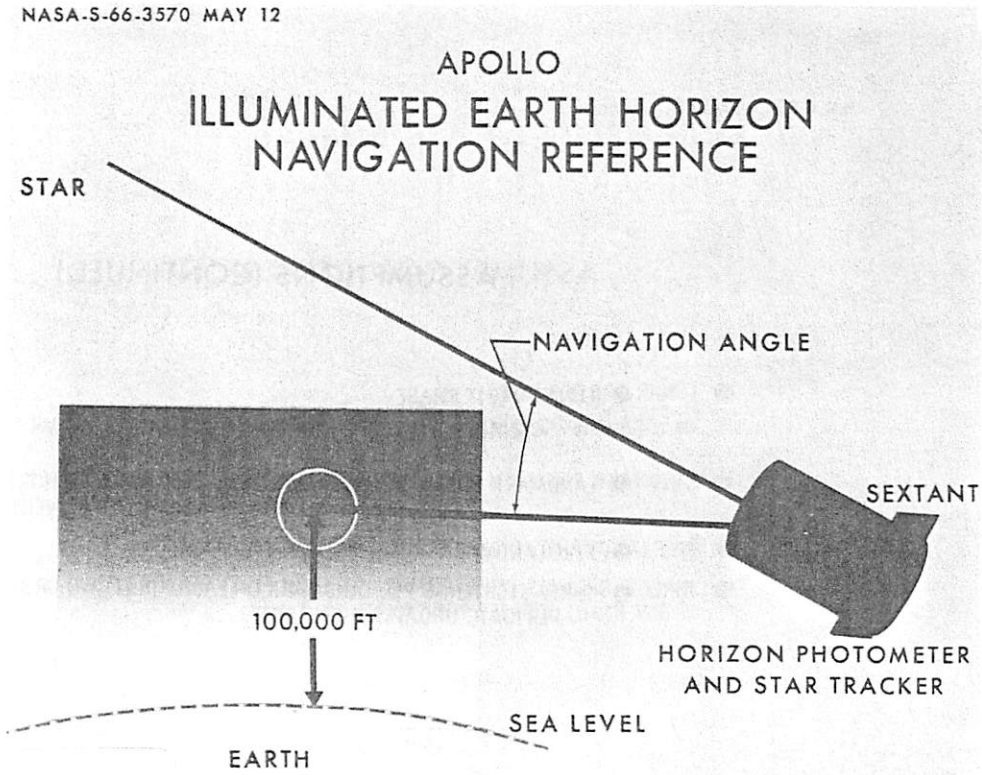


FIGURE 20

NASA-S-65-9998

ASSUMPTIONS
BLOCK II G&C SYSTEM

- EARTH ORBIT PHASE
  - SCANNING TELESCOPE SIGHTING UNCERTAINTY - 4 MR
  - LANDMARK POSITION UNCERTAINTY - 1000 FT
  - SEVEN LANDMARK SIGHTINGS TOTAL
- TRANSLUNAR / TRANSEARTH PHASES
  - SEXTANT UNCERTAINTY - 10 ARC SECONDS
  - HORIZON UNCERTAINTIES - 1 NM (EARTH)
.5 NM (MOON)
  - 40 LANDMARK SIGHTINGS EACH WAY
  - MIDCOURSE VELOCITY CORRECTION UNCERTAINTIES
    - MAGNITUDE - 1 PER CENT
    - POINTING - 10 MR

FIGURE 21

NASA-S-65-10000

ASSUMPTIONS (CONTINUED)

- LUNAR ORBIT PHASE
  - SCANNING TELESCOPE SIGHTING UNCERTAINTY - 4 MR
- LANDMARK POSITION UNCERTAINTIES 1000 - 5500 FT (HORIZONTAL)
2300 - 3000 FT (VERTICAL)
- FIVE LANDMARKS ON EARTHSIDE CHOSEN ALONG TRACK
- THREE SIGHTINGS PER LANDMARK ON EACH FIRST TWO ORBITS, PERIODIC UPDATES THEREAFTER

FIGURE 22

NASA-S-65-10006

COMPARISON OF MSFN/ON-BOARD NAVIGATIONAL UNCERTAINTIES

PHASE	EVENT	RMS POSITION, NM		RMS VELOCITY, FPS	
		MSFN	ON-BOARD	MSFN	ON-BOARD
TRANSLUNAR	INJECTION (LAND)	0.1	2.2	1.5	13.5
	(SHIP)	2.7		37.1	
	FIRST MIDCOURSE	1.2	5.4	0.7	5.2
	SECOND MIDCOURSE	10.3	8.4	0.2	0.4
	THIRD MIDCOURSE	3.1	2.5	0.8	1.1
	PERILUNE	1.2	1.2	10.2	5.0
LUNAR ORBIT	LEM SEPARATION	0.4	0.9	3.2	2.8
	LEM RENDEZVOUS	0.2	0.3	0.6	1.2
TRANSEARTH	INJECTION	0.4	0.4	1.0	1.8
	FIRST MIDCOURSE	6.0	4.4	0.3	0.7
	SECOND MIDCOURSE	4.8	7.1	0.2	0.6
	THIRD MIDCOURSE	1.4	8.7	1.1	6.5
	ENTRY	0.5	8.7	2.2	45.3

FIGURE 23

NASA-S-65-10032

NAVIGATION UNCERTAINTIES

TRANSLUNAR PHASE

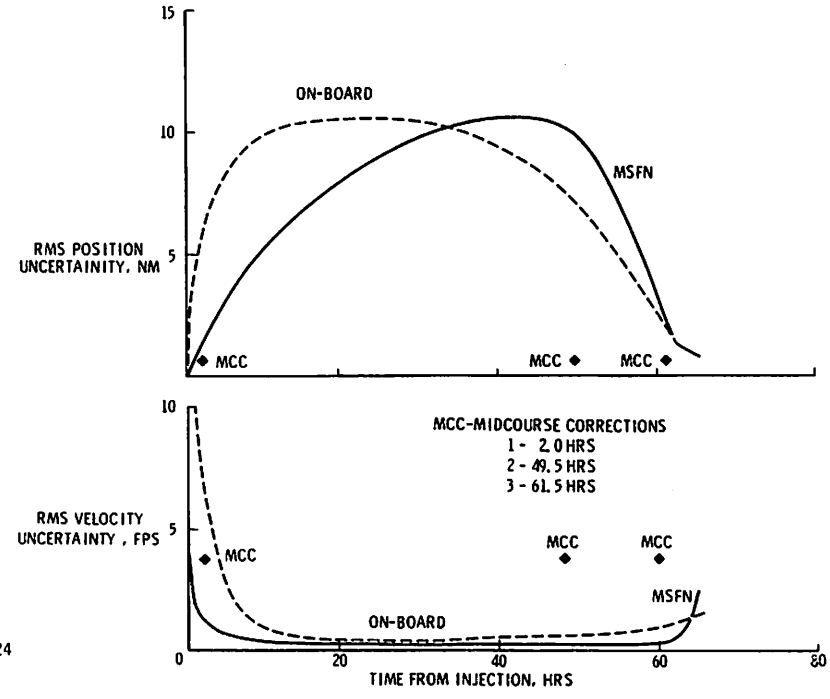


FIGURE 24

NASA-S-66-7324 JUL

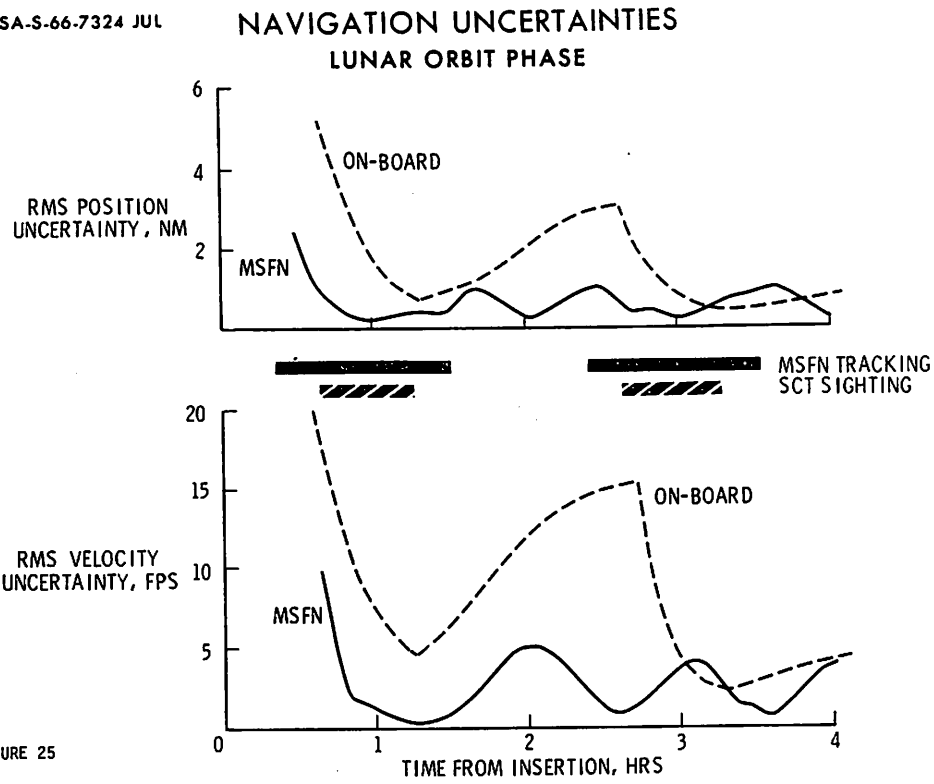


FIGURE 25

NASA-S-65-10034

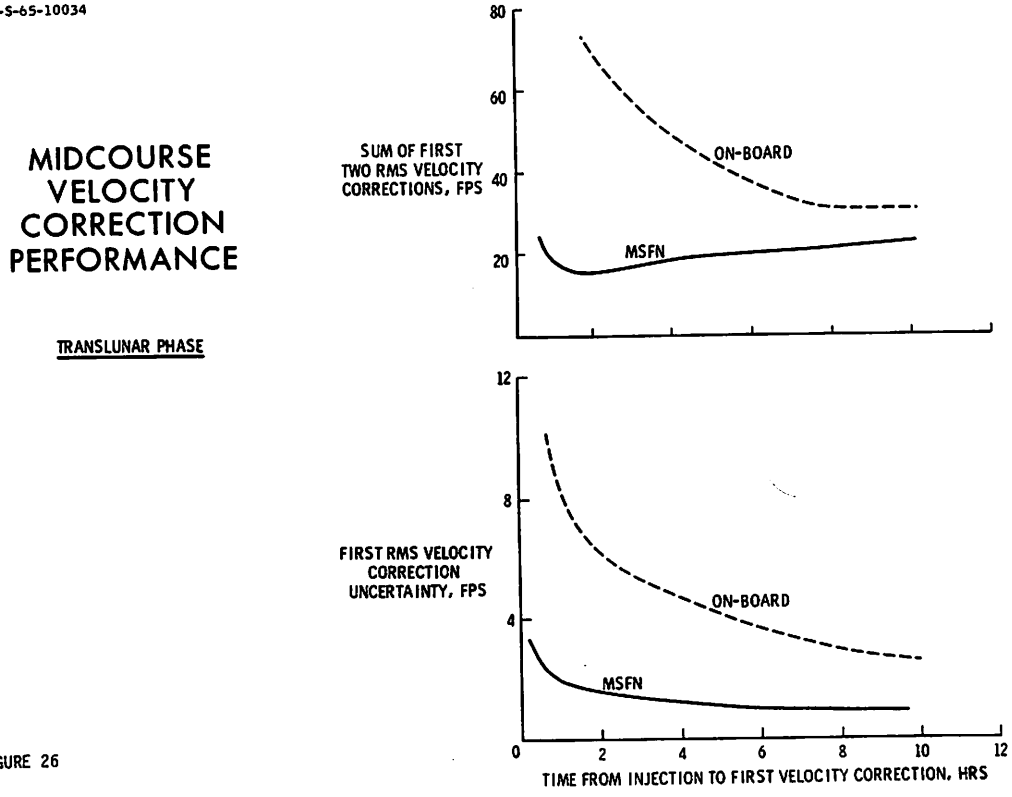
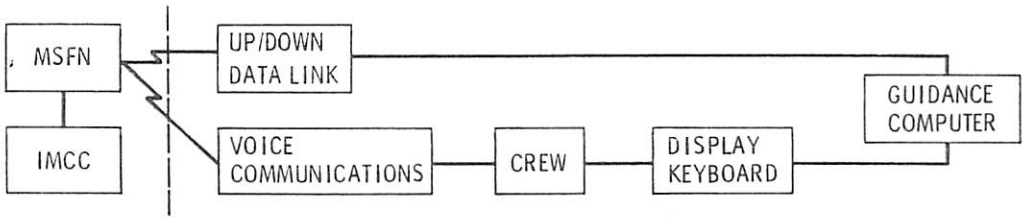


FIGURE 26

DESCRIPTION OF ON-BOARD/MSFN NAVIGATION INTERFACE



- COMPUTER UPDATE RATE - 10 UPLINK WORDS/SEC
- NAVIGATION UPDATE INFORMATION REQUIRED
  - STATE VECTOR - 6 COMPONENTS
  - TIME
  - AIM POINTS
  - COMPUTER TIME ALINE
- UPDATE FREQUENCY REQUIRED - ONCE PRIOR TO EACH GUIDANCE MANEUVER
- TRANSMISSION TIME
  - 60 SECONDS MAXIMUM FOR 99% PROBABILITY OF NO ERRORS
  - NAVIGATION UPDATE NOT TIME CRITICAL

FIGURE 27

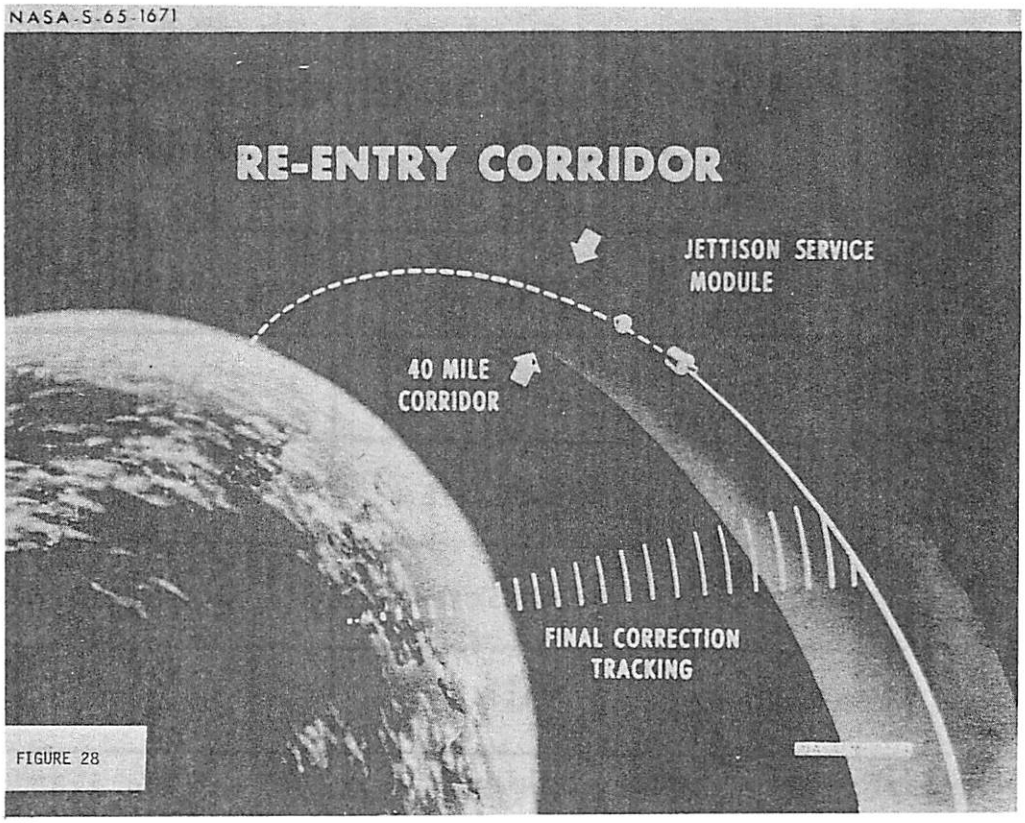


FIGURE 28

NASA-S-65-3769

FACTORS WHICH INFLUENCE DESIGN OF ENTRY GUIDANCE

REQUIREMENTS FOR CREW SAFETY

ENTRY VELOCITY AND ANGLE

PHYSICAL PROPERTIES OF VEHICLE

REQUIRED RANGE CONTROL

PHYSICAL QUANTITIES MEASURED

COMPUTER SIZE AND SPEED

ENTRY MONITORING SYSTEM

FIGURE 29

NASA-S-65-3414

CLASSIFICATION OF AERODYNAMIC PROPERTIES OF ENTRY VEHICLES

L/D	W/C_DS	EXAMPLES
0	CONSTANT	MERCURY
0	VARIABLE	
CONSTANT	CONSTANT	GEMINI, APOLLO
VARIABLE	VARIABLE	

FIGURE 30

NASA-S-66-3574 MAY 12

ERRORS IN LANDING LOCATION OF MANNED GEMINI MISSIONS

MISSION	ASTRONAUTS	LANDING ERROR, N MI
GEMINI II	(UNMANNED)	18
GEMINI III	GRISSOM YOUNG	64
GEMINI IV	MCDIVITT WHITE	47
GEMINI V	COOPER CONRAD	97
GEMINI VI	SCHIRRA STAFFORD	7
GEMINI VII	BORMAN LOVELL	7
GEMINI VIII	ARMSTRONG SCOTT	3

FIGURE 31

NASA-S-65-1414

TYPICAL ENTRY TRAJECTORY

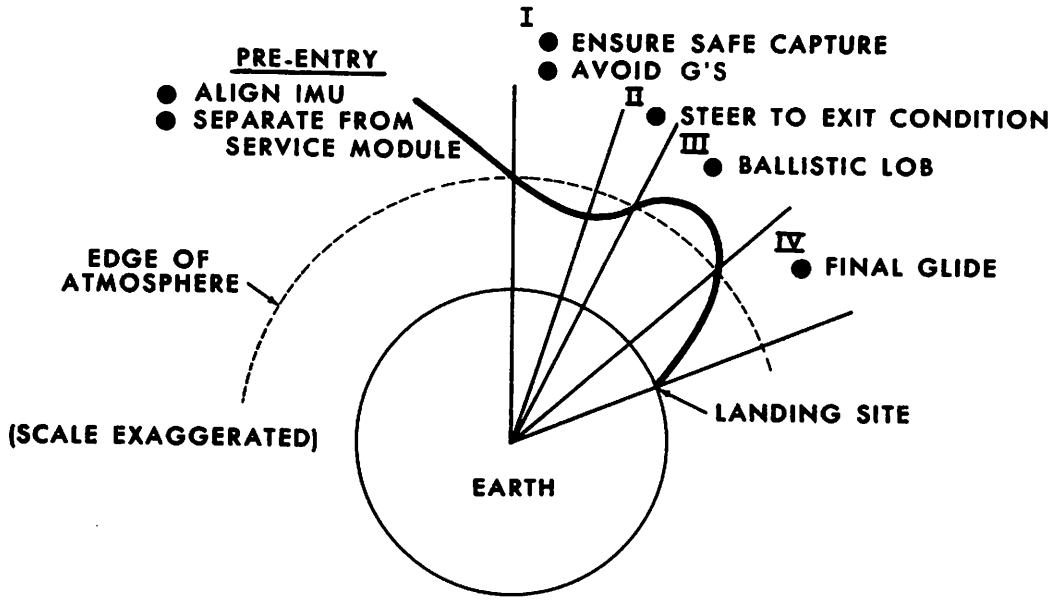


FIGURE 32

NASA-S-65-1413

ENTRY STEERING FLOW CHART

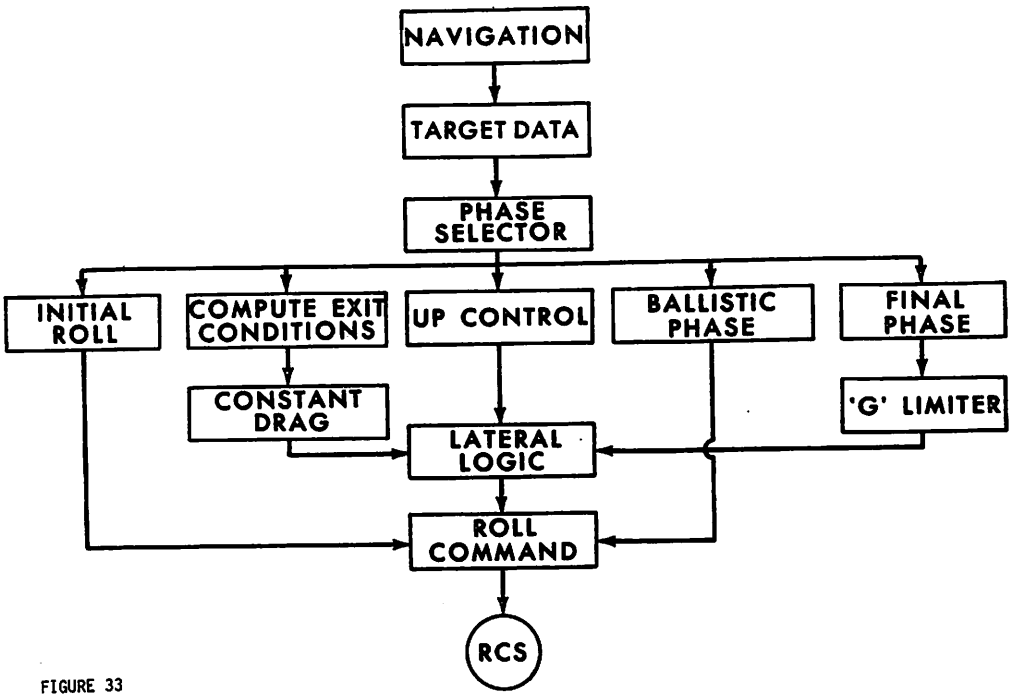


FIGURE 33

NASA-S-65-3768

ENTRY FLIGHT MONITOR

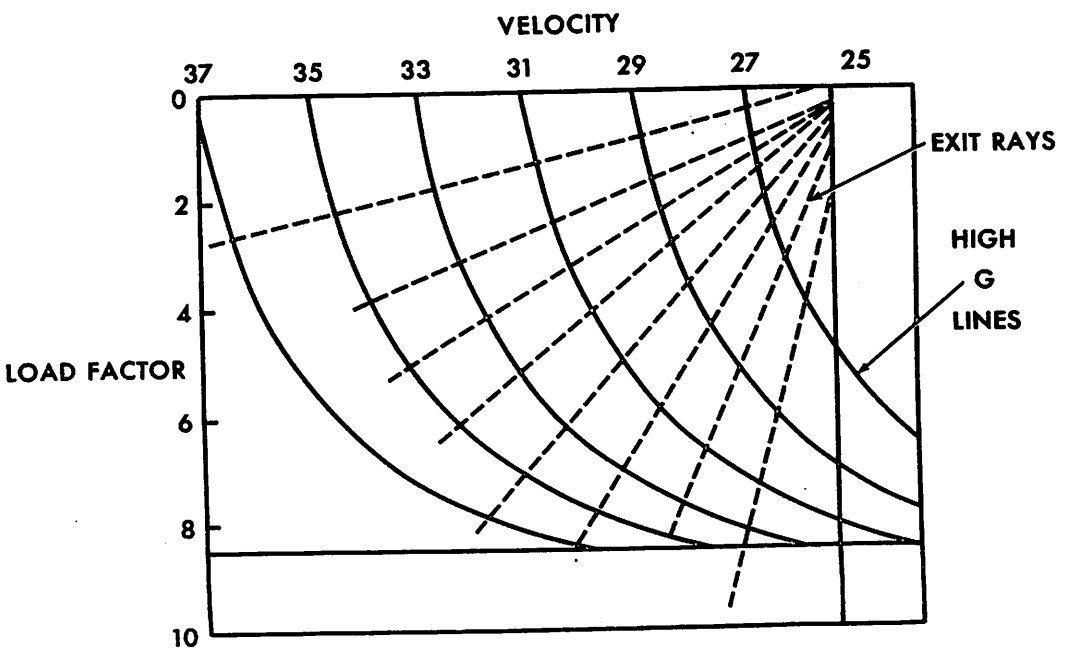


FIGURE 34

NASA-S-65-3793

LEM INTEGRATED GUIDANCE AND CONTROL SYSTEM CONFIGURATION

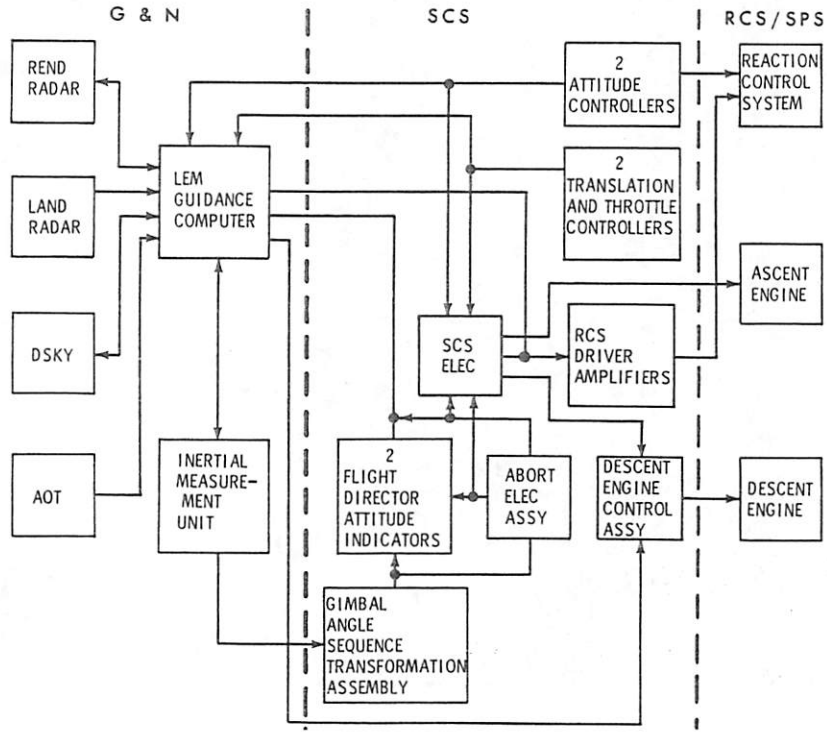


FIGURE 35

NASA-S-65-776

LEM STABILIZATION AND CONTROL SYSTEM BLOCK DIAGRAM

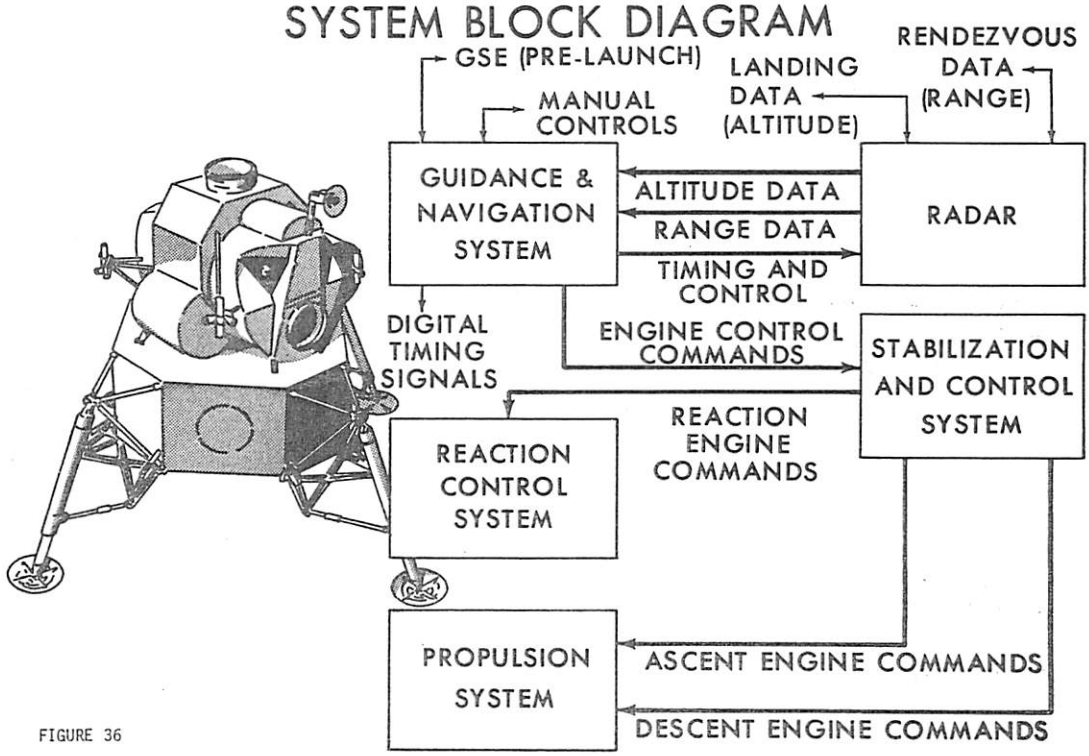


FIGURE 36

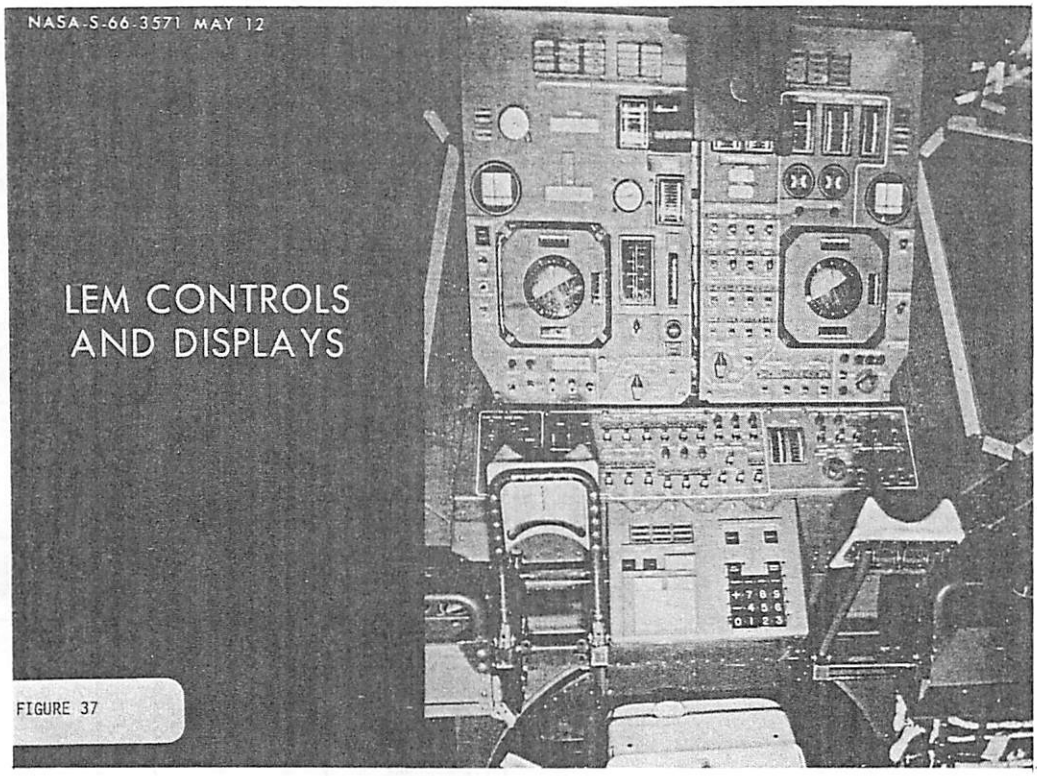


FIGURE 37

NASA-S-65-3982

LEM HOHMANN DESCENT ORBIT

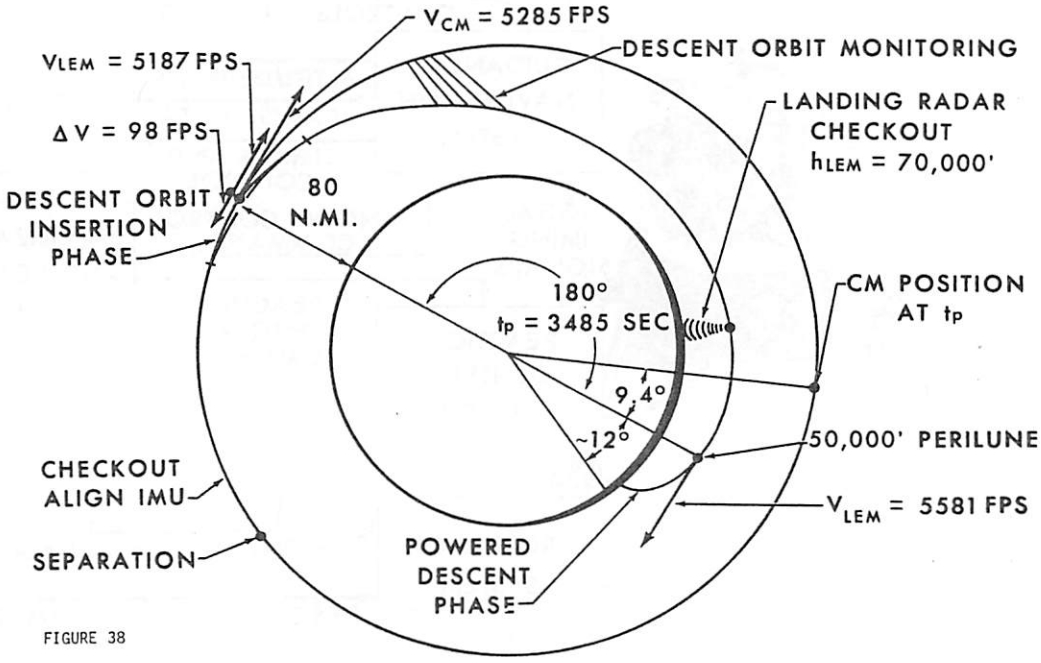


FIGURE 38

NASA-S-66-3573 MAY 12

LUNAR LANDING MANEUVER PHASES

EXAGGERATED SCALE

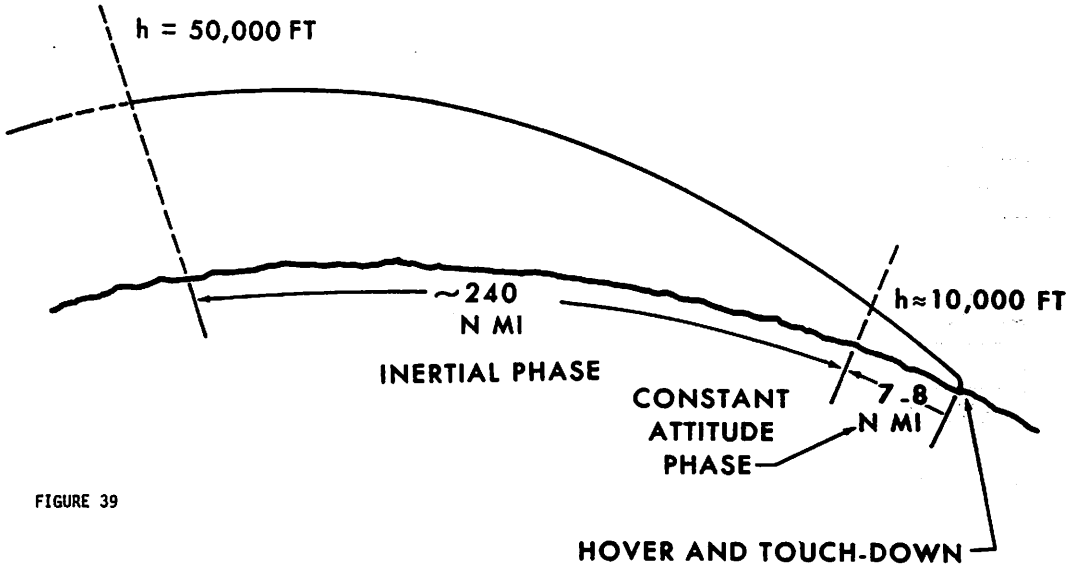


FIGURE 39

NASA-S-66-3575 MAY 12

LUNAR LANDING MANEUVER CONSTANT ATTITUDE PHASE

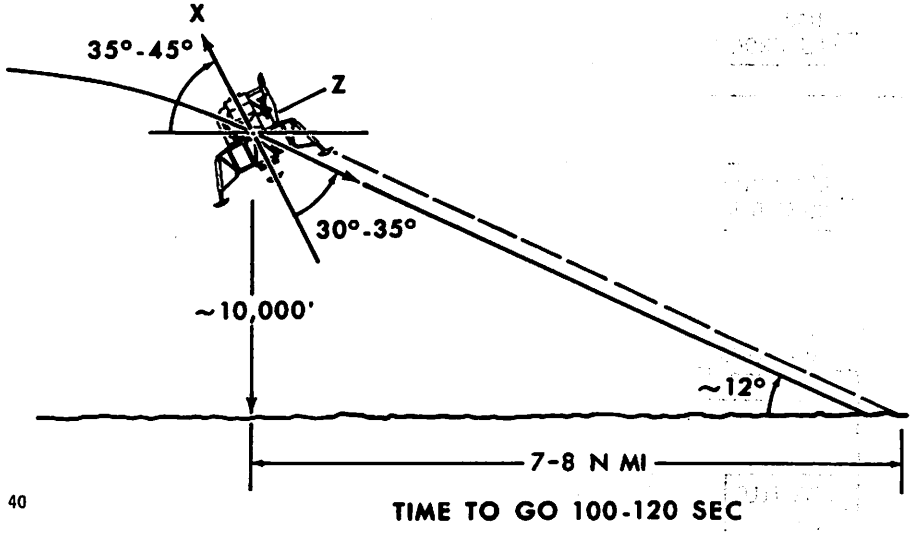
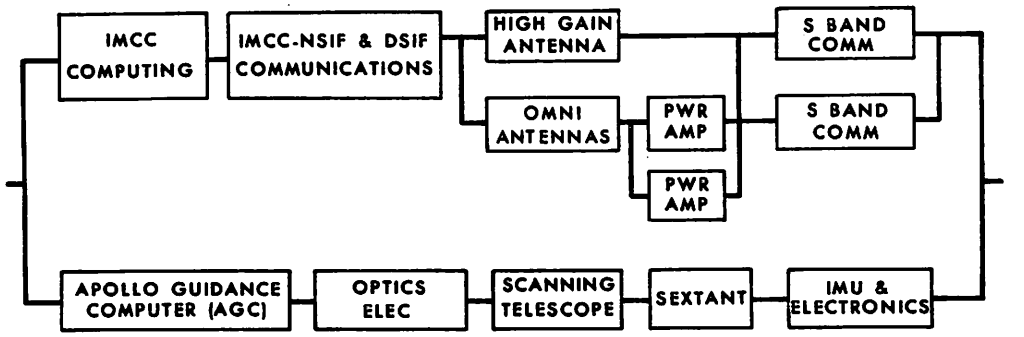


FIGURE 40

NASA-S-65-4482

CSM RELIABILITY
CREW SAFETY - BLK II
NAVIGATION AND GUIDANCE



CONTROL

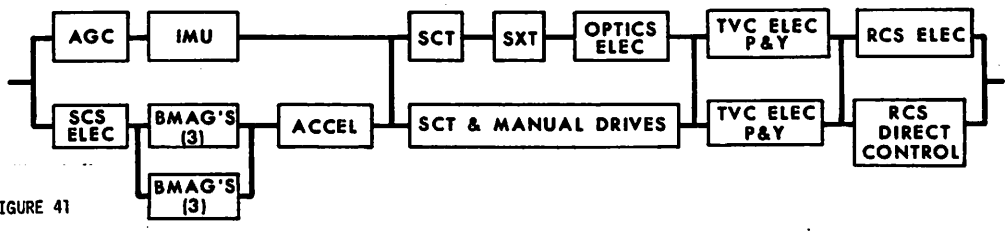
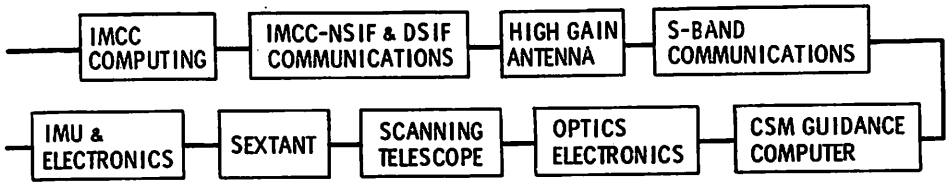


FIGURE 41

NASA-S-65-9999

CSM RELIABILITY
MISSION SUCCESS-BLOCK II

GUIDANCE AND NAVIGATION



CONTROL

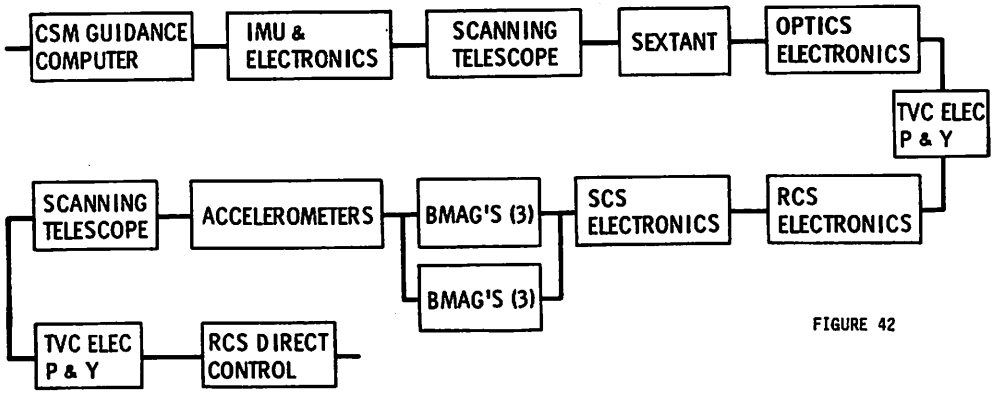


FIGURE 42

DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

by

M. P. Frank

DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

LAUNCH THROUGH EARTH ORBIT INSERTION

The purpose of this paper is to present the various constraints that affect the Lunar Mission planning, primarily in the form of trajectory shaping and the limitation to launch opportunities.

Trajectory geometry constraints and spacecraft performance capability combine to limit the accessible area on the moon. Accessible area limitations combine with operational constraints to limit launch opportunities to certain specifically defined periods. To understand the mission planning considerations and to appreciate the effects of the various constraints, one must closely examine the trajectory characteristics. In this paper an attempt is being made to explain the interrelation of constraints and trajectory shaping. Unfortunately, the explanation becomes quite detailed for some phases of the mission, but this is considered necessary in order to obtain an understanding of the interrelation.

We will begin with the launch and work through the mission, phase by phase. This paper will not describe every trajectory shaping consideration. It will only hit the highlights and discuss the more significant considerations--those that have a major effect on mission planning and the determination of launch windows.

LAUNCH PHASE

The mission planning considerations of the launch phase of the lunar mission are primarily related to launch "windows", booster performance, and contingency planning. Launch windows are defined for two different time intervals. A "daily window" has a duration of a few hours during a given 24-hour period. A "monthly window" consists of a few days during a given month or lunar cycle. The daily window is continuous from opening to closing; but a monthly window may have gaps. For example, a monthly launch window may cover a 7-day period, but a daily window may not exist for some of the intermediate days. A description of the factors that define the launch windows will be presented later in this session. For now, only the effects on the launch phase, of providing a window will be considered.

It is obvious that for operational flexibility it is highly desirable to have as large launch windows as possible, both daily and monthly. A daily window allows delays or holds in

the countdown. The mission would not have to be rescheduled to another day if the window was larger than the cumulative delay or hold times. A monthly launch window allows the mission to be quickly rescheduled. If the daily window were missed, the mission would not necessarily be delayed for a month.

Although the duration and frequency of holds is strongly dependent on the actual vehicle, some estimation of the expected holds can be obtained from historical data. This data shows that for programs employing the Atlas, Titan, and Saturn launch vehicles only in rare cases was a hold of greater than 2 hours followed by a successful launch. This indicates that a 2 hour window should be adequate. If the hold exceeded 2 hours, the mission probably would have to be scrubbed anyway.

For the lunar missions, daily launch windows require changes in launch azimuth; the larger the daily window, the larger is the required azimuth change. The mechanism by which variable azimuth provides launch windows, will be described in the translunar injection phase. For now, let's assume it is required. The limitations to the launch azimuths that can be used, are based on considerations or range safety, booster performance and insertion tracking requirements.

Range Safety--In the early days of the lunar mission planning, the range safety limits were defined as 72° and 108°; however, there is some indication now that these could be increased if necessary. The primary concern of these range safety limits is to keep the space vehicle on the range following any aborts during launch.

Booster Performance--A 90° launch azimuth takes maximum advantage of the earth's rotation in achieving orbital velocity. As the azimuth is shifted away from 90°, the booster performance requirements are increased, or its payload capability is decreased as indicated in Figure 1. The Saturn vehicle is capable of providing lunar mission payloads for launch azimuths beyond the 72° to 108° range; however, its flight performance reserves are drastically reduced. For this reason, the mission will be constrained to these launch azimuth limits.

Insertion Tracking--There is an operational requirement to track the space vehicle from orbit insertion to at least 3 minutes following insertion in order to make a GO/NO-GO decision. Since the Apollo space vehicle is inserted into orbit some 1,400 n. mi. downrange, a ship is required to provide this tracking. The ship speed is relatively slow compared

with the launch azimuth change during a launch window, and it cannot keep up with the changing ground track. Thus the tracking coverage downrange afforded by one ship limits the range of usable launch azimuths to 26° as shown in Figure 2. A 26° range in usable launch azimuths anywhere between the maximum values of 72° to 108° provides at least a 2.5 hour daily window. The choice of where the 26° range is located within the maximum bounds is left up to the mission planner, and is based on such things as maximizing spacecraft fuel reserve, MSFN tracking coverage, launch window duration, and providing a daylight launch.

Another constraint on the launch phase mission planning is the monthly launch window. A monthly launch window allows the mission to be rescheduled as soon as possible in case it is "scrubbed" for any reason for a given day or in case a hold extends beyond the daily window. It also allows some flexibility in the initial planning of the launch day.

How monthly launch windows are obtained and their limitations other than vehicle systems is the subject of the rest of this session. After describing the various constraining considerations during the rest of this talk we will come back to this particular subject. For now, we will discuss only the effects of vehicle recycle characteristics on the requirements and limitations.

The minimum turnaround time, or the space vehicle recycle time, is a major factor in defining the minimum acceptable duration of the monthly launch window. Studies for NASA Headquarters by Bellcomm have provided some significant data that has been used to develop the minimum launch window philosophy. The most important characteristic is the minimum recycle time for the space vehicle, shown in Figure 3.

If the mission were scrubbed after the countdown had reached T - 6 hours, the minimum time to recycle is in excess of 30 hours, and is as long as 40 hours at T = 0. Thus, a minimum window required to guarantee a recycle capability is 3 days. This does not allow any additional time for repairs or replacing components. If this activity could not be done in parallel with the recycling, 3 days would not be sufficient. Therefore, a window of only 3 days duration is not desirable, but is a minimum. In order to allow time for repairs and still make the monthly launch window, it should be as long as possible. The Bellcomm studies indicate that the probability of a

successful launch is between 85% and 90% if a 3-day window is available, and that this increases to about 95% if the launch window is of 5-days duration. Based on this data, the lunar landing mission is being planned only for those periods when at least a 3-day window exists, and every effort is being made to provide 5-day windows.

One final consideration in the launch phase is the desirability of a daylight launch. There are three reasons which make a daylight launch highly desirable--they are all concerned with contingencies.

> Aborts off the pad--The recovery of the crew in the Merritt Island area would be complicated if it had to be performed under conditions of darkness.
>
> Aborts later in the launch require attitude maneuvering of the S/C, and it is desirable to have a sunlit horizon as a backup attitude reference.
>
> Finally, it is desirable to have photographic coverage of the boost phase for post-flight analysis, if a catastrophic failure occurs.

Based on these three considerations, every effort will be made to provide for a daylight launch, although the mission will not be constrained to daylight launches only.

EARTH PARKING ORBIT PHASE

Earth Parking Orbits are required in order to provide launch windows of reasonable duration. Direct lunar injections are possible; however, the launch windows are unacceptably small.

The only major consideration in the earth orbit phase is the duration--or the number of earth orbits. The parking orbit duration is bounded by space vehicle systems considerations. The maximum duration is 4.5 hours from orbit insertion to the beginning of injection and is limited by the launch vehicle capability to provide attitude control and by the battery lifetime. This allows up to three parking orbits prior to the second S-IVB burn. There are other considerations in limiting the parking orbit duration, although they are not "hard" constraints. The S-IVB propellant boil-off and inertail platform drift make it desirable to inject as soon as possible.

The minimum duration of the earth parking orbit phase is limited by the time required to perform system checks and realign the spacecraft platform. Crew timeline analysis indicates that this will require at least 1.5 hours.

There is also a minimum network coverage requirement that states that two tracking stations and a command station must be passed before the GO decision for the second S-IVB burn; however, this is always accomplished in the first orbit. Thus, there is a 3-hour period from 1.5 to 4.5 hours after orbit insertion in which the translunar injection can occur. This means that the injection must occur on the second or third orbit. Figure 4 illustrates the ground tracks for three earth orbits for a typical launch azimuth. The solid line indicates that part of the orbit on which a translunar injection could occur.

TRANSLUNAR INJECTION

The translunar injection position is rigidly constrained by performance considerations. The geometry of the moon's orbit, the energy requirements of the earth to moon transfer trajectory, and the necessity of efficiently burning the S-IVB propellant all combine to place very tight restrictions on the location of this maneuver. An attempt will be made in this section to show how these considerations are interrelated, and how this injection position is determined. It is somewhat involved and will take considerable explanation.

First, let's look at the transfer energy considerations. In order to arrive in the vicinity of the moon, the spacecraft is "aimed" at a position where the moon will be at the time of arrival as illustrated in Figure 5. In order to accomplish this "rendezvous" with a minimum expendature of propellant, the injection must occur very close to the extension of the earth-moon line at the time of arrival. This negative of the unit vector of the moon's position is called the moon's antipode. Something closely akin to a Hohmann transfer is what is being strived for.

This minimum energy transfer would place the perigee on the antipode if the moon's mass did not perturb the trajectory. However, the moon does perturb the trajectory, as shown in the figure, and the perigee must lead the antipode by approximately 8° to compensate. The apogee altitude of the osculating conic trajectory is determined by translunar flight time which defines the trajectory energy required.

To inject to the moon in the most efficient manner, an impulsive velocity would be added along the orbital velocity vector, giving an injection at perigee of the translunar conic. Since an impulsive addition of velocity is not possible, a finite burn time is required, and the actual injection position is on the order of 20° ahead on the antipode. The thrust is directed approximately along the velocity vector, and as the speed increases above orbital, the altitude and flight-path angle increase. For the Apollo configuration, by the time a sufficient energy increase is realized, the altitude increases 60 n. mi. above the orbit and a positive flight-path angle of 6° to 7° has been gained. Since the conic trajectory is very nearly parabolic (eccentricity $\approx .97$), the true anomaly is approximately equal to twice the flight-path angle, so perigee is approximately 12° to 14° behind the burn cut-off position. The burn arc itself is 25°, so that ignition always occurs within a few degrees of the antipode.

The preceeding discussion has shown that the injection position is very closely related to the moon's antipode. To go to the moon efficiently the spacecraft must inject near it, so that we must now address the problem of getting to the antipode from the launch pad at the Cape.

The antipode, being a unit vector from the center of the earth in the direction negative to the moon's position, moves as the moon travels in its orbit. The launch pad is rotating with the earth, and both of these motions must be compensated for in order to rendezvous with the antipode. It is convenient to divide the description of the antipode movements into two categories--a long period cycle and a short period cycle.

The long period cycle is due to the moon's orbital travel about the earth. Figure 6 will be used to illustrate this effect. Assume that the earth is a fixed, motionless sphere and is not rotating about its axis. The moon's orbit plane cuts this sphere as shown. As the moon revolves around the earth, its antipode would trace a great circle in this plane around the surface. Note that the direction of travel is from West to East. The orbital period is some 28 days, and thus at the end of this time the antipode would be back where it began, traveling at the rate of about 0.54° per hour. The latitude of the antipode would have a time history similar to that shown in Figure 6.

The short period motion of the antipode across the surface of the earth is due to the earth's rotation. To illustrate this it is assumed that the moon is fixed at some position in its

orbit, and the earth is now allowed to rotate about its polar axis. The antipode travel is illustrated in figure 7. In this case the latitude is constant and the longitude changes from East to West at 15° per hour. The complete picture of the antipode travel across the earth's surface is obtained by combining the long period and the short period motions. The latitude varies sinusoidally with time with an amplitude of 28.5° (in 1968) and a period of 28 days. The longitude variation is at a nearly linear rate of 14.5° per hour.

The launch must occur at a certain time for each launch azimuth in order to intercept the antipode. This correct launch time is defined by the anitpode's position, the time interval from launch to arrival at this position, and the antipode travel during this time interval. Figure 8 illustrates this problem. Consider an inertial sphere of radius equal to the earth. A trace of the launch pad travel as a function of time on this sphere is represented by a fixed latitude completely encircling the sphere. The launch pad completes one revolution per day. The trace of the antipode is given by the intersection of the moon orbit plane and the sphere. The antipode completes a revolution every 28 days. The launch at any given azimuth must be timed so that the inertial plane of the resulting orbit contains the antipode at the time the space vehicle crosses the moon orbit plane. Later launch times require greater launch azimuths. If additional parking orbits are required, the launch must occur later to account for the additional antipode travel. For each 360° travel of the launch pad, there are two launch times for each azimuth which allow interception of the antipode. This is better illustrated in figure 9, which shows the same situation in earth-fixed coordinates.

In this figure, the launch pad is now fixed, and the antipode travels rapidly over the surface of the earth. The antipode position is shown at four different times during the day, corresponding to the positions at intercept for 72° and 180° launch azimuths. The launch must be timed so that the vehicle intercepts the moving antipode. The time required for the antipode to travel from the interception of the 72° launch azimuth trajectory defines the launch window duration. The intersection of the MOP is drawn in for each position. This figure shows how two different launch times for one azimuth can provide intercept with the antipode. One provides injections going south over the Atlantic Ocean and the other provides injections going north over the Pacific Ocean. For the day illustrated in this figure, the Atlantic injection

gives a trajectory that is nearly in the moon's orbit plane, and the Pacific injection results in a trajectory that is highly inclined to the moon's orbit plane. Half a lunar cycle later, the Pacific injection would be highly inclined. The magnitude of this relative inclination depends on the lunar declination and is a maximum when the moon is near the equator. When the moon is near maximum declination, both windows provide trajectories with low relative inclinations.

It is of special significance that the Pacific injection always results in a trajectory above the moon orbit plane, regardless of the moon's declination or whether it is ascending or descending. This effects the relative location of available landing areas on the moon from these two injection windows. This effect will be described later.

It can be seen that when the launch azimuth bounds are defined, the proper launch time can be found, allowing for the number of parking orbits to be employed prior to injection.

If for some reason the injection opportunity were missed, it could be attempted one orbit later when the space vehicle again approached the antipode. However, since the antipode is traveling in a plane that is not necessarily the same as the vehicle's orbit, a plane change would be required. This is illustrated in figure 10. It can be seen that the antipode has traveled out of the parking orbit plane when the vehicle returns to the position of injection. The magnitude of the out-of-plane travel is dependent on the relative inclination between the parking orbit and the moon orbit planes. The maximum value is about $.6^{\circ}$. This second injection would require a greater propellant expenditure by the S-IVB because of the plane change involved. If two injection opportunities are to be provided, the launch would be timed so that both would require a plane change, because this minimized the propellant required. The launch would occur a little bit later so that the first time the vehicle crosses the moon orbit plane, the antipode has not reached the parking orbit plane. The second time the vehicle crosses the moon orbit plane, the antipode has passed through the parking orbit plane. If three injection opportunities are to be provided, the launch would be timed so that the antipode was in the parking orbit plane for the second one.

Figure 11 illustrates the effects of different targeting methods on the characteristic velocity required to provide additional injection opportunities. Three cases are shown. The first shows additional ΔV required when the launch is timed for the first injection to be coplaner. The second and third opportunities have large additional ΔV requirements.

In the second case, the launch is timed to split the delta azimuth between the first and second injection opportunities. This would be used for two injection opportunities.

In the third case, launch is timed so that second injection opportunity is coplaner. This method would be to provide three injection opportunities.

The penalties shown are only illustrative; the actual values strongly depend on relative inclination between the two planes.

Because injection is limited to the second or third orbit, only two injection opportunities are planned for 504 mission, and the second targeting technique is being used.

The combination of launch azimuth limits, parking orbit duration constraints, and the geometry of the moon's orbit confine the location of the injection positions to two geographical areas. These areas are generally centered over the South Atlantic Ocean and the Pacific Ocean, and for this reason are distinguished by these names. The bounds, as shown as figure 12, are defined by the first orbit for a 72° launch azimuth, the third orbit for a 108° launch azimuth, and the extremes of lunar declination. The areas shown in figure 12 contain all of the possible injection positions.

TRANSLUNAR COAST PHASE

In this, the description will be confined to: the effects the trajectory inclination relative to the moon's orbit plane; the effects of the "free-return" flight plan, and its relationship to the lunar orbit insertion maneuver; and finally a discussion of some alternatives to the free-return flight plan.

The first point to be made in describing the translunar coast trajectory is in regard to the relative location of the trajectory to the moon's orbit plane. It was stated earlier that Pacific injections always result in translunar trajectories from Atlantic injections are always below it as shown in figure 13. The amount of out-of-planeness is a function of the moon's declination and whether or not it is ascending or descending in its orbit. These parameters influence the magnitude of the effects, however they do not change the general conclusions.

Following a Pacific injection, the spacecraft approaches the moon from above the moon orbit plane. This forces the trajectory below this plane on the far side of the moon, where the lunar orbit insertion maneuver takes place. The resulting lunar orbit then is constrained to be approximately as illustrated

in figure 14. A plane change during the orbit insertion can modify the resultant orientation somewhat, but the basic conclusion can still be drawn that to land at northern latitudes on the front side of the moon, a Pacific injection will result in lower propellant costs. Conversely, Atlantic injections favor the southern latitudes. This will be clearly demonstrated later when the accessible lunar areas are defined.

FREE RETURN

One of the most constraining requirements of the lunar landing mission is the free-return trajectory. It incurs sever limitations to the area on the moon to which Apollo missions can be conducted. Although it is costly in terms of spacecraft performance requirements, the inherent safety feature of a free return makes it a highly desirable method of getting to the moon.

A circumlunar free-return trajectory, by definition, is one which circumnavigates the moon and returns to earth as shown in figure 15. The perigee altitude of the return trajectory is of such a magnitude that by using negative lift the reentering vehicle can be prevented from skipping out of the atmosphere, and the aerodynamic decelleration can be kept below 10 g's. Thus, with a complete propulsion system failure following the translunar injection, the spacecraft would return safely to earth.

The range in return perigee altitudes that provide this feature is called the reentry corridor and is primarily a function of the lift-to-drag ratio of the reentry vehicle. For the Apollo vehicle this corridor is approximately $\pm$ 12 n. miles centered around a 25 n. mile altitude. The injection velocity accuracy required to achieve a free-return trajectory is less than a tenth of a foot per second. Obviously, this is well beyond the capability of any guidance system when the total ΔV involved is on the order of 10,000 fps. However, it is still valid to plan for a free-return trajectory, because this procedure at least will minimize the ΔV requirements to return to earth should an SPS failure occur. In this situation, there is a good probability that the RCS can provide the necessary velocity corrections to overcome the injection errors.

The free-return trajectory severely limits the accessible area on the moon because of the very small variation in allowable lunar approach conditions and because the energy of the lunar approach trajectory is relatively high. The high approach energy causes the orbit insertion ΔV to be relatively high. However, the main limitations to accessible area are a result

of the small range in flight times from earth to moon. Figure 16 illustrates the effect of flight time on the location of perilune. All free-return trajectories have translunar transit between 60 and 80 hours, and it can be seen in figure 16 that perilune is limited to a region within about 10° of the negative of the earth-moon line or approximately 180° longitude. For non-free-return trajectories, the transit time can be anything from 50 hours to 110 hours.

Perilune could be adjusted from 140° W longitude to 140° E longitude merely be selecting the appropriate flight time. This narrow region of perilune position of free-return trajectories combined with a small range of approach inclinations is what limits the accessible area.

The relative inclination between the free-return trajectory and the moon orbit plane is less than 11°. Any trajectory with a greater inclination than this, simply does not return to the entry corridor at earth, regardless of the perilune position. The range of free-return trajectory conditions near the moon is illustrated in figure 17. Note the relatively small cone formed by the locus of perilune positions.

The braking maneuver to decellerate the spacecraft from the hyperbolic approach trajectory to a lunar orbit is performed at or near perilune. For illustrative purposes, it will be assumed that it occurs at perilune. In order to land at a site that is not contained by the approach trajectory plane, a plane change must be made. It is generally more efficient to combine this plane change with the decelleration at orbit insertion. When the landing site is near the mode, however, an excessively large plane change is required to cause the trajectory to pass over the site. This is illustrated in figure 18. Since the approach trajectories have low inclinations and orbit insertion occurs near the 180° longitude, it can be seen that to cause the lunar orbit to pass over sites at high latitudes in the region near 0° longitude large plane changes would be required. The propellant capacity of the spacecraft limits the magnitude of the plane change that can be made.

As was noted in the preceeding figure, there is a locus of perilune positions; it is not that there is not one focal point through which all of the trajectories must pass; there is an area. This tends to relieve the limitations slightly, but the fact remains that a plane change at deboost is relatively ineffective in achieving higher latitudes near the zero longitude. Note also that as the landing site is moved away from the 0° longitude, the plane change requirements become much less.

So far, we have only described the lighting effects on lunar landing site longitudinal location and the effect on launch dates. The latitudinal location of the landing site also has an effect on available launch dates because of its effects on Service Module performance requirements. The lighting may be acceptable, but if the landing site is outside the latitude bounds of spacecraft performance, then a launch opportunity still does not exist. The latitude limits which can be attained are defined by SPS propellant available are a function of selenographic longitude, lunar declination and librations, and the translunar injection window.

The accessible area for a typical day is illustrated in figure 28. Note that the area available from the Pacific injection window is somewhat north of that available from the Atlantic window. These areas shift to the south as the moon travels to northern declinations and vice versa. In order for an acceptable launch window to exist on any given day, the landing site must be within the area defined by these latitude bounds and within the longitude region defined by the lighting bounds. The latitudinal shift in the accessible area boundaries is cyclic, with a period approximately equal to the lunar orbit period.

If a region on the lunar surface lies in the accessible area throughout the month, regardless of the daily shifting of these areas, it is said to be 100% accessible. A 100% accessible area has great significance in the selection of lunar areas to be examined for possible landing sites. If the candidate sites can be located in a region that is always accessible, then the mission planner is releived of one very troublesome constraint, namely spacecraft performance. It can be guaranteed that no mater what the lunar declination or libration, when the lighting is acceptable, the landing site is attainable.

The area available every day of the month presents a pessimistic picture, in that it does not consider the fact that only about 8 days are really usable because of the lighting constraint. A more realistic picture of the area available for a month, would be obtained if the latitude limits were defined for the longitudinal regions on the days when the lighting was acceptable in those regions. That is, select only those days when the sun elevation is between 7° and 20° for the longitudes between 45° E and 45° W; and furthermore on any one of these days define the latitude limits only for the longitude region which had acceptable lighting. The area available during the month of February 1968 under these conditions is illustrated in figure 29. The available area defined in figure 29 cannot

be extrapolated from month to month because the lighting cycle and the declination and libration cycles do not have the same period. So that one of these figures must be made for every month of interest.

The accessible areas for each month during a year can be combined to define an area available for the entire year. This is illustrated in figure 30.

The purpose of all of this discussion of performance limitations has been to show that the landing site location has a major effect on launch opportunities; not only through lighting conditions and longitude interactions but also through latitude and performance interactions. This total interaction can be summarized as follows: Given a landing site location, a launch is possible only on the day that the lighting is acceptable and then only if the landing site is within the latitude bounds attainable for that longitude on that day.

If lunar landing sites could be selected entirely on the basis of performance and lighting constraints, they could be located so that there would be no restrictions on launch windows. Unfortunately, there are many factors that must be considered in the selection of lunar landing sites. These other factors force compromises to be made, with the result being that launch windows are in fact constrained by the available lunar landing sites. A complete description of the lunar landing site criteria, other than the launch window considerations will be presented in a later session.

We have seen that lunar landing sites, lighting requirements, and spacecraft performance are all very effective in constraining the launch opportunities. Just how constraining, will be illustrated shortly. In order to illustrate the limitations on launch opportunities, it will be necessary to assume certain lunar landing sites. For the purposes of this illustration, we will assume that the sites to be photographed on the lunar Orbiter A and B missions are found to be acceptable for Apollo landings.

First, let's consider the seven sites to be photographed by Orbiter A. The launch opportunities provided by these 7 sites throughout the year 1968 are summarized in figure 31. The interesting features of this figure are the pattern and frequency of launch opportunities, and the frequency of night launches.

During some of the months, many launch opportunities are not shown for the Pacific injection window. This is because some or all of the sites were outside of the performance boundaries and were not available when the lighting was acceptable. The Pacific injections are the hardest hit by this performance limitation because the western sights to be photographed by Orbiter A are generally south of the equator, away from the best performance region of the Pacific windows. In the early part of the year, the moon's declination and lighting are in a favorable phase, and the southern sites are well within the performance capability of the Pacific injection. However in the latter part of the year, the moon is at a northern declination when the lighting is acceptable and the southern sites are outside of the accessible area.

Since the best performance region for the Atlantic injection window is south of that for the Pacific, the lighting and libration combination is favorable for this window in the latter part of the year. The result is that for those months when the Pacific injections are unavailable, the Atlantic injections are. So launch windows exist all year.

The effect of constraining the mission to be launched only in daylight can also be determined from figure 31. Although this is not considered to be a firm constraint for all lunar missions, it is highly desirable and every effort will be made to have a daylight launch at least for the first one.

From figure 31, it can be seen that to limit the launch to daylight hours eliminated the Atlantic injection window for the entire year. In addition, several of the launches using the Pacific injection window occur at night during the winter months and would also be lost.

The net result of the combination of performance limitations and a daylight launch constraint would be to virtually eliminate all launch windows in the last quarter of 1968.

The effect of a minimum launch window duration constraint is demonstrated in figure 32. This figure shows the remaining launch opportunities if only those launch windows of 5-days duration were considered. It is clear that the launch opportunities afforded by the seven sites used in this analysis are entirely satisfactory during the first half of 1968. And that the situation rapidly changes from marginal to unsatisfactory during the third quarter and remains that way for the rest of the year because night launches would be required. Five-day daylight windows are available in March, April, May, and early June. However, beginning in late June and continuing throughout the rest of year, a night launch would be required.

To improve the daylight launch capability, the sites to be photographed on the Orbiter B mission were located north of the A sites. This provided sites in a area more favorable from the Pacific injection window. The resulting 5-day launch windows are summarized in figure 33. It can be seen from this figure that 5-day windows with a daylight launch could be obtained throughout the year if the Orbiter B sites were available.

Questions and Answers

LAUNCH THROUGH LUNAR ORBIT

Speaker: M. P. Frank

1. Does the launch azimuth change through the launch window?

   ANSWER - Yes; however, the launch azimuth is changed in discrete steps.

2. Dr. Reiffel - At sufficiently high altitudes both the solar cycle and individual solar effects can markedly affect the scale heights and density. In view of the mechanization of the re-entry monitor being planned which senses, as I understand it, very low G forces and rates of change to decide whether the spacecraft is in the corridor - what would be the effects of a flare heating the high atmosphere (and perhaps also requiring a radiation induced abort) and what data are really available on atmospheric structure for the proper time in the solar cycle on which to base the corridor design? Is significant solar-induced dispersion possible in other words?

   ANSWER - MSC will investigate the effects of solar flares on the entry conditions.

NASA-S-66-6460 JUN

EFFECT OF LAUNCH AZIMUTH ON PAYLOAD CAPABILITY

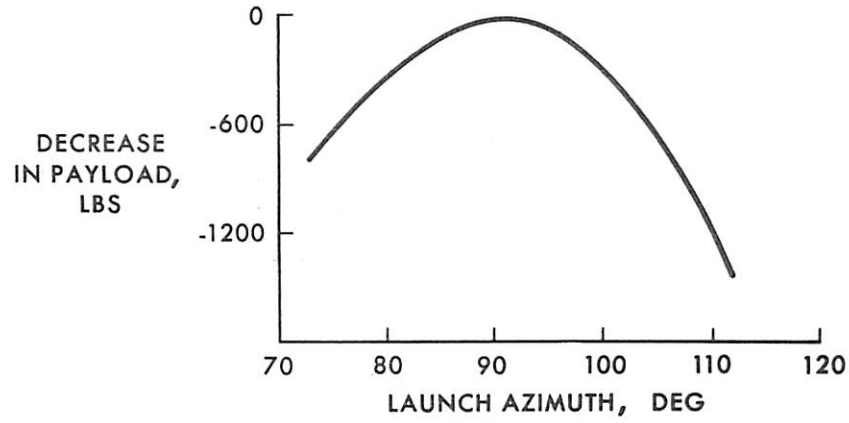


Figure 1

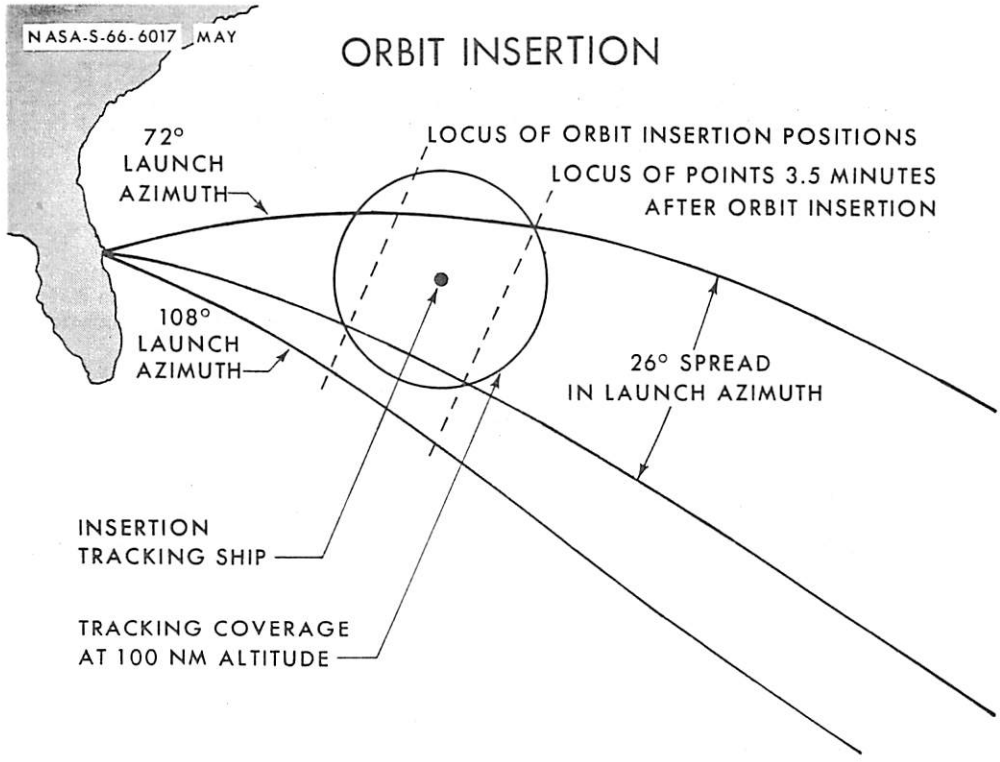


Figure 2

NASA-S-66-6003 MAY

LAUNCH ABILITY BY HOLD OR RECYCLE

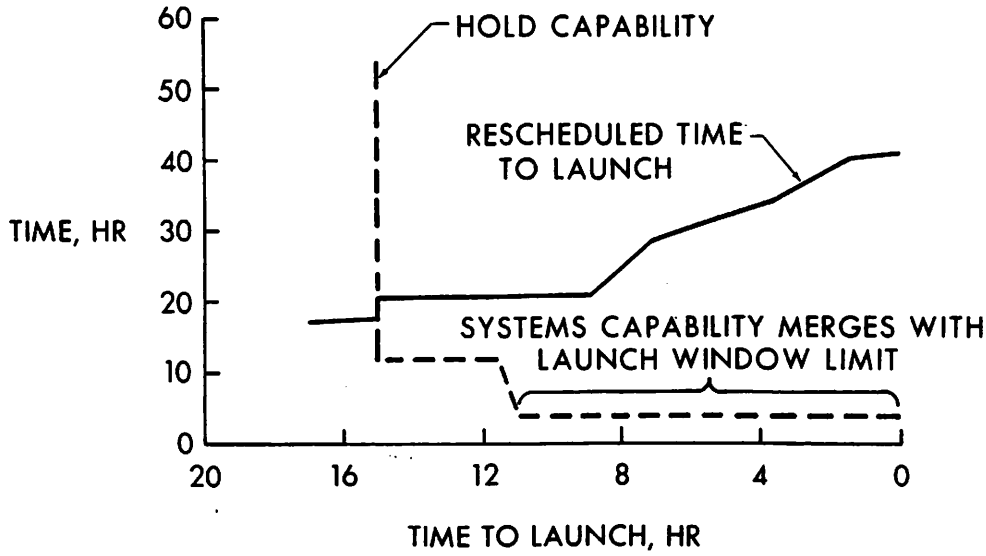


Figure 3

NASA-S-66-6073 MAY

EARTH PARKING ORBIT GROUND TRACK FOR THREE ORBITS FOR A 90 DEGREE LAUNCH AZIMUTH

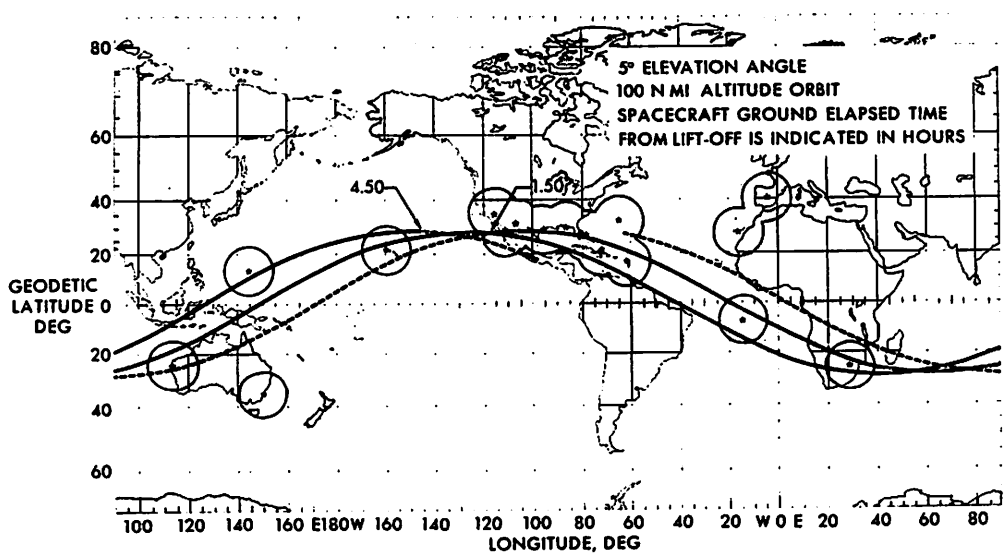


Figure 4

NASA-S-66-6011 MAY

TRANSLUNAR INJECTION GEOMETRY

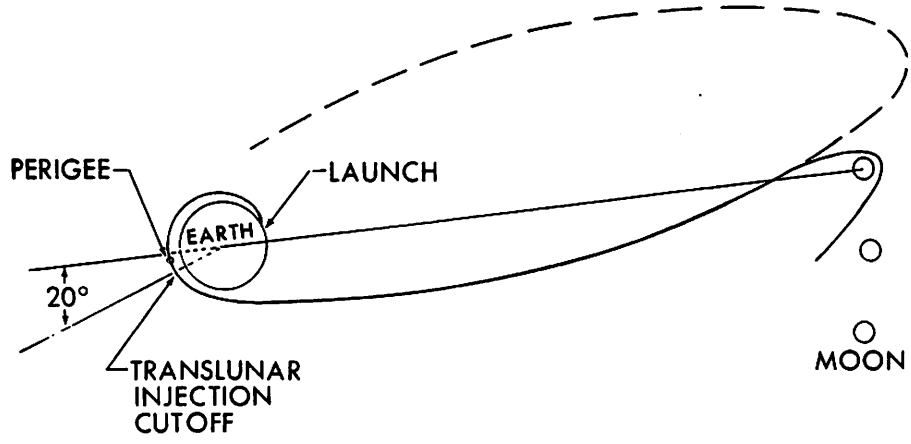


Figure 5

NASA-S-66-6015 MAY

LONG PERIOD MOTION OF ANTIPODE

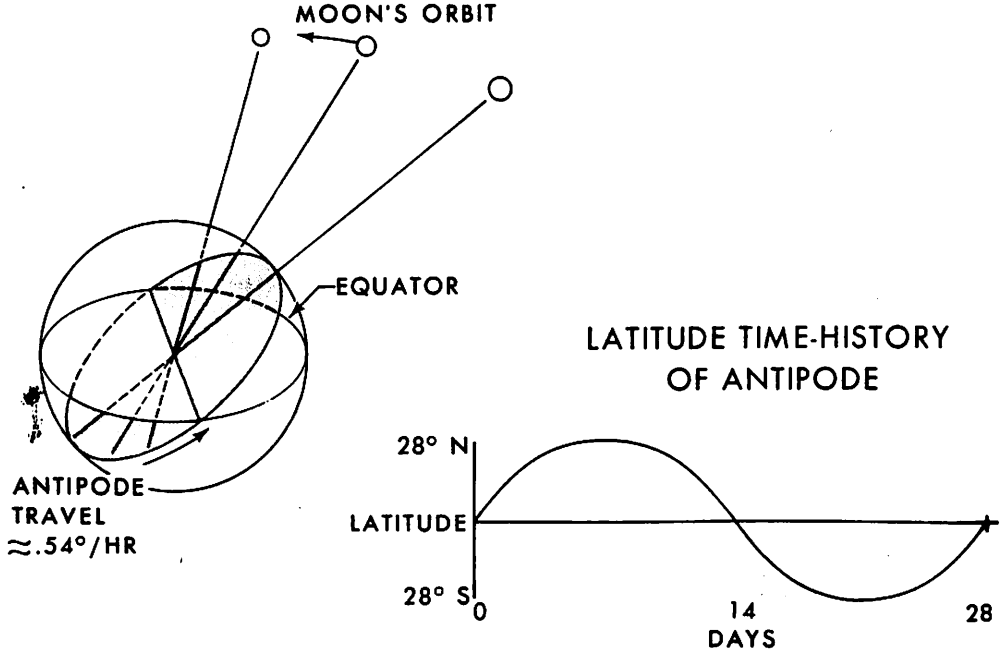


Figure 6

NASA-S-66-6023 MAY

SHORT-PERIOD MOTION OF ANTIPODE

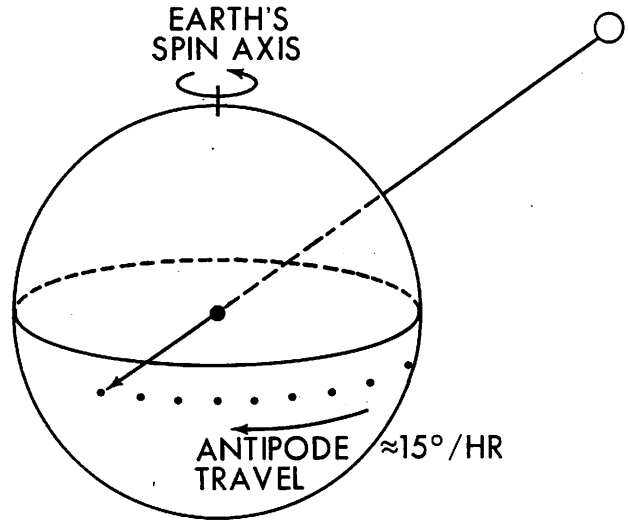
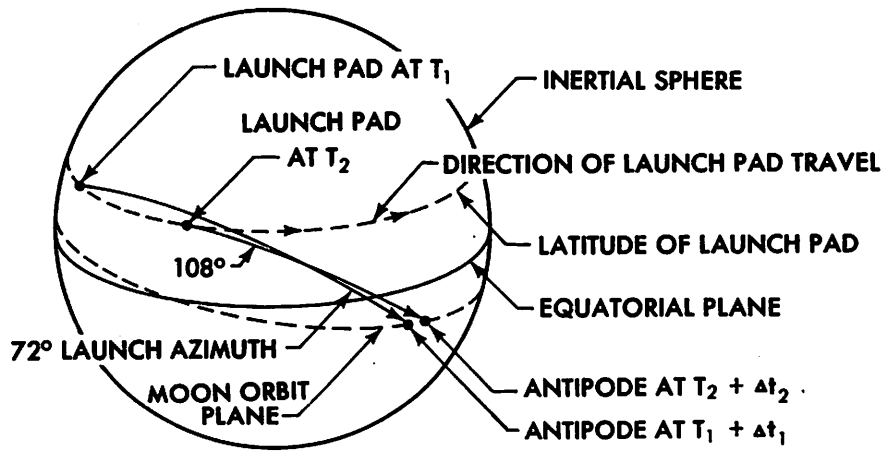


Figure 7

NASA-S-66-6419 JUN

ANTIPODE RENDEZVOUS GEOMETRY



Δt_1 = TIME FROM LAUNCH TO FIRST CROSSING OF MOP FOR 72° LAUNCH AZIMUTH

Δt_2 = TIME FROM LAUNCH TO FIRST CROSSING OF MOP FOR 108° LAUNCH AZIMUTH

Figure 8

NASA-S-66-6016 MAY

TRANSLUNAR INJECTION CHARACTERISTICS

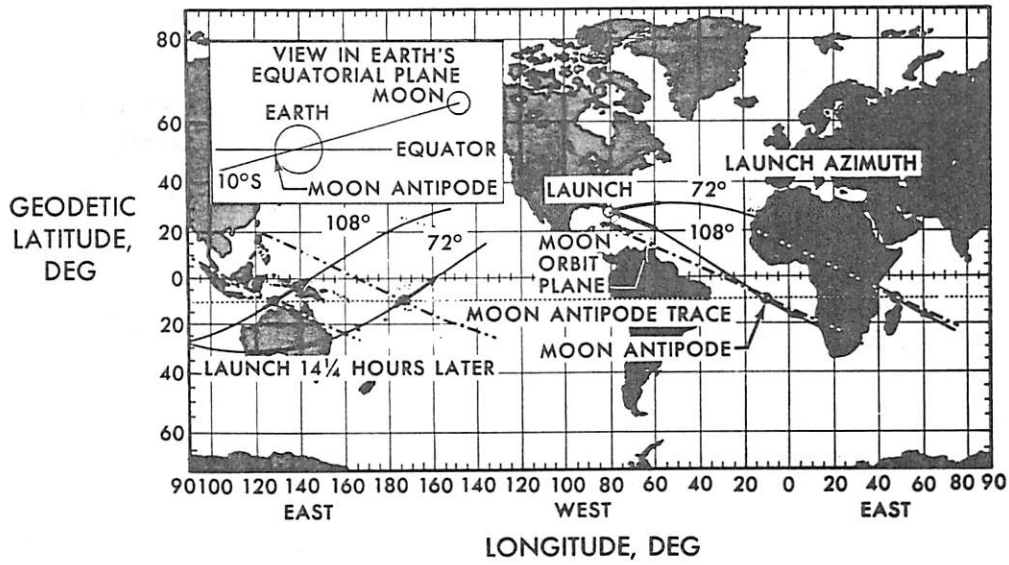


Figure 9

NASA-S-66-6012 MAY

EFFECT OF MISSING 1ST INJECTION OPPORTUNITY

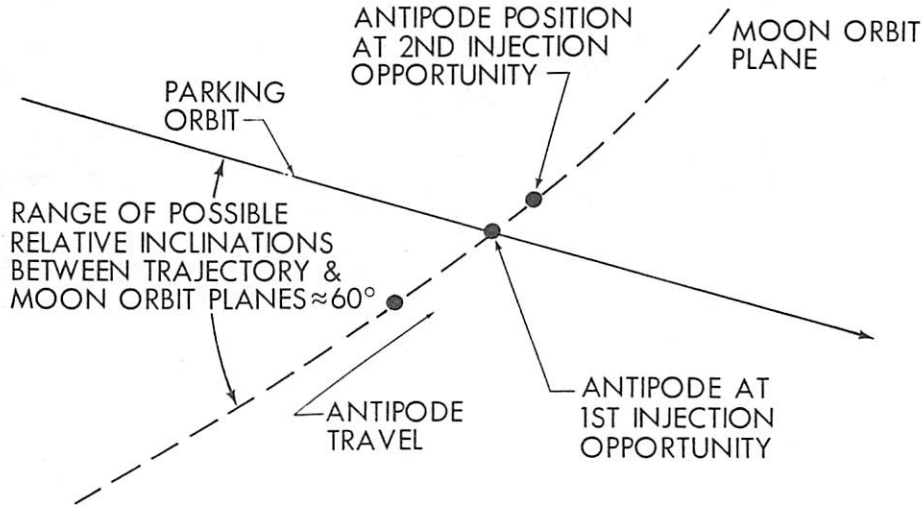


Figure 10

NASA-S-66-6013 MAY

INJECTION OPPORTUNITY

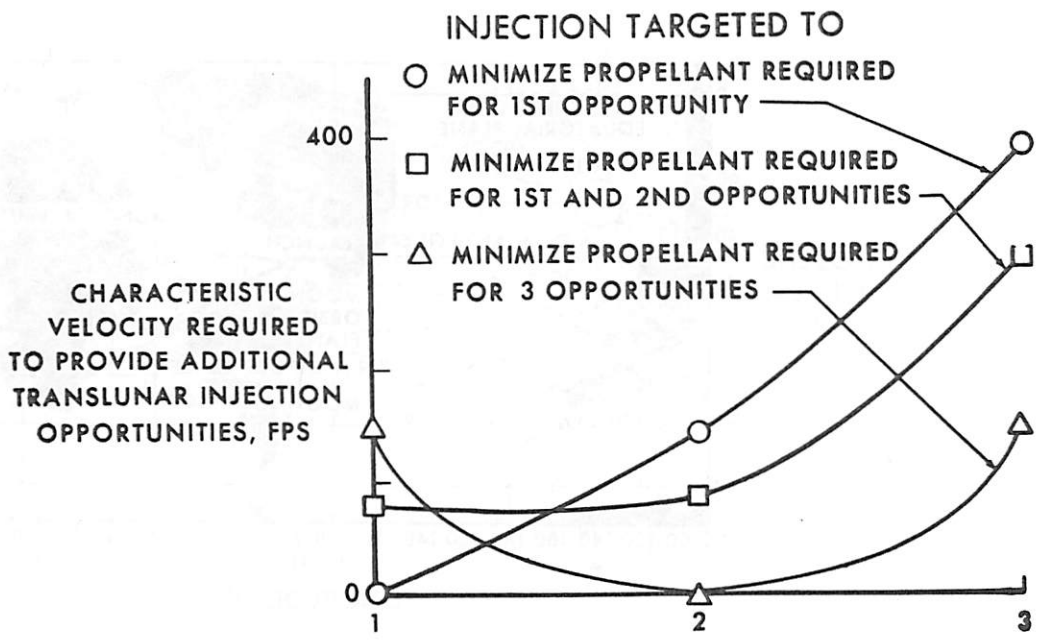


Figure 11

NASA-S-66-6019 MAY

ENVELOPE OF FREE RETURN TRANSLUNAR INJECTION CUTOFF POSITIONS FOR 1968 AND 1969

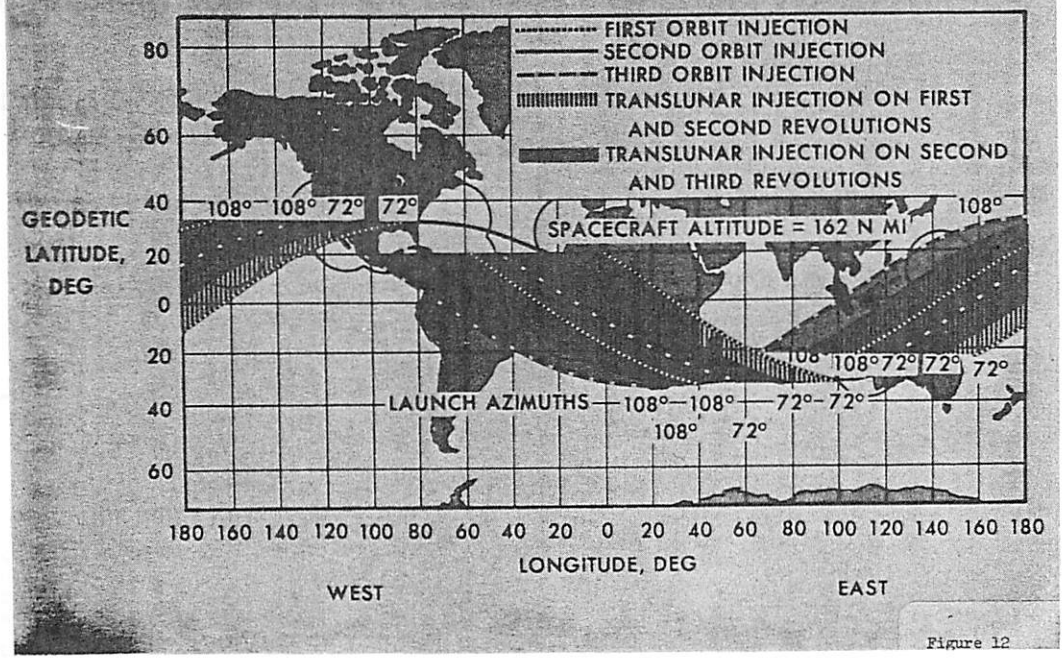


Figure 12

NASA-S-66-5022 MAY 31

EFFECT OF INJECTION WINDOW ON TRANSLUNAR TRAJECTORY GEOMETRY

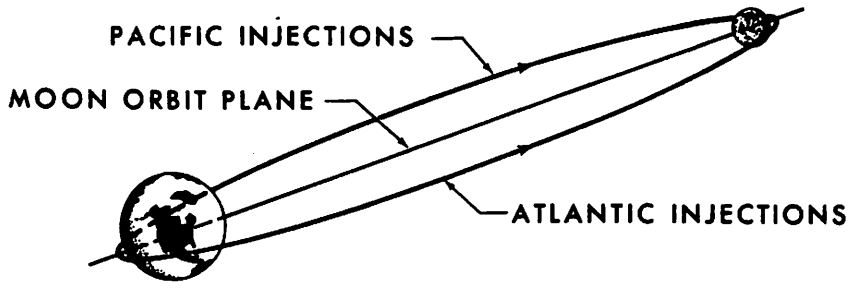


Figure 13

NASA-S-66-6390 JUN

EFFECT OF INJECTION WINDOW ON LUNAR APPROACH CONDITIONS

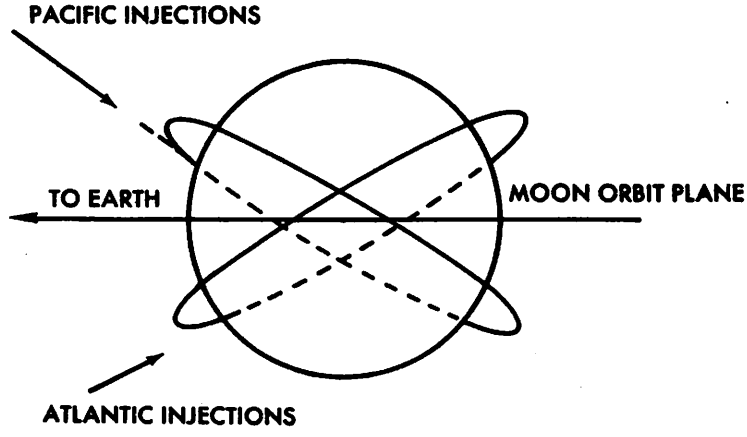


Figure 14

NASA-S-66-6388 JUN

FREE RETURN TRAJECTORY

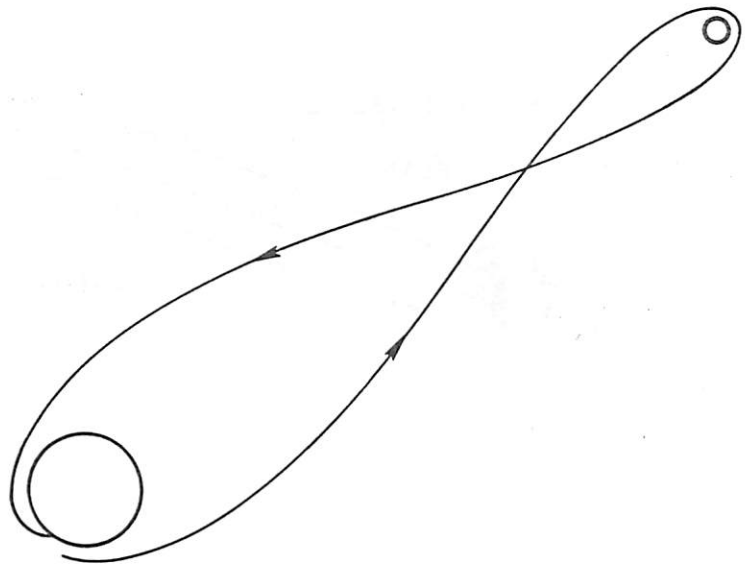


Figure 15

NASA-S-66-6007 MAY

PERILUNE POSITION FOR DIFFERENT TRANSLUNAR TRANSIT TIME

(ALTITUDE OF PERILUNE ≈ 80 NM)

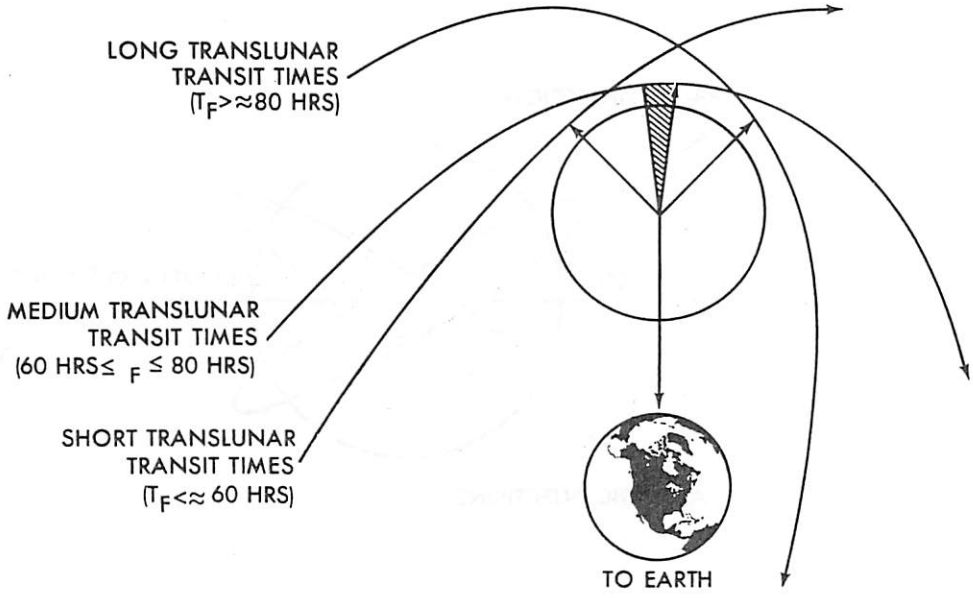


Figure 16

NASA-S-66-5010 MAY 31

LUNAR APPROACH GEOMETRY OF FREE-RETURN TRAJECTORIES

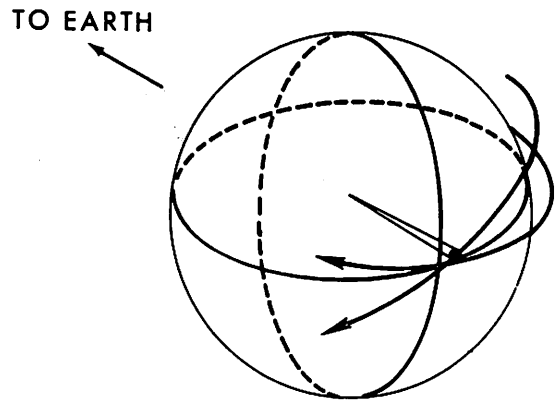


Figure 17

NASA-S-66-5015 MAY 31

EFFECTIVENESS OF PLANE CHANGE AT LUNAR ORBIT INSERTION ON INCREASING ACCESSIBLE LANDING SITES

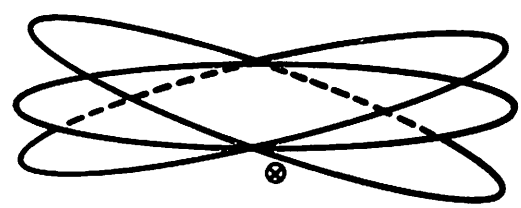


Figure 18

NASA-S-66-5014 MAY 31

RELATIVE TRAJECTORY GEOMETRY OF LUNAR ORBIT INSERTION MANEUVER

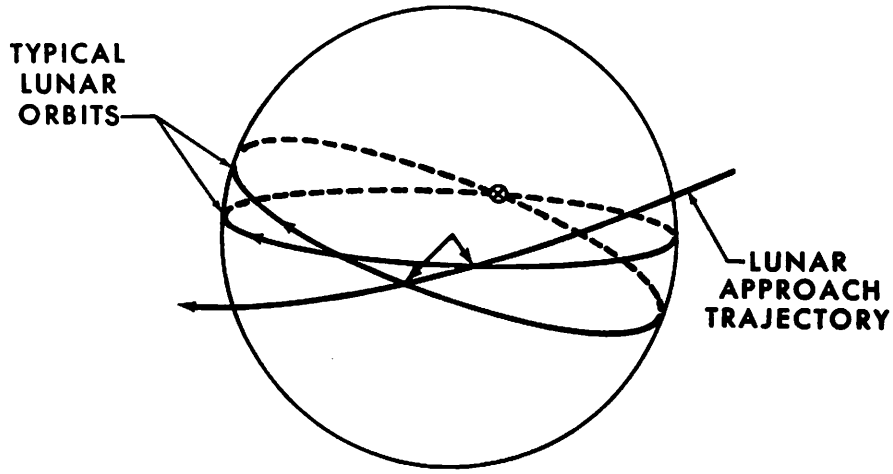


Figure 19

NASA-S-66-6393 JUN

GEOMETRY OF LUNAR ORBIT INSERTION

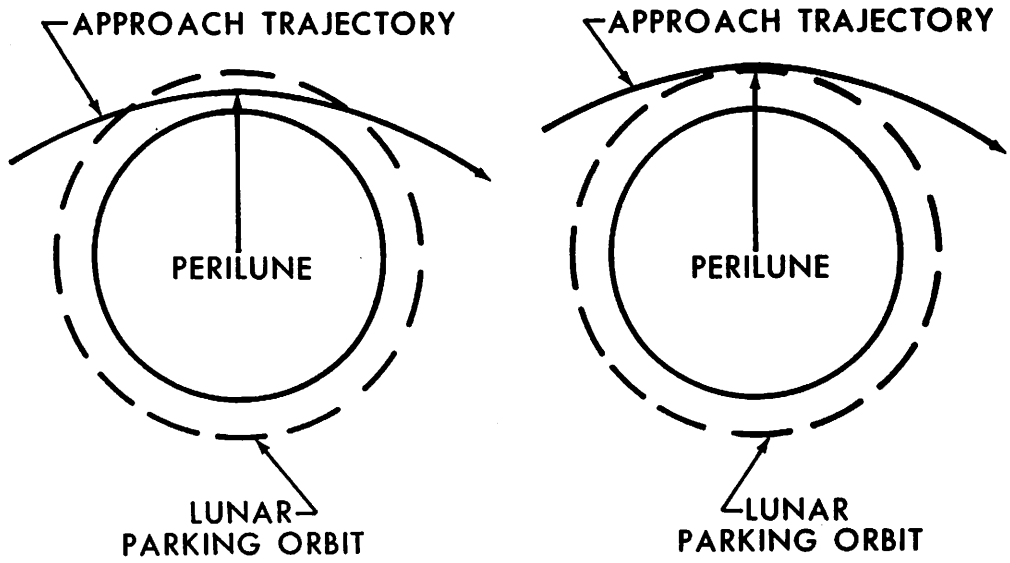


Figure 20

NASA-S-66-6022 MAY

HYBRID FLIGHT PLAN

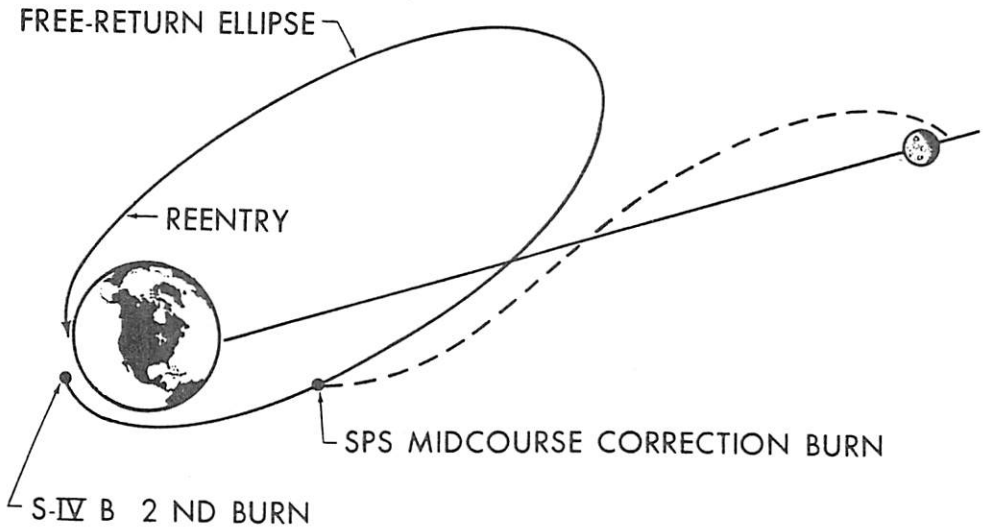


Figure 21

NASA-S-66-5013 MAY 31

EFFECT OF NON FREE RETURN TRAJECTORIES ON 100% ACCESSIBLE AREAS

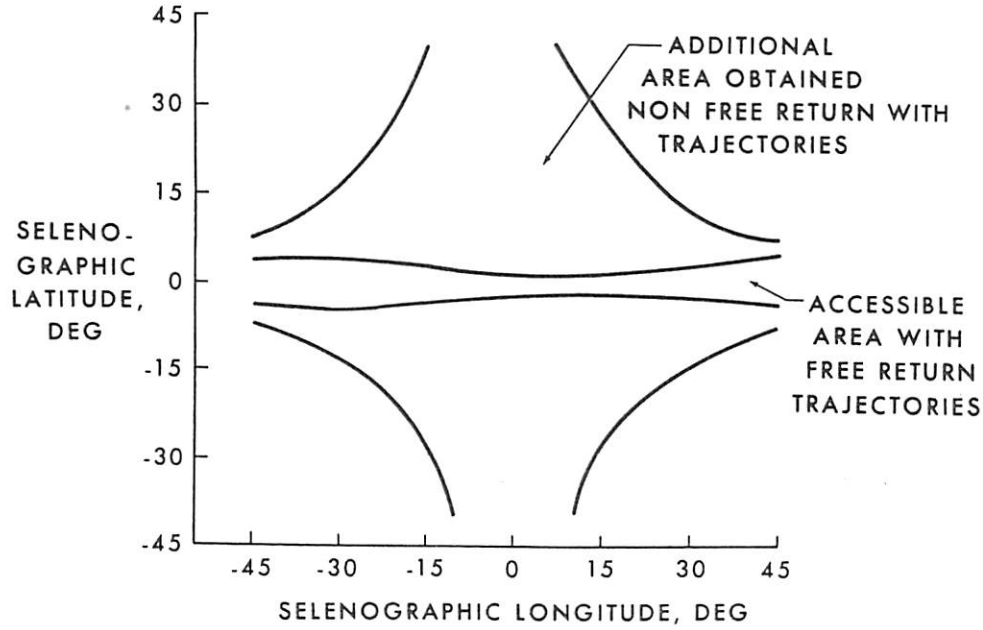


Figure 22

NASA-S-66-6459 JUN

ROTATION OF THE LUNAR LANDING SITE RELATIVE TO LUNAR PARKING ORBIT PLANE

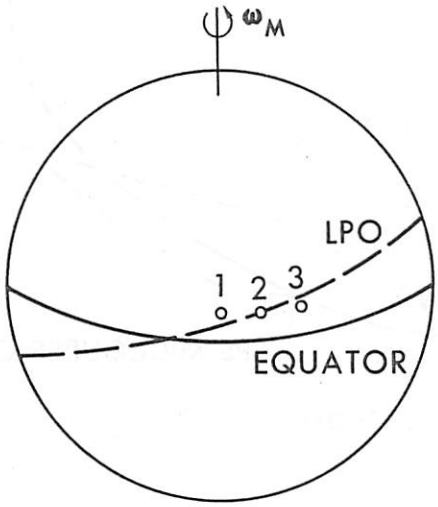


Figure 23

NASA-S-66-5016 MAY 31

EFFECT OF LAUNCH AZIMUTH CONSTRAINT

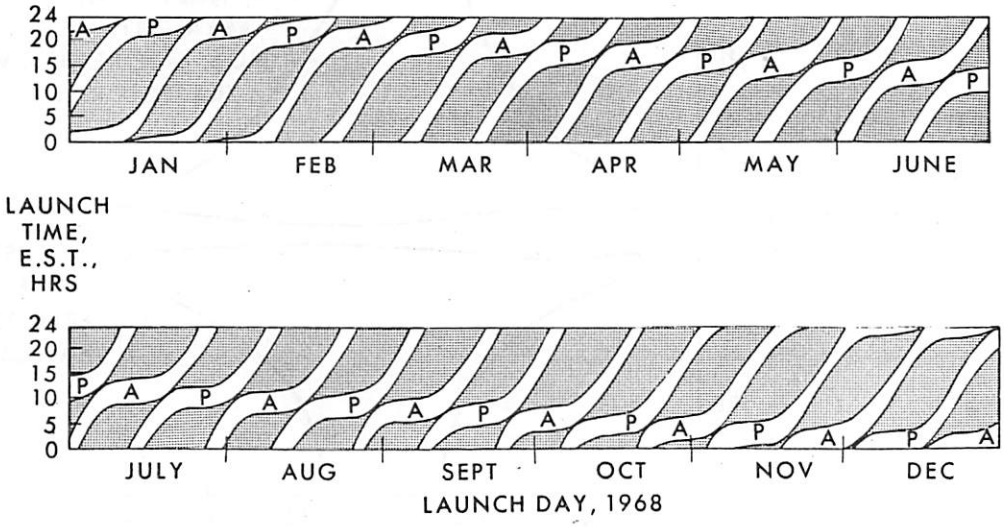


Figure 24

NASA-S-66-5938 JUL 1

LAUNCH AZIMUTH CONSTRAINTS

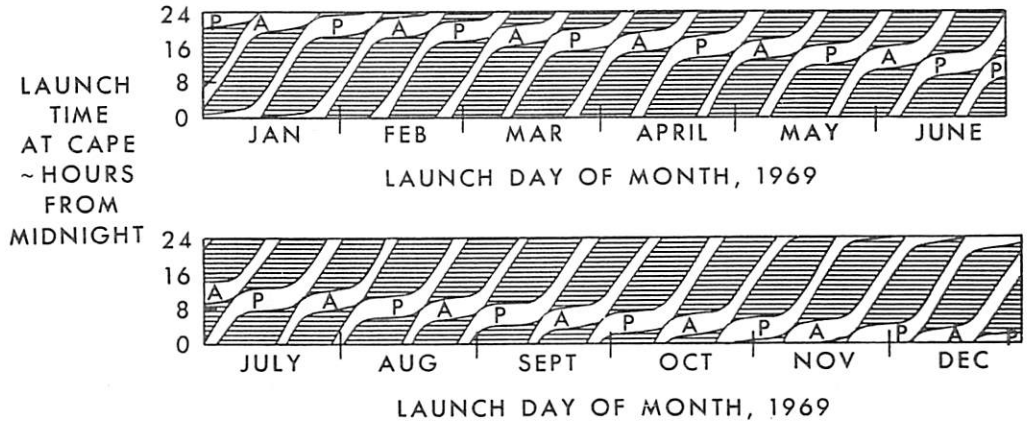


Figure 24A

NASA-S-66-5011 MAY 31

SUN LIGHT CONDITIONS FOR LUNAR LANDING

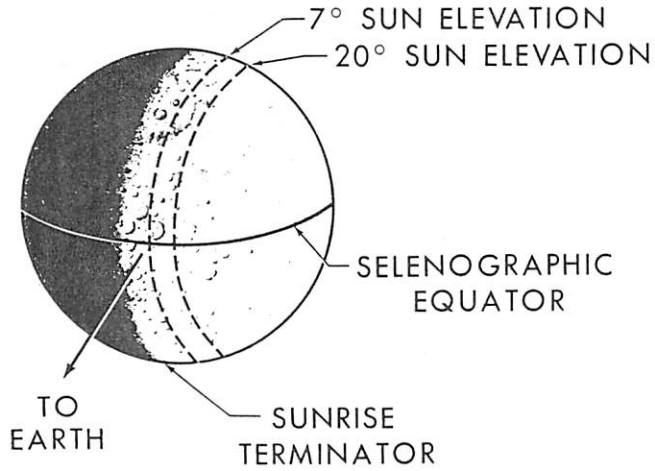


Figure 25

NASA-S-66-5008 MAY 31

EFFECT OF LIGHTING CONSTRAINT ON AVAILABLE LANDING DAYS

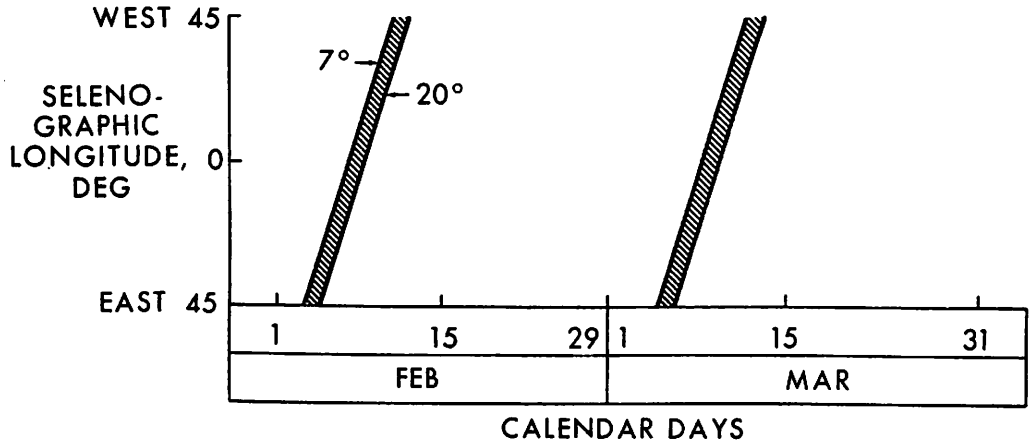


Figure 26

NASA-S-66-5021 MAY 31

LUNAR LANDING OPPORTUNITIES

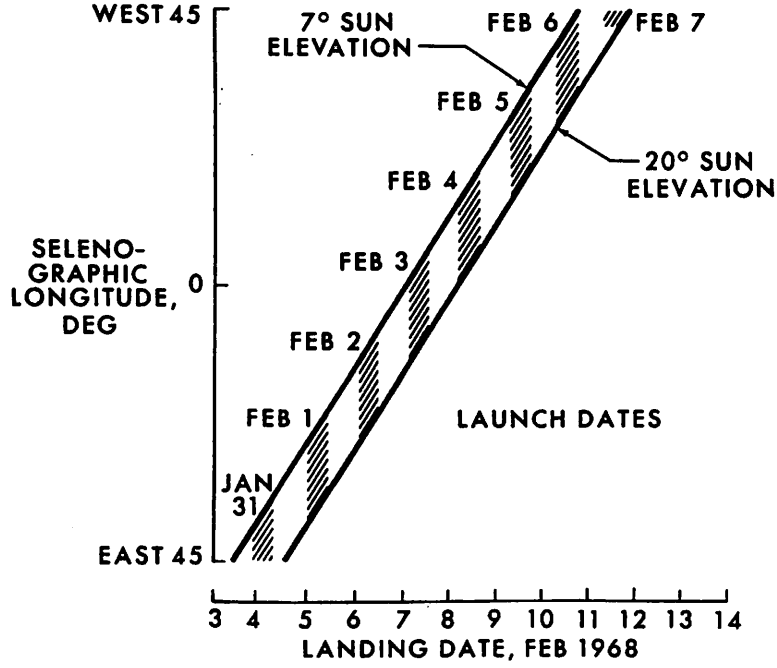


Figure 27

NASA-S-66-5941 JUL 1

LUNAR LANDING OPPORTUNITIES

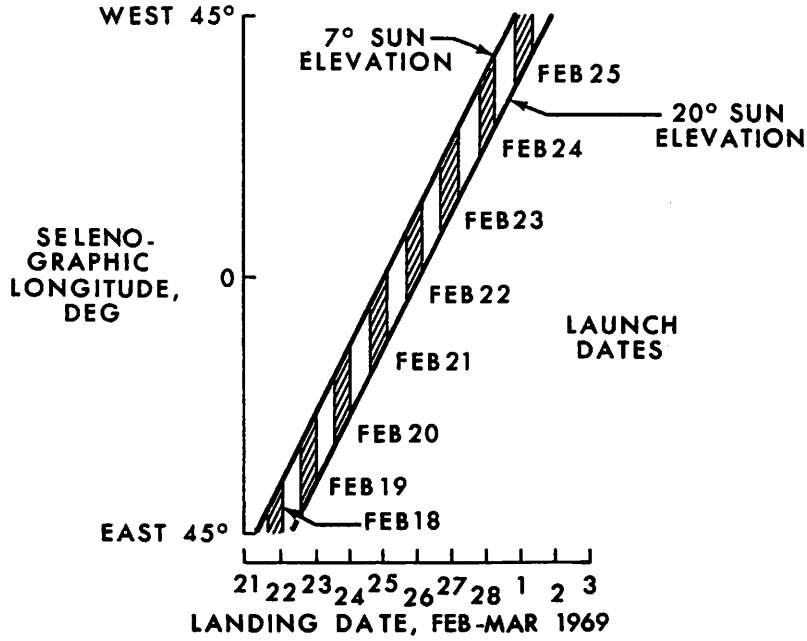


Figure 27 A

NASA-S-66-5017 MAY 31

ACCESSIBLE LUNAR LANDING AREA FOR FEB 1, 1968

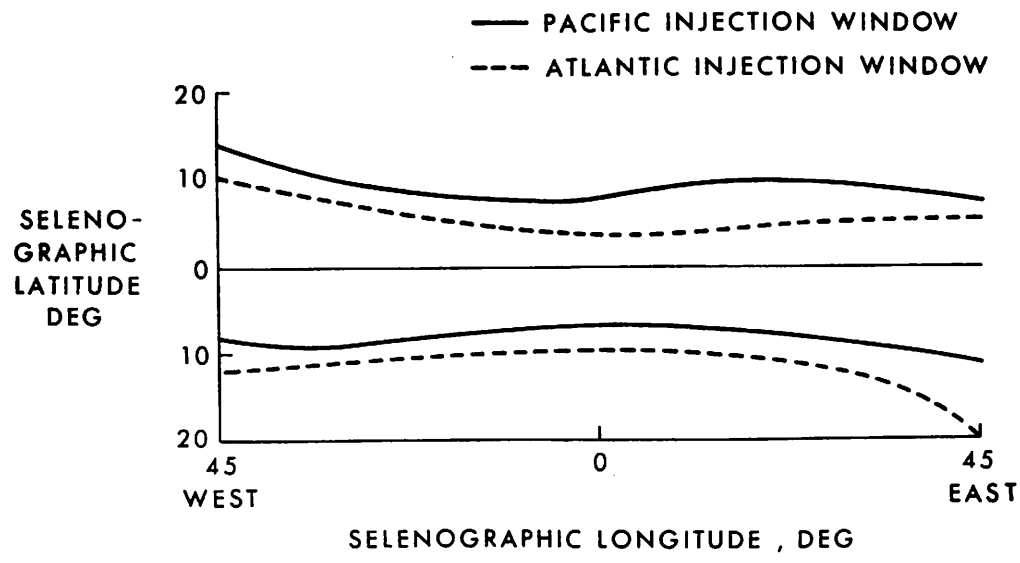


Figure 28

NASA-S-66-5939 JUL 1

ACCESSIBLE LUNAR LANDING AREA FOR FEB 18, 1969

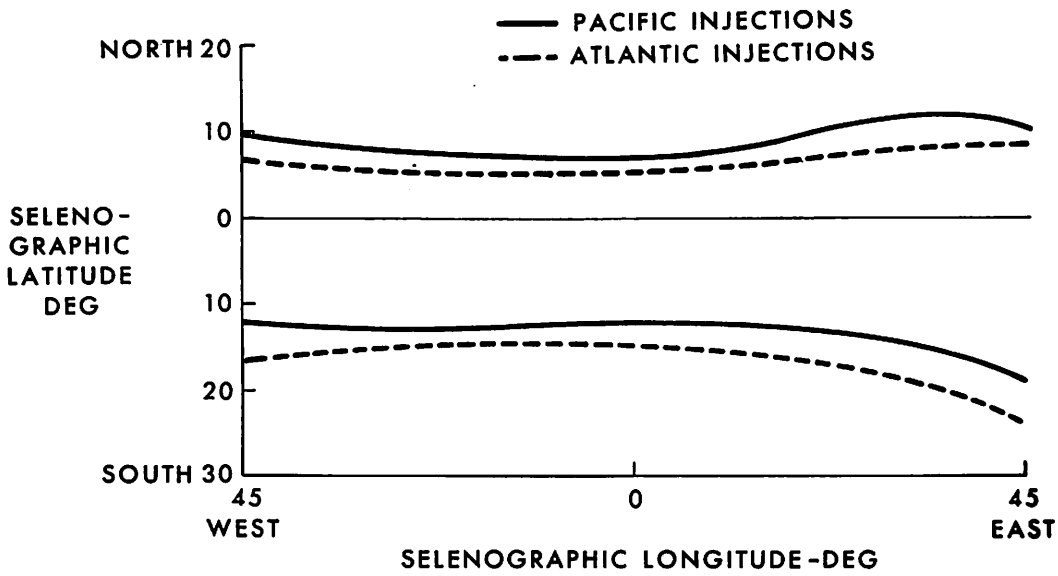


Figure 28 A

NASA-S-66-5179 JUN 13

ACCESSIBLE LUNAR LANDING AREA IN FEB 1968 WHEN THE LIGHTING CONDITIONS ARE CORRECT

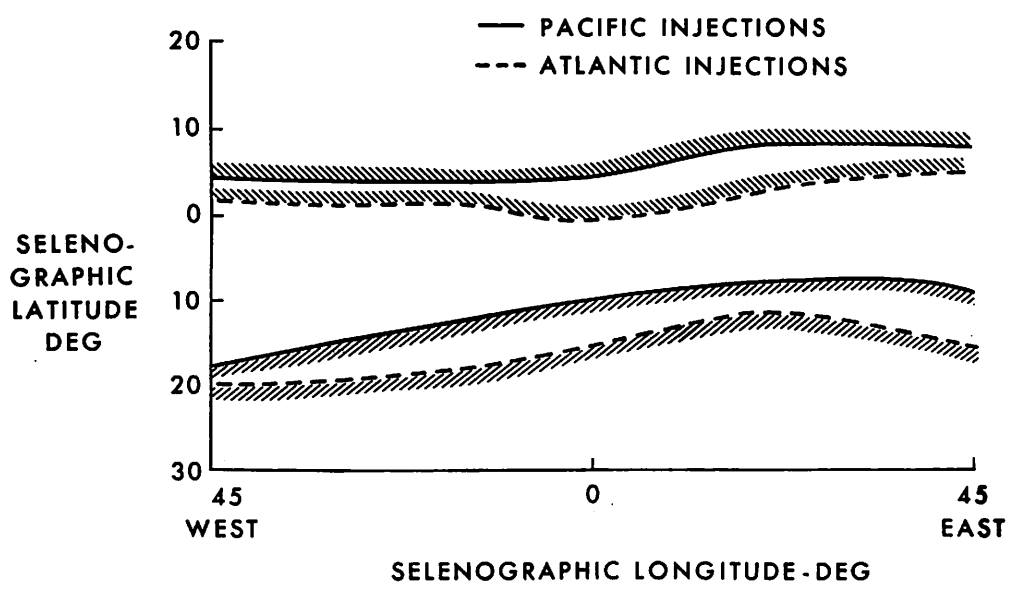


Figure 29

NASA-S-66-5940 JUL 1

ACCESSIBLE LUNAR LANDING AREA IN FEB 1969 WHEN THE LIGHTING CONDITIONS ARE CORRECT

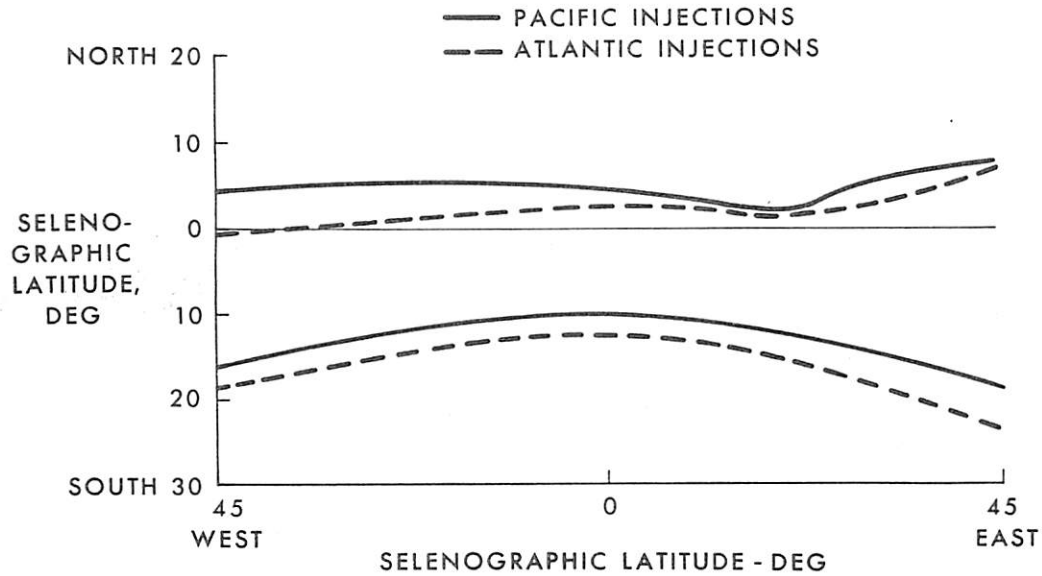


Figure 29 A

NASA-S-66-5182 JUN

100% LUNAR SURFACE ACCESSIBILITY FOR 1968

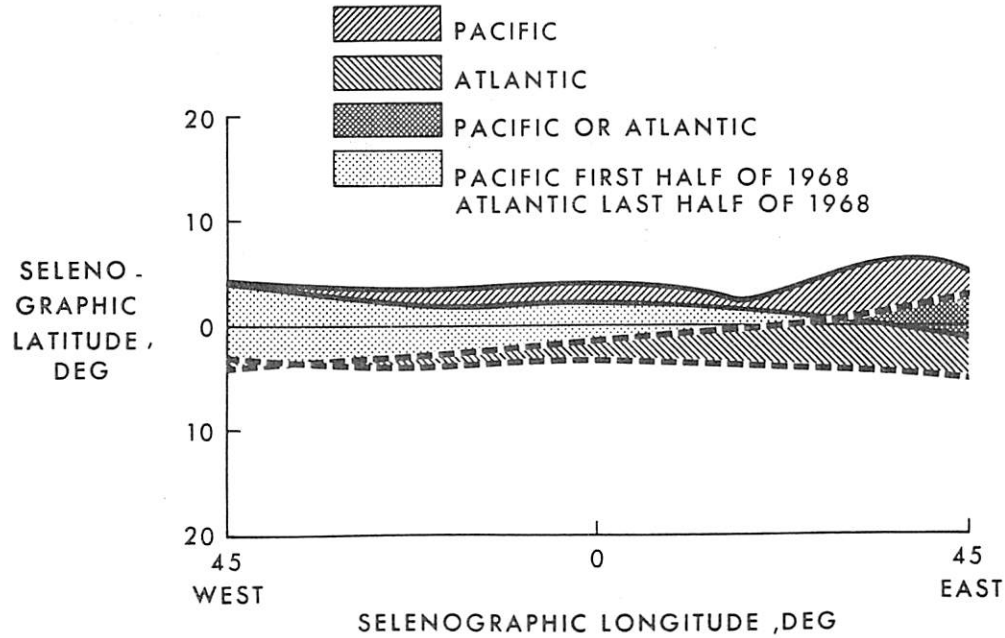


Figure 30

NASA-S-66-5943 JUL 1

100% LUNAR SURFACE ACCESSIBILITY FOR 1969

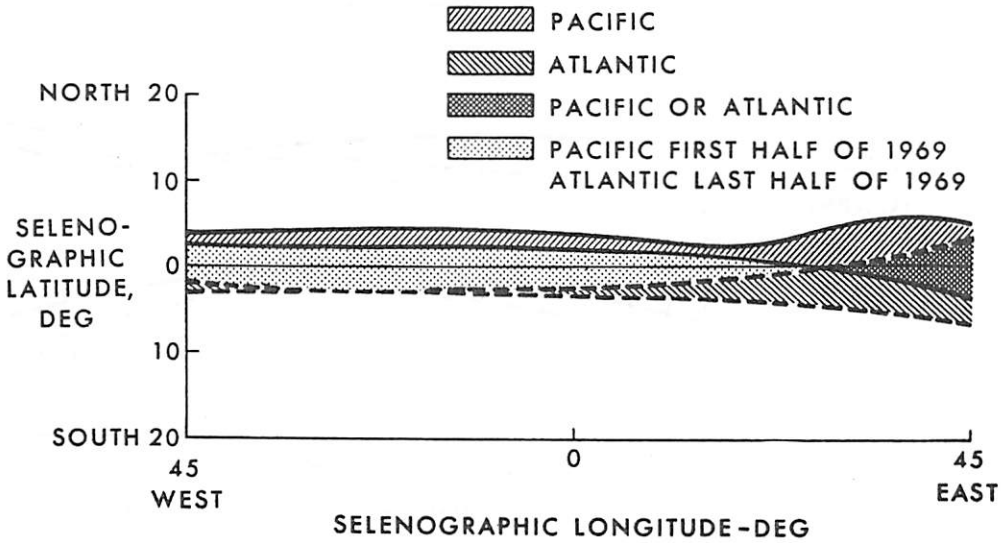


Figure 30 A

NASA-S-66-5005 MAY 31

APOLLO LAUNCH OPPORTUNITIES IN 1968

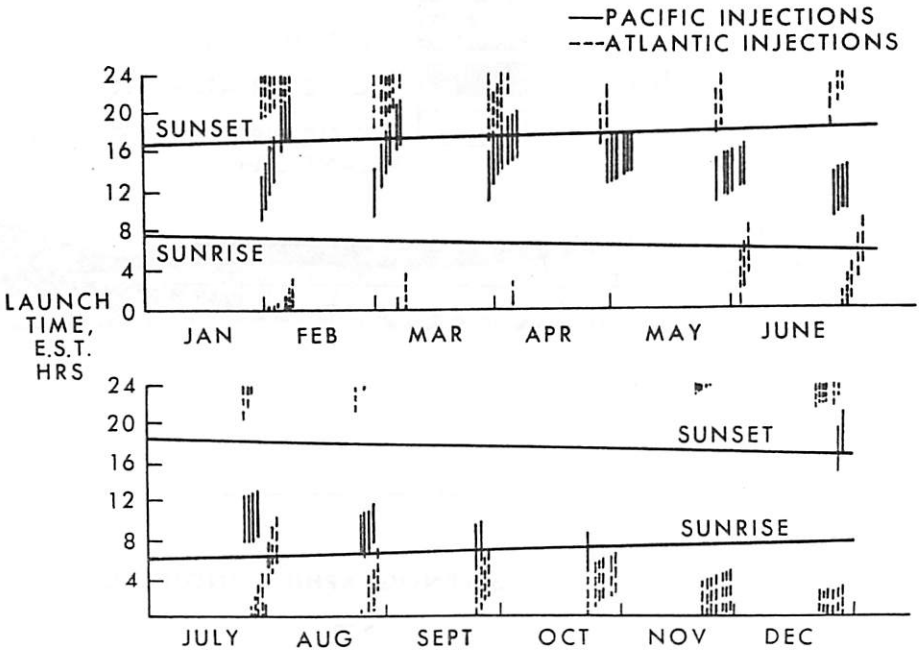


Figure 31

NASA-S-66-5937 JUL 1

APOLLO LAUNCH OPPORTUNITIES IN 1969

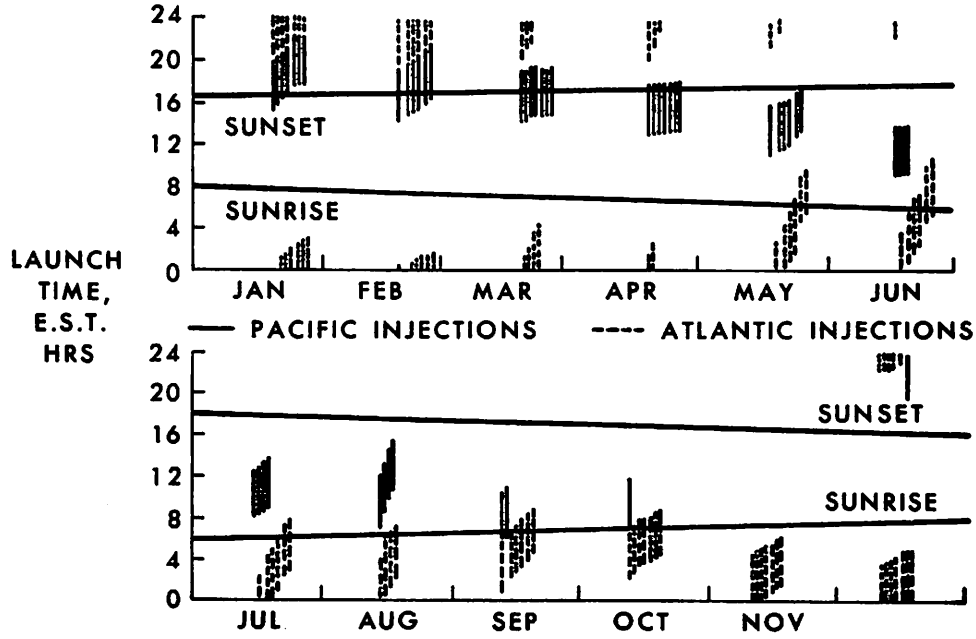


Figure 31 A

NASA-S-66-5006 MAY 31

APOLLO LAUNCH OPPORTUNITIES IN 1968 THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW

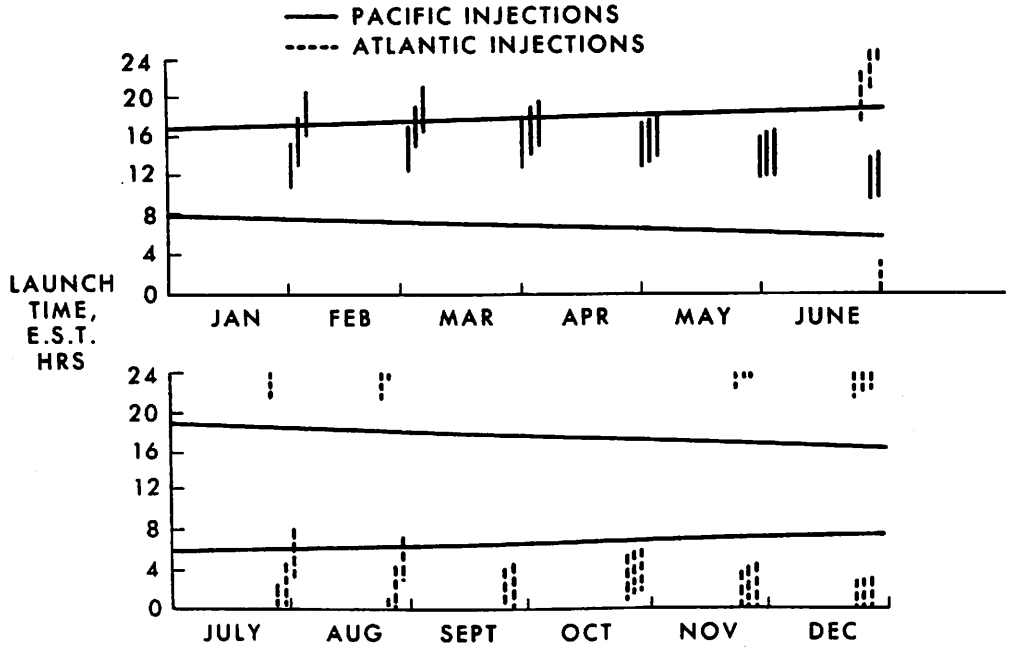


Figure 32

NASA-S-66-5944 JUL 1

APOLLO LAUNCH OPPORTUNITIES IN 1969 THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW FOR ORBITER B SITES

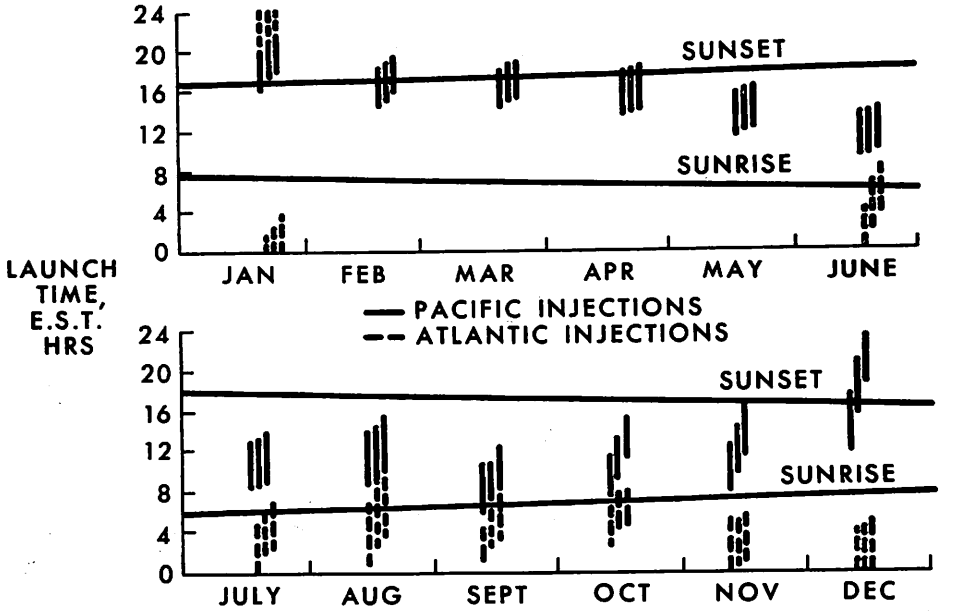


Figure 32 A

NASA-S-66-5206 JUN

APOLLO LAUNCH OPPORTUNITIES IN 1968 THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW FOR ORBITER B SITES

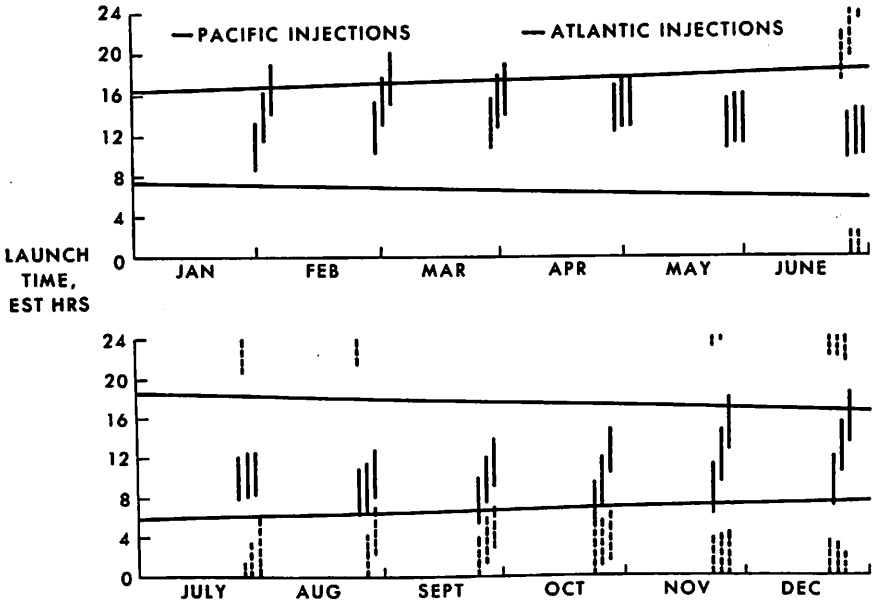


Figure 33

APOLLO LUNAR MODULE LANDING STRATEGY

Donald C. Cheatham
Assistant Chief for
Engineering and Development
Guidance and Control Division

Floyd V. Bennett
Assistant Chief
Theoretical Mechanics Branch
Guidance and Control Division

APOLLO LM LANDING STRATEGY

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SUMMARY

APOLLO LUNAR MODULE LANDING STRATEGY

1.0 INTRODUCTION

The landing of the Lunar Module (LM) upon the surface of the moon will be the climax of the Apollo mission, although the importance of the return phases is not to be de-emphasized. The LM landing approach will be the first time that the complete LM system will have been operated in the lunar environment. This also will be man's initial face-to-face encounter with the exact nature of the terrain in the landing area and of the problems of visibility as they may affect the ability to land the LM; although, these aspects of the landing will be simulated many times in fixed-based simulators and partial preflight simulators. These simulations are extremely important in the preparations for the mission; but only after the mission is completed will it be known how adequate the simulations have been.

Considering the entire LM descent after separation from the Command Module in lunar orbit, a theoretical landing maneuver could consist of a Hohmann transfer impulse on the back side of the moon with a delta V, or change in velocity, of 109 ft/sec, followed 180° later by an impulsive velocity change of about 5622 ft/sec as the LM approaches the lunar surface, as illustrated in figure 1. The flight path angle in the final portion of the approach would be zero degrees. Such a theoretical approach would require infinite thrust-to-weight ratio by the descent engine. This, of course, is an impossible and impractical approach. A finite thrust-to-weight ratio of the descent engine must be used and the approach path must account for lunar terrain variations and uncertainties in the guidance system. Since lunar terrain variations of as much as $\pm$ 20,000 ft. could be expected, and, also, uncertainties in the value of the lunar reference radius, coupled with guidance dispersions, could add another 15,000 ft. to the uncertainty, a conservative safe value of 50,000 ft. was chosen as a pericynthion altitude. From a performance standpoint, the choice of 50,000 ft. as opposed to either 40,000 or 60,000 ft. was quite arbitrary because the difference from the standpoint of fuel requirements was very slight, as indicated in figure 2. The initial thrust-to-weight of the LM descent engine will be about three-tenths. Combining this thrust-to-weight with a perigee altitude of 50,000 ft. leads to the descent profile, as shown in figure 3. The separation and Hohmann transfer maneuver requires slightly

less delta V due to the pericynthion altitude increase. The powered descent portion approaching the landing area, however, requires a delta V of 5925 ft/sec, which is a considerable increase over the infinite thrust requirement. A scaled trajectory profile of this theoretical LM powered descent is shown in figure 4, indicating that the entire descent takes approximately 220 n. m. The LM velocity and attitude is shown periodically along the flight profile. This trajectory has the predominant characteristics of a low flat profile terminating with a flight path angle of about 9 degrees. An obvious feature is that the crew, considering the location of the LM window, never have the opportunity to see where they are going. They can look either directly up, or, if the LM is rotated about its thrust axis, can look down at the surface, but they are never able to see in the direction they are going. If the crew is to perform any assessment of the landing area or out-the-window safety of flight during the approach, it is obvious that the latter portion of the trajectory must be shaped so that a different attitude of the LM can be used during the approach. Shaping the trajectory away from the fuel optimum approach will result in a penalty in fuel requirements. Both the amount of time the crew will require to assess the landing area, and the range from which the landing area can be adequately assessed must be traded off against the amount of fuel involved in the penalty of the shaping. It soon becomes obvious that a strategy is needed that will trade off the system capabilities of the spacecraft and the crew capabilities against the unknowns of the lunar environment encountered during the descent from the orbit, in order to insure that proper utilization of the onboard systems can be made to greatest advantage. The development of this strategy, then, is the subject of this paper.

2.0 STRATEGY CONSIDERATIONS

The LM landing strategy can be defined as the science and and art of spacecraft mission planning exercised to meet the lunar environmental problems under advantageous conditions. In order to plan strategy, the objectives, the problems to be faced, and the characteristic performance of available systems need to be well known. As indicated in figure 5, the objectives of the LM landing planning strategy are to anticipate the lunar environmental problems and to plan the landing approach so that the combined spacecraft systems, including the crew, will most effectively improve the probability of attaining a safe landing. The major factors that must be considered in this strategy are the problems brought about by the

orbital mechanics of the landing maneuver, the limitations of the spacecraft systems (including limitations in fuel capacity and payload capability), and the constraints of the lunar environment (including terrain uncertainties, visibility, and determination of suitable landing positions). The orbital mechanics aspects have been discussed in the preceeding section. The lunar environmental constraints will be discussed in a subsequent section. The remainder of this section is concerned with descriptions of the spacecraft systems and the mission landing position requirements. Although all of the LM systems are important to attain the lunar landing, those affecting the strategy are (a) the guidance and control system, (b) the landing radar, (c) the spacecraft window, and (d) the descent propulsion system.

Spacecraft Systems

Guidance and control system - The guidance and control system is important to the landing strategy in that it has a direct effect upon the area over which the landing may be accomplished and on the problems of landing at a desired point. The functional description and accuracies of this system have been discussed in a preceeding paper. The effect of the guidance, navigation, and control system of the LM on the landing begins with navigation in the lunar orbit. The accuracy of this navigation, whether performed by the onboard system or by the Manned Space Flight Network, determines the uncertainties at the start of the powered descent. Assuming that the guidance system will be updated by landing radar to eliminate the altitude dispersions, the landing dispersions will be a function of the initial condition uncertainties brought about from lunar orbit navigation coupled with the inertial system drift during the powered descent. A summary of the guidance system capability for attaining a given landing point on the moon is presented in figure 6a and the associated assumptions in figure 6b. Both the MSFN and the spacecraft onboard navigation in lunar orbit are considered. The Apollo system specification of a landing CEP of 3000 ft. is met in either case when the inertial system performs within specification.

The 3σ landing dispersion ellipses are shown in figure 7 for cases where the lunar orbit navigation was done by the MSFN and also onboard the CSM. The ellipses are quite similar with the major axis for the MSFN case being slightly shorter and the minor axis for the MSFN being slightly longer than that for the case utilizing CSM onboard navigation. A special case in which the downrange distance was allowed to

be unconstrained is also shown on figure 7. In this case the downrange or major axis of the ellipse is primarily a function of the thrust uncertainties of the fixed-throttle position of the descent engine that will be discussed subsequently. The crossrange axis is equal to that of the 3σ ellipses for guidance to a specific point and is determined by the method of lunar orbit navigation.

Landing radar system - The control of the LM during the descent to the surface can be provided automatically through steering commands generated by the guidance system and also manually by the crew by inputs through an attitude controller. The primary control system stabilization utilizes digital autopilot mode of the guidance computer. Figure 8 shows the attitude thruster firing combinations to create control moments. The engines are located on an axes system rotated about the LM descent engine thrust area 45° from the spacecraft axes. They are operated as control couples for three-axis attitude control. As can be seen in figure 8, two pairs of control couples are available for each axis. The method of providing translational control while in the hovering condition is to tilt the spacecraft by means of the attitude control system. This produces a lateral component of acceleration from the descent engine thrust in the desired direction which is stopped by returning to vertical and reversed by tilting in the opposite direction. During the descent the attitude control system is also coupled to a slow moving gimbal actuator system of the descent engine to enable a means of trimming the descent engine thrust direction so that it passes through the LM center of gravity. The trimming system reduces undesirable torques from the descent engine in order to conserve RCS propellant. The LM landing radar system is important in landing strategy. As indicated earlier, it is used to eliminate the guidance system altitude dispersions and, also, the uncertainties of knowing the altitude from the lunar surface prior to beginning the descent. The LM landing radar is a 4-beam dopple system with the beam configuration shown in figure 9. The center beam measures the altitude, and the other three beams measure the three components of velocity. Two positions of the landing radar antenna provide both altitude and velocity measurements over a wide range of spacecraft attitudes. The first antenna position is tilted back from the thrust axis by approximately forty-three degrees so that the altitude beam will be nearly vertical during the early portions of the descent and, hence, will still provide accurate altitude information. As the LM approaches the landing maneuver, the antenna is physically switched to the second position making the altitude beam parallel to the X-axis of the LM. The landing radar will