the actual phase and height relationship such that in emergency backup circumstances they can manually perform the TPI maneuver making use of the properties of the TPI maneuver previously mentioned, i.e., they would also make use of the set elevation angle of the line of sight and a knowledge of the height differential. Normally the LM prime guidance or backup guidance will control the maneuver.

With the sun direction and line of sight direction as indicated in Figure 7, it is a matter of conjecture whether the crewman in the right hand seat sitting in front of the "shaded" window will be able to see the sun illuminated CSM at time of TPI. The distance is 34 mi. but the possibility hinges on the amount that the right hand window is shaded.

An amplification of the determination of the targeting for TPI occurs in the above reference. It will suffice here to say that the TPI maneuver is intended to achieve an intercept at a position corresponding to 140° of target vehicle travel or an equivalent elapsed time of target vehicle travel. It will incur a ΔV of approximately 25 ft/sec.

At a fixed elapsed time after TPI, a midcourse correction will be applied. Again, the objective will be to bring about intercept at the same position intended at TPI. It will be noted later that there are cross checking navigational guidance systems being brought into play during the concentric sequence flight plan.

At 3 mi. relative range, the first braking will occur. This will be a maneuver again aimed at an intercept, but at the same time instantaneously reducing the closing rate to approximately 20 ft/sec. Subsequent similar maneuvers at reduced relative ranges will accomplish orbit matching characteristics with the CSM. The summation of the ΔV's of these terminal phase maneuvers will be somewhat more than 25 ft/sec.

The docking will be performed manually which will take some time and sometime later behind the moon, crew transfer will be achieved. However, it will be noted that the MSFN was able to participate in the critical navigational guidance phases. The discussion is centered around a nominal launch lift-off time which may be defined as that which would result in a 15 n. mi. differential if navigation and guidance conditions were perfect. The late lift-off is 5-1/2 min. later and corresponds to a zero CSI maneuver, i.e., the late window is essentially defined by the safe orbit

at insertion, referred to AGS launch insertion errors. In the case of the early lift-off, the CDH maneuver will occur approximately 150 from the CSI. In the case of the late lift-off, the CDH maneuver occurs 90 later after the nominally zero CSI maneuver. In general, the CSI maneuver for the late lift-off will not be zero because of launch errors. It should be noted however, that only posigrade CSI maneuvers are tolerated. If the dispersed conditions at CSI for the case of the late lift-off potentially demand a retro-impulse, then in this case, no maneuver will occur and the consequence will be that the finally selected TPI time after CDH will be displaced from the original nominal TPI time. In this case, perhaps a little more than desirable, but the probability of the case is low.

Figure 8 - Concentric Flight Plan in Relative Coordinates

The flight plan indicated on Figure 7 is shown in this coordinate system so that it can be seen to scale. The coordinate system is set in the CSM and height differential is shown in the direction of the "current vertical" through the CSM and phase difference is shown in the direction of the current horizontal through the CSM plane. Essentially, the moon has been opened up and made flat. Hence, the 80 n. mi. circular orbit becomes a horizontal line. The concentric coast is shown parallel to it.

The notes of background refer to the conditions with which the sextant has to view the LM. The nautical mile figures refer to the relative range between the LM and the CSM. The orbital rate of the CSM is approximately 3 per minute, hence there is a large loss of phase during the 6.9 min. launch period.

Study of Figure 8 gives a fresh perspective to the CFP.

In the next section a brief summary of the propulsion systems used is given.

Figure 9 - Utilization of Propulsion Systems

The main engine, with a fixed nozzle, is used to provide the power for launch. In addition, a forward thrusting RCS engine is used to null out the effects of cg thrust offset. Both engines are using Ascent Propulsion System (APS) fuel. It should be noted that there is sufficient APS fuel to launch into a safe orbit even if there is an RCS failure where resort has to be made to an RCS engine thrusting backwards to null out the cg-thrust offset. This is true when referred to all probable adverse cg offset conditions.

It is considered wise to use the residual useable APS fuel for the next maneuver after launch, i.e., the CSI maneuver. However, the pressurized RCS tanks must first provide fuel for ullage action by the X axis RCS thrusters. Ullage having been provided, then following a switching sequence, the same RCS thrusters are able to use the residual APS fuel.

For the remainder of the maneuvers for rendezvous, the RCS engines are used and the fuel is taken from the pressurized RCS tanks. It is planned to use Z axis thrusters for nearly all cases from TPI on.

At the bottom of Figure 9, there is a reminder that the CSM may be involved in active rendezvous maneuvers. When the ΔV is more than 12 ft/sec, CSM RCS will be used for ullage and then main engine thrust is applied. If the ΔV required is less than 12 ft/sec, then the CSM RCS alone will be used. An example of this is the final terminal thrustings to complete CSM active rendezvous.

The next section reminds us that a time critical rendezvous may be required.

Figure 10 - Time Emergency Rendezvous

This figure indicates the time critical flight plan which will be employed. It consists of a standard insertion from launch exactly as in the nominal mode. This is followed by approximately a 5 min. review of the launch cutoff conditions. Subsequent to this, a direct transfer of approximately 100° is selected. Subsequent to the transfer initiation, midcourse corrections will occur. At a relative range which will call for braking maneuvers, the closing rate prior to first braking will be higher than that in the nominal mode. As indicated in the figure, the time line is a busy one and this mode of rendezvous will only be employed in time emergency circumstances.

Necessary studies have taken place to examine vehicle to vehicle tracking possibilities for the CFP.

Figure 11 - Optical Tracking Assumptions

Figure 11 summarizes the sextant tracking assumptions for the studies and the conditions have two major categories:

1. Where the line of sight has a radial component outwards.

2. Where the line of sight has a radial component inwards.

Associated with the first category, sextant viewing conditions are acceptable if the angle between the sun direction and the line of sight is greater than 20°.

Associated with the second category, is the probability that there will be a lunar background to the LM. This corresponds with the LM being below the horizon.

The figure indicates the maximum viewing range for various background conditions and it should be noted that it is the sun illuminated LM rather than the flashing light of the LM, that can be detected against a sunlit lunar background.

The kind of analysis that has been made will be shown in the next series of figures, but first, attention must be paid, to the notation used.

Figure 12 - Explanation of Line of Sight Convention

Figure 12 gives an explanation of the line of sight convention. The zero datum for the local horizontal through the CSM in the direction of motion and a 360° convention is used.

Figure 13 - CSM Sextant Tracking LM (Nominal Flight Plan)

This is an example of conditions you have when a landing has been made with the sun elevation of 18° and when the lift-off corresponds to the nominal and the concentric altitude of 65 n. mi.

The status of the mission is given by the trace with the events of the CFP indicated with appropriate symbols. The status of the mission is given in terms of the line of sight, using the convention shown in the previous figure, and the longitude of the LM.

At insertion, the CSM looks back to see the LM on the horizon and as the LM at a lower altitude begins to catch up, the line of sight steepens and the LM then becomes viewed against a lunar background. First the background is sunlit and then it is that of earth reflected light, but the range is greater than 250 n. mi. and the flashing light can be acquired. Hence, the trace shows a series of dots to indicate trajectory status. However, the tracking summary at the bottom of the diagram quite correctly indicates a gap in tracking.

After CSI, the LM altitude increases and the LM is seen against a space background which corresponds to a dark background, and

since the range is less than 500 n.mi., the LM flashing light can be seen and the tracking summary shows a definite line of "footballs" indicating that acquisition is possible. Following the circularization or CDH maneuver, the line of sight steepens again and the LM is viewed against a sunlit background. However, the range at TPI is 34 n.mi. and hence, just prior to it, the range is 40 n.mi. or less and hence, the sun illuminated LM can be seen against the sunlit background. After TPI, the relative range becomes less and hence, sextant acquisition continues to be possible. Toward main braking, the lunar background is that of earth reflected light and again, by reference to the Optical Tracking Assumptions in Figure 11, it will be seen that acquisition is possible and this is recorded in the tracking summary. In fact, favorable lighting conditions prevail right through to docking.

The assumption was that the rendezvous radar could acquire when the relative range was 400 mi. or less and in the example given, the range is always less than 400 mi. and, therefore, rendezvous radar acquisition is continuous.

MSFN acquisition can occur from approximately 100° east to 100° west and this is indicated in the tracking summary.

In the case of the late window, the concentric altitude is 30 n.mi. and the concentric coast between circularization and transfer would be done with the LM below the horizon and at a greater relative range with the LM against a sunlit background. Hence, when a spectrum of flight plans is reviewed later, a gap in sextant viewing corresponding to this phase for the late window should be anticipated.

The late window resembles very closely, the conditions for abort from hover. Therefore, it should be borne in mind when considering the late window that as regards lighting conditions, a similar pattern prevails for the abort from hover.

Figure 14 - CSM Sextant Tracking LM (Abort from Initiation of Powered Descent)

This is an example of an abort from the initiation of powered descent. The landing site is at zero longitude and the sun elevation was 30° upon landing. The CSM is 10° behind the LM initially at the abort warning and the LM commences its powered descent at 50,000 ft. It will be seen on Figure 14 that the sextant is viewing the LM below the horizon at insertion and the LM is viewed against a sunlit background.

At CSI, the LM is inserted into an orbit the apocynthion of which, is 130 n. mi. and a concentric coast above the CSM commences at this altitude. Therefore, before circularization, the sextant will have to look above the local horizontal to see the LM, and hence, the trace giving the status of the mission indicates an elevation angle of above zero for that mission phase.

After circularization, the line of sight comes within 20° of the sun direction and when it does, it will be noted that there is a gap in the tracking summary at the bottom of the figure. Thereafter, there is a dark space background for the sextant to see the LM flashing light while the line of sight steepens and sextant tracking conditions will remain favorable right through to main braking, which is followed by subsequent terminal phase maneuvers. However, it is not suggested that the sextant is of signifcant operational use during the terminal maneuvers.

The rendezvous radar is within its specification acquisition range of 400 n. mi. and therefore, continuous rendezvous radar range tracking is indicated.

Again, it will be noted that the MSFN can track approximately between 100° east and 100° west.

This is an example of sextant tracking when the LM has to go above and let the CSM catch it up and it will be seen that there are no particular sextant tracking problems in this case. The next figure shows several cases involving nominal flight plans for early and late window, and different sun elevation angles at landing. Similarly, abort cases from the initiation of powered descent are shown.

Figure 15 - Tracking Summary

The cases are identified on the left of the figure, the sun directions during rendezvous are shown on the right. The key to this figure has been explained on the two previous figures. It will be noted that from CSI onwards, there are always two sources of navigation. It will be noted in general, that from TPI onwards, there are three sources of navigation. The exceptions to this are the late window cases and similarly, aborts from hover would suffer loss of CSM sextant tracking. The reason for the gap around TPI was mentioned in the discussion of the late window nominals.

However, the potential of the MSFN should not be forgotten. A reminder of this potential is given in the immediately following sections.

Figure 16 - MSFN Error Analysis Assumptions

A summary of the error assumptions is given on this figure. It will be seen that standard 2-way and 3-way error source assumptions have been adopted. MSC and GSFC are in agreement on the quantitative values.

Station location biases were inserted into the study. Station positions were of the order of 100 ft. in error. However, it must be realized that this is not particularly important as the only observation used in the trajectory determination processes is Doppler rate. The systematic error on the moon's gravitational constant is indicated and the JPL derivation is indicated. The analysis used a simulation of the proposed RTCC orbit determination scheme. The three components of velocity are solved for, the three components of position are solved for, and the individual systematic errors on both of the two slaved situations are solved for. This is what is meant by bias on 3-way Doppler as shown at the bottom of the figure.

The next two figures summarize some of the results of the analyses.

Figure 17 - MSFN Tracking Uncertainties (3σ)

This is an analysis of the MSFN tracking during a CFP where the launch was from 45° west. The short tracking pass subsequent to launch is, in itself, of marginal use. This is, in some measure, due to the length of time for update by voice and telemetry verification by the ground. The complete update allowance is of the order of 7 min. However, tracking continues during this time and it is the basis for good a prior information for when the MSFN reacquires when the LM reappears on the eastern limb. Whereupon, convergence is rapid and uncertainties become low. It should be noted that the out of plane uncertainties are reduced because of the two tracking passes, the initial one on the western limb, and the subsequent one on the eastern limb. The analysis did assume what are currently considered to be 3σ execution errors.

The next figure indicates the potential of the MSFN in assisting the AGS.

Figure 18 - (3σ) MSFN Errors at Intercept

In this figure, it must be recognized that a midcourse correction is applied always at 30 min. elapsed time after TPI. The abscissa indicates that TPI is delayed by varying increments of time after MSFN reacquisition on the eastern limb. The left and right ordinates

indicate the errors in position and velocity respectively for intended zero miss conditions at intercept. The execution errors at TPI and midcourse are assumed to be insignificant and it is only the navigation performance of the MSFN to guide the LM to an intended zero miss that is illustrated.

Some emphasis has been given to the fact that it is the potential MSFN performance that has been illustrated. This is done because those with operational experience at MSC consider that the network, coupled with the trajectory determination organization of the ground control, must have exercised this capability prior to the lunar landing mission in order to achieve the potential. It must be recognized that none of the development flights orbit the moon; therefore, there is great dependence on the opportunities that could be provided by the Lunar Orbiter program. The extent of the opportunities remain undefined at this time.

The next diagram gives a brief reminder of the main programs involved in the rendezvous capability.

Figure 19 - Main Components of Onboard Rendezvous Programs

This diagram reminds us that three major programs previously mentioned in the paper are formulated as onboard programs in the prime and backup guidance computers of the LM.

The programs referred to are as follows:

1. The "standard insertion" which is used for launch and is the major component of the powered phase of the abort from powered descent. This applies to both the LM prime and backup guidance systems.

2. The second one is direct transfer. This routine is contained in the LM prime and backup guidance systems and is also formulated for the CMC. The final major transfer is always made using this routine and it is essentially used with subsequent midcourse action.

3. The concentric sequence is the third program. This program has, of course, been mentioned and additional information on it is contained in the above reference.

There is a fourth program which has not been mentioned and this is the "external ΔV" program, and is used when a required ΔV input derived externally, either from pilot's notes, charts, etc., or from ground control. The input is referenced to local polar coordinates such that the crewmen have an understanding of what they are entering into the computer.

All of these programs are being formulated in the RTCC. This is in order that advantage may be taken of the MSFN tracking capability, which in many cases, would provide the source for the determination of targeting to be transmitted by voice.

In the following section, a reminder is given of the total G&N information during rendezvous.

Figure 20 - Provision of G&N Information for the Operation

It will be observed on Figure 20 that there are three sources of observation, i.e., rendezvous radar, MSFN, and the CSM sextant. The diagram indicates the way in which the observations find their way into the onboard computers. If data has to be entered manually through the keyboards, then this is also shown. If this is subsequent to voice transmission, this is shown. Hence, the diagram gives reasonable clarity and an immediate picture of how navigation data finds its way into the computers. The CSM orbital elements are inserted into the LM computer. The LM inertial orbit is subsequently determined by rendezvous radar observations of the CSM. At the same time, the CSM sextant can take observations of the LM and derive the LM inertial orbit and, therefore, potentially this provides an onboard navigational check.

It is most important to note the resultant capabilities indicated on the right hand side of this diagram. Both LM computers have a concentric sequence capability and a contingency time critical "equivalent direct ascent" capability.

The CMC has a direct transfer capability only. However, if the CSM state vector is entered into the LGC, and the LGC calculates a concentric sequence as though the LM had that state. In this way, the LM can prepare targeting information for the CSM which is transmitted by voice and is entered through the DSKY of the CMC.

In the next section, a number of contingency flight plans are being introduced in order to give an indication of how the CFP can be used in contingency cases.

Figure 21 - LM Active Contingency Rendezvous (Phasing at Insertion 60°)

The coordinates of this diagram and the succeeding three are exactly the same as used on Figure 8 (Concentric Flight Plan in Relative Coordinates). An explanation of the coordinates is contained in that section. Briefly, the moon is transformed into a flat moon and measurement in the local vertical direction is that of differential altitude and measurement along the local horizontal is that of phase lead.

In this first diagram, the phase lead of the CSM at insertion is 60°. The familiar CFP maneuvers are employed. It should be realized that when the differential altitude is small, the catch up rate is small. Therefore, though the contingency flight plan has been framed such that the differential altitude subsequent to CDH is of a differential altitude not too dissimilar from that of a nominal lift-off case, more than one orbit of concentric coast is incurred.

Figure 22 - LM Active Contingency Rendezvous (Phasing at Insertion 140°)

In this case, the CSM is further ahead, 140° at insertion. In this case, the greater differential altitude is selected and multiple orbits of concentric coast are incurred. However, the familiar concentric sequence is easily recognized.

Figure 23 - LM Active Contingency Rendezvous (Phasing at Insertion -50°)

In this instance, the CSM is 50° behind the LM at insertion. Consequently, the CSI maneuver drives the LM to an apocynthion above the CSM whereupon the CDH maneuver will be executed to achieve a concentric coast above the CSM prior to TPI. The transfer following CSI is the equivalent of a Hohmann transfer initiated of equal period to the previous orbit.

Figure 24 - LM Active Contingency Rendezvous (Phasing at Insertion -160°)

In this instance, the LM is very much in front of the CSM, in fact, 160° at insertion. It also is in a lower orbit following insertion. This would call for the CSI maneuver to be initiated immediately after insertion in order to end the increasing adverse phase situation. The CSI maneuver transfers the LM high above the CSM and at apocynthion, the CDH maneuver achieves a concentric coast at a considerable differential altitude above the CSM in order to retain the required negative catch up rate. Finally, the TPI maneuver initiates a rendezvous from above.

One of the main reasons for drawing attention to these contingency flight plans shown in these particular coordinates was in order that a particular feature could be recognized. This feature is that the concentric sequence pattern is just as applicable in these cases as in the nominal case, i.e., the concentric sequence program onboard has the capability to cope with these contingency cases.

The next section will give a summary of the overall Apollo rendezvous situation.

Figure 25 - Rendezvous Categories

The main correlating parameter by which a rendezvous situation can be immediately judged is phase lead. Particularly is this the case if a nominal CSM circular orbit of 80 n.mi. altitude is a constant assumption. Therefore, in these circumstances, all rendezvous situations can be categorized according to phase and a summary is graphically indicated on this diagram.

All phase leads of the CSM are calibrated relative to the LM at insertion and zero phase lead is found at the bottom of the circle. All phase leads referred to on the left hand side of the circle refer to the central angle that the CSM is in front at LM insertion. The number of degrees on the right hand side of the circle refer to the central angle lag that the CSM has at LM insertion.

First of all, the nominal launch window is identified, i.e., for a nominal lift-off, the CSM will be 18° ahead at launch cutoff. If the lift-off occurred $5\text{-}\frac{1}{2}$ minutes later, corresponding to the end of the launch window, then the CSM would be 32° ahead of the LM at insertion.

In the late case, the concentric differential altitude is 50 n.mi. In the early window case, the concentric differential altitude is 15 n.mi. Extrapolating this information, it can be understood that when the phase lead is 10° at LM insertion, then the corresponding concentric differential altitude would be zero. In this way, we have identified a physical boundary at which the CFP neither acquires a positive or negative catch up rate during the concentric coast. It should be noted that in the instance of this particular phase lead being incurred, then the CFP can be framed to have a concentric differential altitude coast by advancing or retarding the time of CSI. It is now convenient just to use the 10° boundary as a phase delineation where, if the CSM phase lead at insertion is greater than 10°, then the concentric altitude coast will be below that of the CSM. Alternatively, if the phase lead of the CSM is less than 10°, and this includes phase lags, then the LM concentric coast will be above that of the CSM for completely LM active rendezvous.

Circular arror A_1 indicated all the CSM phase leads at insertion from which the LM can catch up from below within the ascent module lifetime of 11.3 hours by rendezvousing from below.

If it is desired that the concentric coast shall be specifically 15 n.mi. below the CSM, then circular arrow A_2 indicates the reduction of range of lead angles corresponding to which it is possible for LM active rendezvous within 11.3 hours.

Circular arrow A_3 indicates, in the main, the phase lags of the CSM at insertion corresponding to which LM active rendezvous may be achieved within the ascent module lifetime following a concentric coast above the CSM. Because this entails the LM going higher than the CSM, it must be noted that A_3 is bounded by ΔV capability allowing sufficient negative catch up for rendezvous within required lifetime.

It should be noted so far that we have only dealt with in plane rendezvous cases. This is not unnatural in that nominally the CSM has made a desirable plane change prior to LM ascent, or in the case of multiple orbit rendezvous, if for any reason the CSM has not previously made a plane change, then it has the opportunity to make the change at a desired node. It can do this without using any of the ΔV allocated to CSM rescue since there is a 2° allowance in the nominal mission. However, it is interesting to note that an entirely LM active rendezvous with a $.5^{\circ}$ wedge angle incurred by dispersions that circular arrow A_4 indicates the reduced range of phase lags, in which we include a small range of phase leads 0° to 10°.

Circular arrow B indicates that by maneuvering the CSM that the "black shaded" region between 130° and 160° phase lag can easily be covered. In fact, when the CSM is called upon to maneuver not only can it negate any out of plane situation by a Hohmann transfer down to a lower circular orbit, it can improve the phase situation such that the total time to rendezvous will be reduced.

In the event of an any time lift-off where the LM is forced into undesirable phase situations, together with the circumstances that the LM propulsion systems just manage to insert the LM into a safe orbit, then it should be noted that the nominal CSM plane change capability, plus the ΔV allowance for CSM rescue, makes it possible in the last resort for the CSM to rescue the LM regardless of the initial phase situation and out of plane displacements of less than 2°. Again, the lifetime assumption for the ascent module is assumed to be 11.3 hours and nominal allowances were made for docking and crew transfer.

In the next section, rendezvous modes in order of preference, are discussed.

Figure 26 - Rendezvous Logic

Before considering the order of preferences, it should be remembered that the main constraints are: (1) ascent stage lifetime, (2) LM ΔV available, and (3) clear pericynthion transfers for both LM and CSM.

A desirable condition is that of satisfactory lighting during the terminal phase. Another very desirable condition is that the final terminal maneuvers are accomplished by the LM RCS thrusters.

The first preference is a LM active rendezvous. Furthermore, in non time critical situations, the concentric sequence will be performed and rendezvous shall be achieved in approximately one orbit. In time critical situations, the "equivalent direct ascent" plan shall be employed. If entirely LM active rendezvous can be accomplished from below by employing multiple orbits and the rendezvous can be accomplished well within the ascent module lifetime, then this is considered preferable to combined maneuvers.

Combined maneuvers involving CSM maneuvers prior to the employment of the CFP for the LM may be utilized for either of the following reasons:

1. To accomplish the rendezvous well within the LM ascent stage lifetime.

2. To avoid the situation where the LM has to rendezvous subsequent to a substantial differential concentric altitude above the CSM prior to rendezvous from above. This latter case may also be bounded by the LM V budget. In addition, it may be contrary to satisfying operational requirements.

The third preference is an entirely CSM active rendezvous and would only be employed in contingency cases where the LM propulsion systems were just capable of inserting the LM into a minimum safe orbit.

At this juncture, an overall summary is given.

Figure 27 - Summary

There are rendezvous plans for every phase situation and for all practical out of plane conditions, specifically, less than 2°.

In all normal cases, both nominal and non time critical contingency cases, the concentric sequence standard pattern of procedures will be employed.

An equivalent direct ascent plan is available for time critical rendezvous situations.

Steps have been taken to utilize more efficiently the LM backup guidance system capability. The concentric sequence can be controlled by it and in addition, it also has the equivalent direct ascent capability. In this way, the probability of having to resort to an entirely CSM active rendezvous has been reduced.

- Finally, it should be noted that in all non time critical rendezvous situations, joint ground and onboard decisions may be made prior to and subsequent to time critical intercept trajectory.

Questions and Answers

LUNAR EXCURSION MODULE ASCENT AND RENDEZVOUS

Speaker: Morris V. Jenkins

1. Mr. Richter - Are DEDA inputs required for inputs to the AGS using normal flight plan? Subsequent to launch?

   ANSWER - Radar information must be keyed into the AGS.

2. Dr. Rees - Is such a short lunar prelaunch checkout adequate?

   ANSWER - There is an extensive checkout of the LM prior to earth launch. The LM is also checked out as thoroughly as possible following LM landing and again prior to launch.

3. Dr. Rees - What is the maximum LM lunar surface stay time?

   ANSWER - Approximately 35 hours for the descent stage. The ascent stage has a contingency capability of about 9 hours, plus an ascent time of about 2.5 hours.

4. Mr. Stern - What data exists on coverage by 2 MSFN stations, as compared to 3 station coverage shown in presentation?

   ANSWER - This data is available and will be sent to Mr. Stern.

NASA-S-66-6848 JUN

APOLLO RENDEZVOUS

- NOMINAL AND CONTINGENCY PLANS ARE AT AN ADVANCED STAGE
- EVALUATING OPERATIONAL TRADE-OFFS
- IMPLEMENTATION IN LGC, CMC, AEA, RTCC
- CREW G AND N PROCEDURES AS-278 ⟶ AS-503
- CROSS CHECKING NAVIGATIONAL SOURCES
- LEM ACTIVE RENDEZVOUS POSSIBLE IN ALL CASES
- LM-CSM COMBINATION PLANS NEXT PREFERENCE
- CSM POTENTIAL RENDEZVOUS IN ALL CASES
- STEPS TO LESSEN PROBABILITY OF RESORTING TO CSM TOTAL ACTIVE RENDEZVOUS

Fig. 1

NASA-S-66-6572 JUN

THE CHANGE

PAST	PRESENT
● DIRECT ASCENT – NOMINAL – CONTINGENCIES WHENEVER POSSIBLE • E G COMMENCEMENT OF POWERED DESCENT	● CONCENTRIC FLIGHT PLAN – NOMINAL – ALL NORMAL NON-TIME CRITICAL CONTINGENCIES • E G ABORTS FROM ALL PHASES OF DESCENT, COAST RIGHT THROUGH TO HOVER
● INTERMEDIATE PARKING ORBIT • CONTINGENCIES REQUIRING LARGE CATCH UP RATES • E G ABORT FROM HOVER	● EQUIVALENT DIRECT ASCENT • FOR ABNORMAL TIME CRITICAL SITUATIONS • E G FAILURE IN LIFE SUPPORT SYSTEM

Fig. 2

NASA-S-66-6548 JUN

THE OBJECTIVES

- TO FACILITATE A TIME LINE IN WHICH CREW AND GROUND SPECIALISTS CAN PARTICIPATE IN OPERATIONAL DECISIONS WITHOUT A TIME PRESS
- TO FACILITATE EFFECTIVE CREW MONITORING TECHNIQUES
- CONSERVATIVE STEP-BY-STEP COMMITMENT
- FULL UTILIZATION OF MSFN AS AN INDEPENDENT SOURCE OF NAVIGATION TO CONFIRM ONBOARD DETERMINATION OF CORRECTIVE MANEUVERS
- ONBOARD GUIDANCE FAILURE: MSFN UPDATE OF AGS OBJECTIVES FOR TRANSFER AND MIDCOURSES
- STANDARD PLAN OF ACTION FOR NOMINAL AND ALL NORMAL ABORT CASES
- ΔV PAYLOAD SAVING FOR A SUBSTANTIAL LAUNCH WINDOW

Fig. 3

NASA-S-66-6574 JUN

PRELAUNCH

- CSM PLANE CHANGE
- UPDATE OF CSM ORBITAL ELEMENTS INTO LGC
- CALCULATE LIFT-OFF TIME
- ALIGN IMU
- REFERENCE AGS ATTITUDE MEASUREMENTS TO NEW IMU ALIGNMENT
- UPDATE LM AND CSM STATE VECTORS
- RE-SET AGS TIME DATUM

Fig. 4

NASA-S-66-6573 JUN

LAUNCH

MONITORING

- ALTITUDE ERRORS - COMMAND AND RESPONSE (COMPARE PGNCS AND AGS) ON 8 BALL DISPLAYS
- COCKPIT DISPLAYS: h, $\dot{h}$, V_{CR}
- KEYBOARD DISPLAYS: h_P, V_I, V_{ACC}
- OUT-OF-WINDOW TRACK MONITORING
- RENDEZVOUS RADAR
- MSFN $\dot{r}$ AND $\dot{r}_{CR}$

Fig. 5

NASA-S-66-6559 JUN

LAUNCH (CONCLUDED)

TARGETING

- PGNCS AND AGS EQUIVALENT
- YAW STEER INTO CSM PLANE
- $|V|$ CUTOFF
- γ CONSTRAINED TO ZERO
- CUTOFF ALTITUDE

Fig. 6

NASA-S-66-6529 JUN

CONCENTRIC FLIGHT PLAN

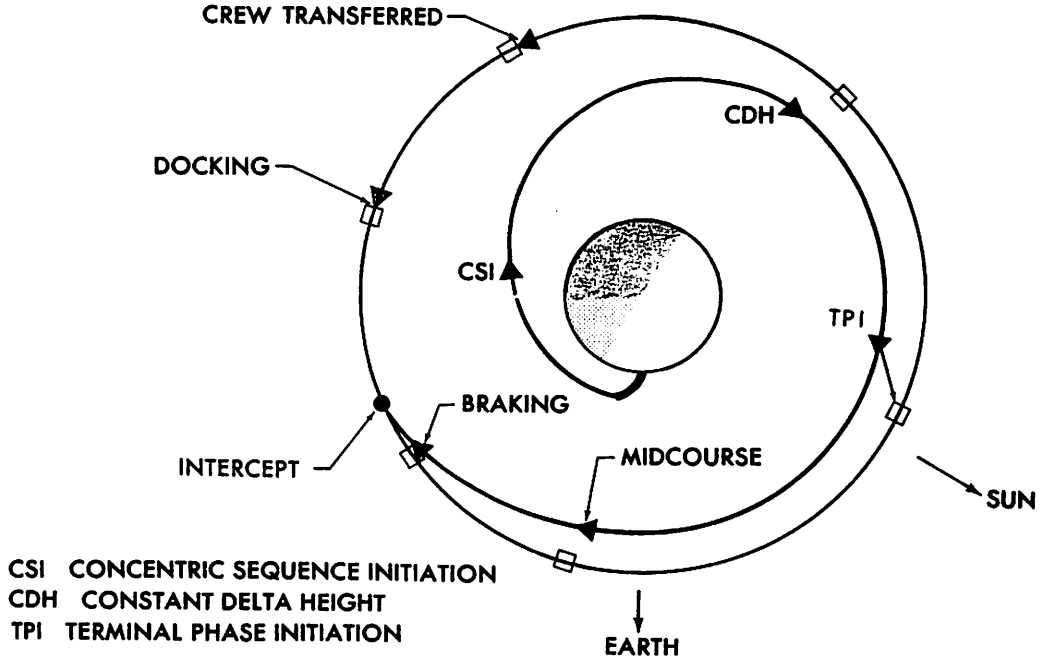


Fig. 7

NASA-S-66-6564 JUN

CONCENTRIC FLIGHT PLAN IN RELATIVE COORDINATES

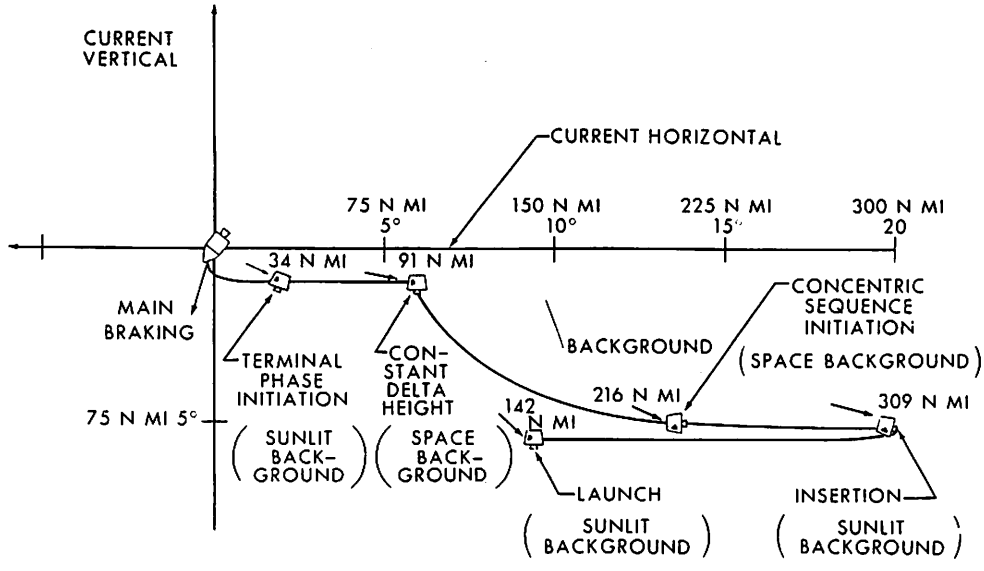


Fig. 8

NASA-S-66-6536 JUN

UTILIZATION OF PROPULSION SYSTEMS

● LM			
• CONCENTRIC FLIGHT PLAN POWERED ASCENT	MAIN ENGINE RCS	MAIN ENGINE FOR POWER RCS NULLS THRUST OFFSET	APS FUEL
• CONCENTRIC SEQUENCE INITIATION	RCS	RCS TANKS FOR ULLAGE THEN APS FUEL	
• CONSTANT ΔH	RCS	RCS TANKS	
• TERMINAL PHASE INITIATION	RCS	RCS TANKS	
• TERMINAL THRUSTINGS	RCS	RCS TANKS	
● CSM			
• ALL MANEUVERS		RCS ULLAGE THEN MAIN ENGINE	
V_G > 12 FT/SEC			
V_G < 12 FT/SEC		RCS e g FINAL TERMINAL THRUSTING IS COMPLETELY CSM ACTIVE RENDEZVOUS	

Fig. 9

NASA-S-66-6521 JUN

TIME EMERGENCY RENDEZVOUS

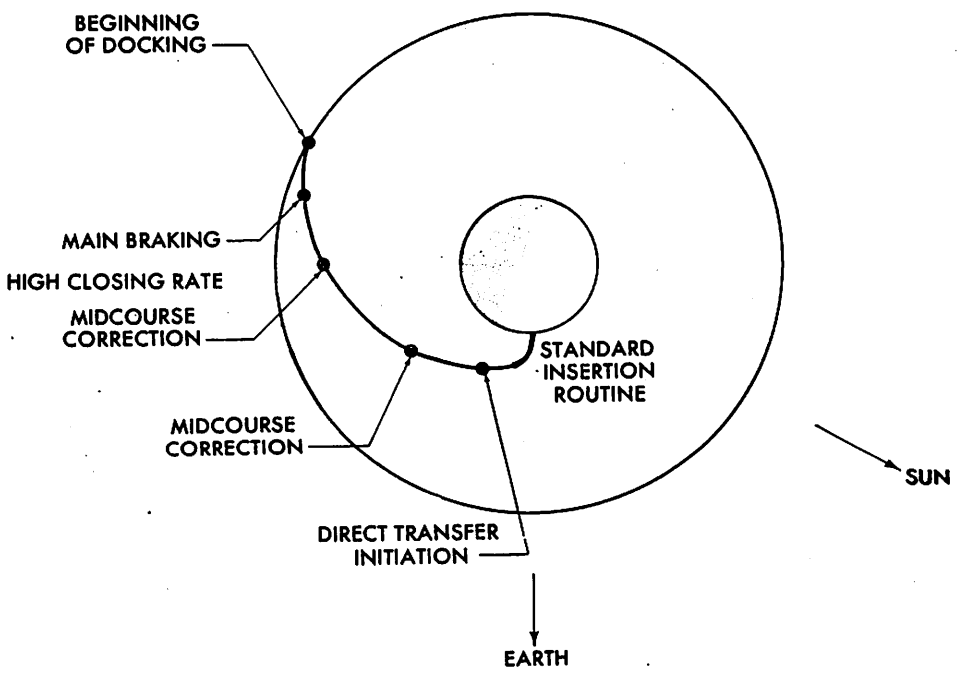


Fig. 10

NASA-S-66-6546 JUN

OPTICAL TRACKING ASSUMPTIONS

	ACCEPTABLE CSM SEXTANT VISUAL CONDITIONS		
LOS HAS RADIAL COMPONENT OUTWARDS	ACCEPTABLE IF ANGLE BETWEEN SUN DIRECTION AND LOS IS GREATER THAN 20°		
LOS HAS RADIAL COMPONENT INWARDS		LUNAR BACKGROUND	
	RANGE (N MI)	TARGET	LUNAR SURFACE LIGHT
	250	FLASHING LIGHT	EARTH REFLECTED
	40	SUN ILLUMINATED LM	SUNLIT
	500 N MI	FLASHING LIGHT	DARK

Fig. 11

NASA-S-66-6567 JUN

EXPLANATION OF LINE-OF-SIGHT CONVENTION

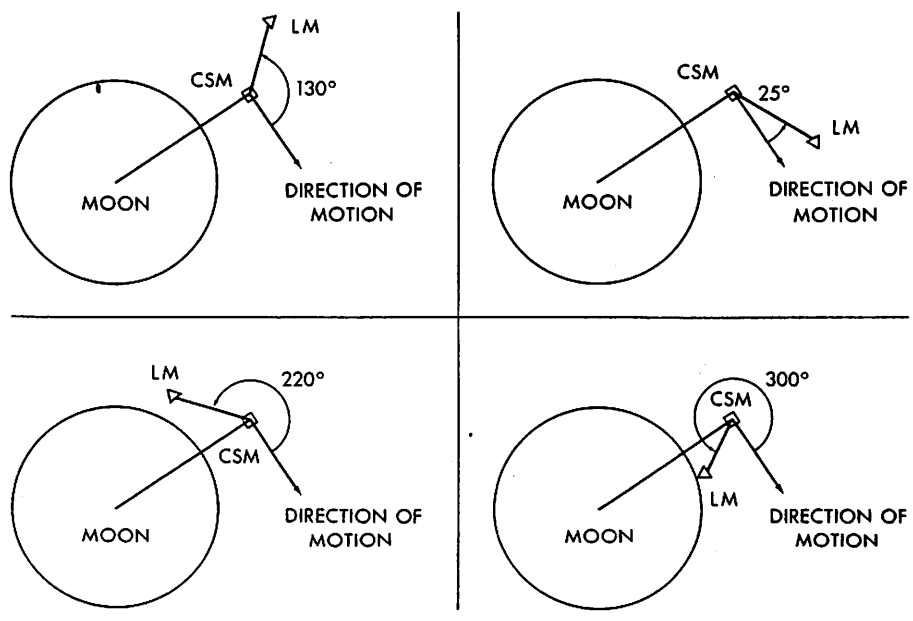


Fig. 12

NASA-S-66-6121 JUN

MPAD 149

CSM SEXTANT TRACKING LEM

NOMINAL FLIGHT PLAN

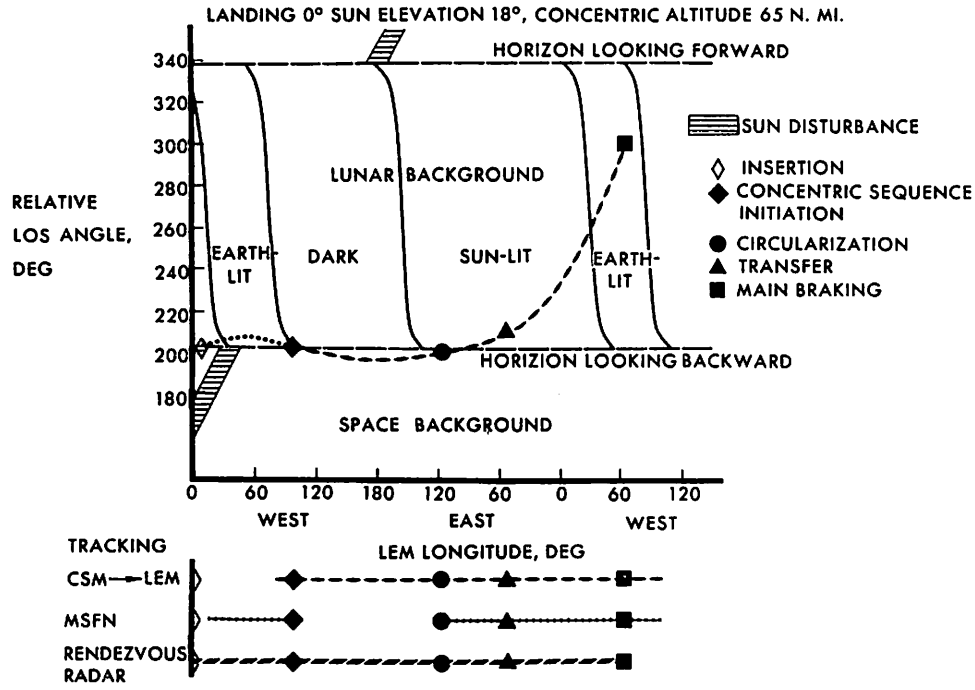


Fig. 13

NASA S 66 6124 JUN

CSM SEXTANT TRACKING LEM

ABORT FROM INITIATION OF POWERED DESCENT

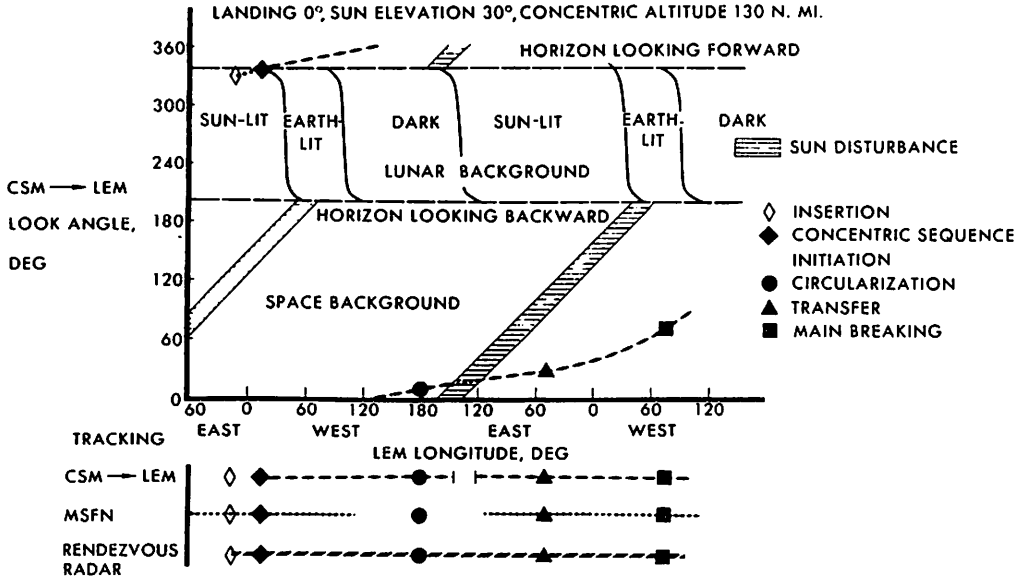


Fig. 14

NASA-S-66-6115 JUN

MPAD 100

TRACKING SUMMARY

RENDEZVOUS RADAR, CSM SEXTANT, MSFN

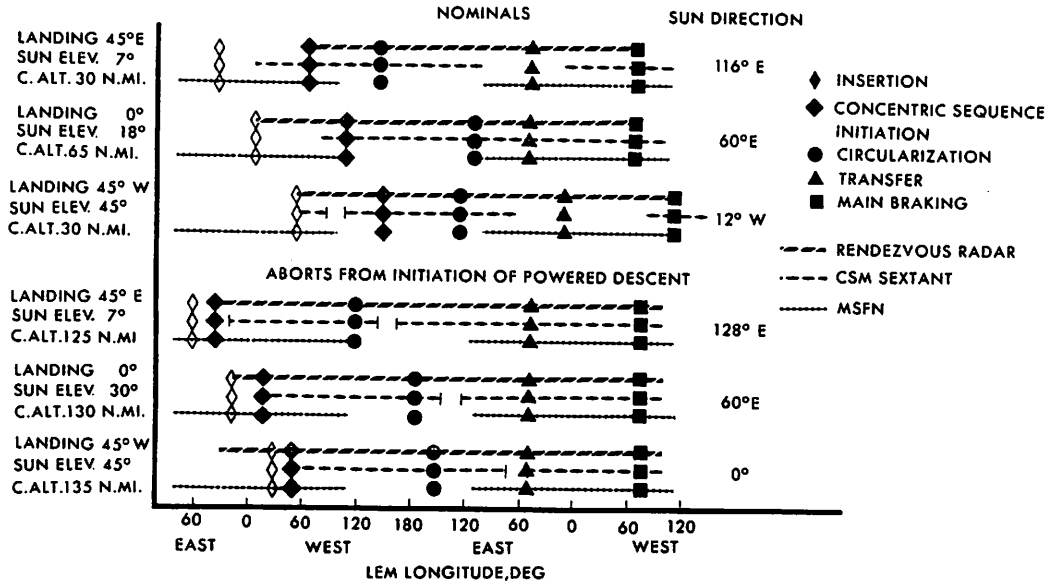


Fig. 15

NASA-S-66-6549 JUN

MSFN ERROR ANALYSIS ASSUMPTIONS

- MSFN TRACKING IN 3 - WAY DOPPLER MODE WITH 1 MASTER STATION AND 2 SLAVE STATIONS
- NOISE AND BIASES ON MEASUREMENTS:

	NOISE (1σ)	BIAS (1σ)
2 - WAY DOPPLER	0.1 FPS	0.07 FPS
3 - WAY DOPPLER	0.1 FPS	0.2 FPS

- STATION LOCATION BIASES AS GIVEN IN APOLLO NAVIGATION WORKING GROUP TECHNICAL REPORT NO. 65 - AN - 1.0, FEB. 5, 1965
- BIAS ON GRAVITATIONAL CONSTANT OF MOON,

$$\sigma_{\mu_M} = 6 \times 10^9 \text{ FT}^3/\text{SEC}^2,$$

- FROM JPL TECHNICAL REPORT NO. 32 - 694, DEC. 15, 1964
- ORBIT DETERMINATION SOLVES FOR
  - 3 COMPONENTS OF VELOCITY
  - 3 COMPONENTS OF POSITION
  - BIAS ON 3 - WAY DOPPLER

Fig. 16

NASA-S-66-6563 JUN

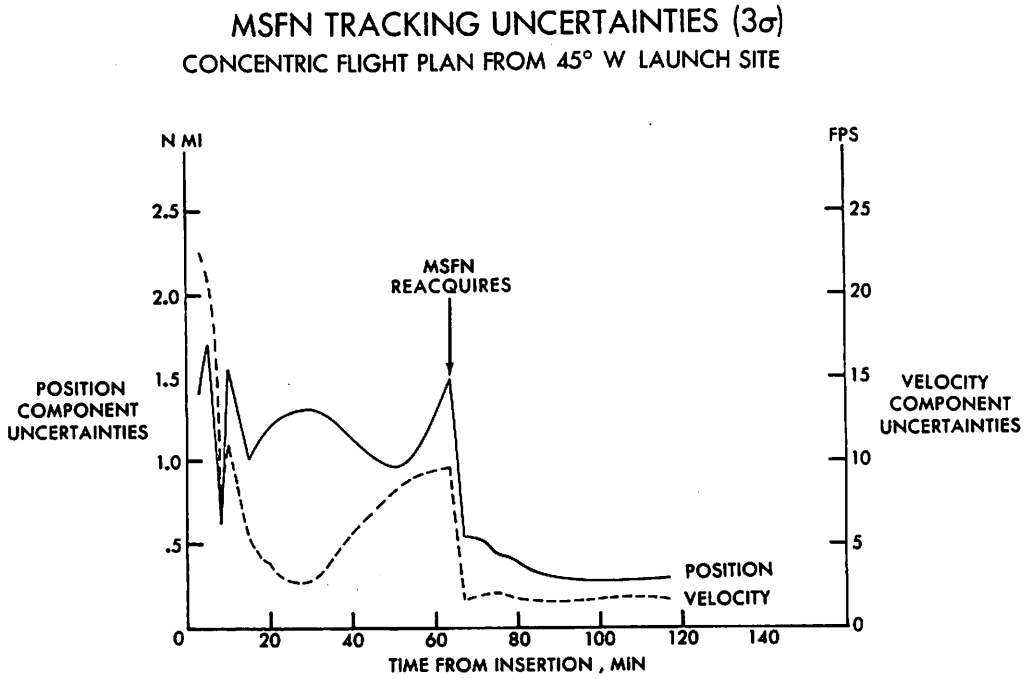


Fig. 17

NASA-S-66-6541 JUN

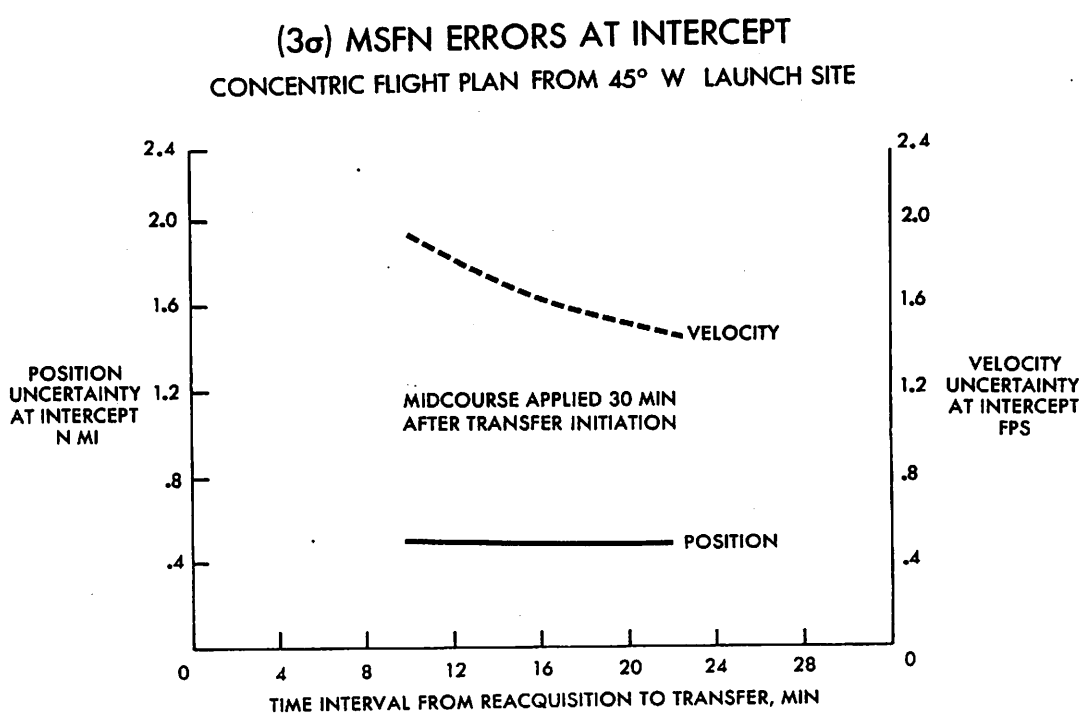


Fig. 18

NASA-S-66-6554 JUN

MAIN COMPONENTS OF ONBOARD RENDEZVOUS PROGRAMS

PARALLEL DETERMINATION ON GROUND

- STANDARD INSERTION
  - LAUNCH; ABORT FROM POWERED DESCENT
- DIRECT TRANSFER
- CONCENTRIC SEQUENCE
  - CSI ⟶ CDH ⟶ TPI ⟶ TERMINAL MANEUVER
  - DETERMINED AS A WHOLE BUT CREWMAN GIVES PROCEED THROUGH DSKY FOR EACH MANEUVER
- EXTERNAL ΔV

Fig. 19

NASA-S-66-6571 JUN

PROVISION OF G AND N INFORMATION FOR THE OPERATION

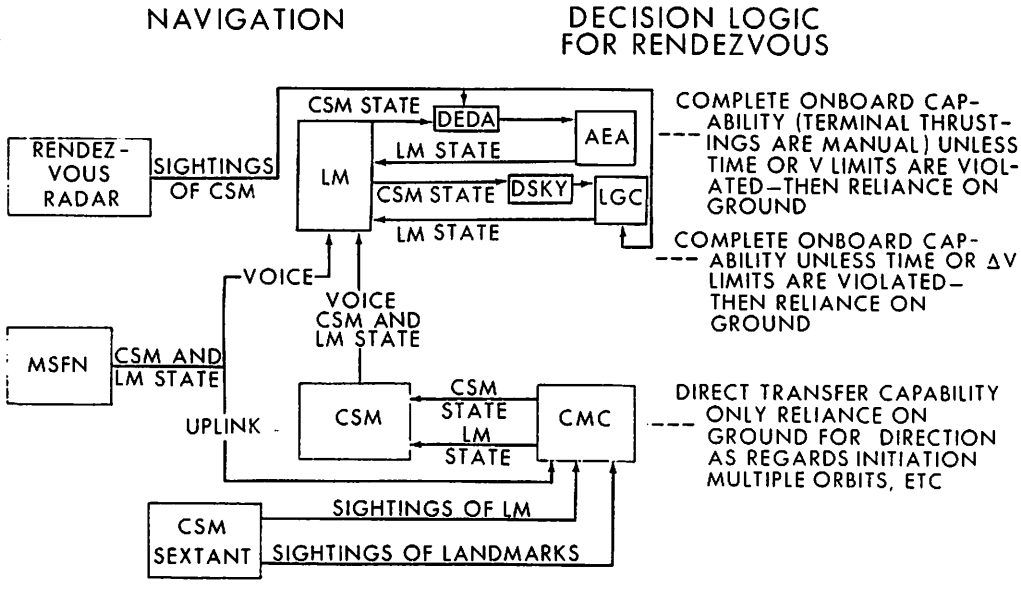
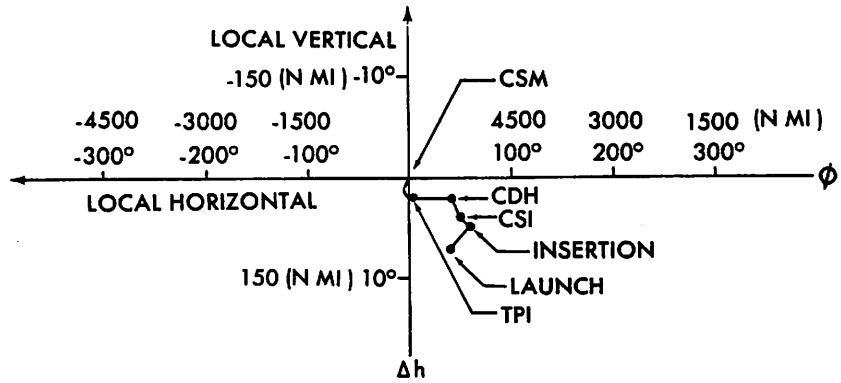


Fig. 20

NASA-S-66-6578 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (60°)



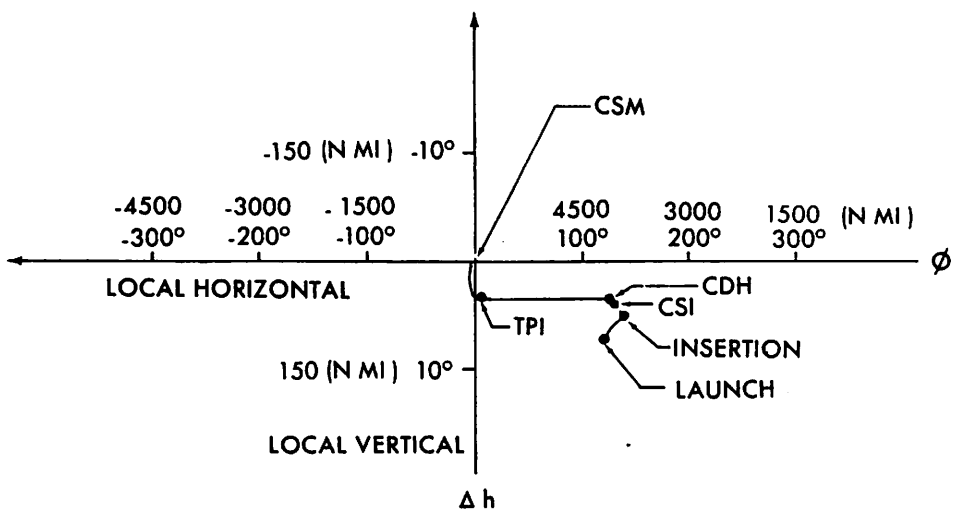
RELATIVE FLIGHT PLAN

Fig. 21

NASA-S-66-6577 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (140°)



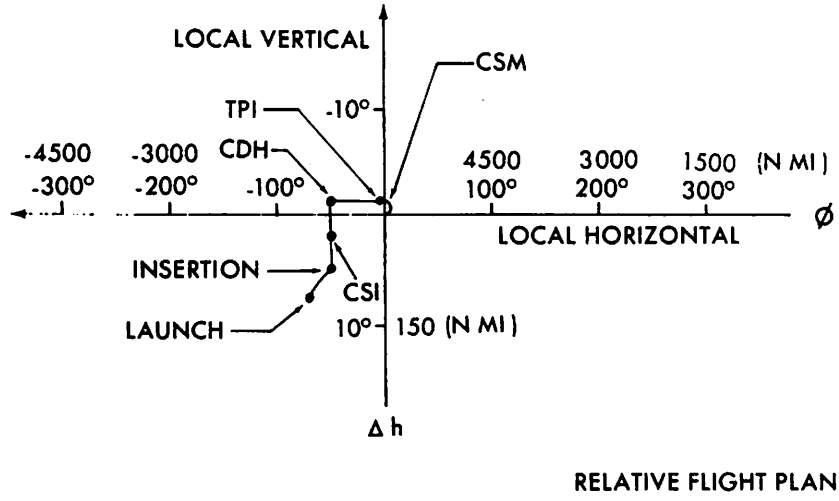
RELATIVE FLIGHT PLAN

Fig. 22

NASA-S-66-6557 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (-50°)



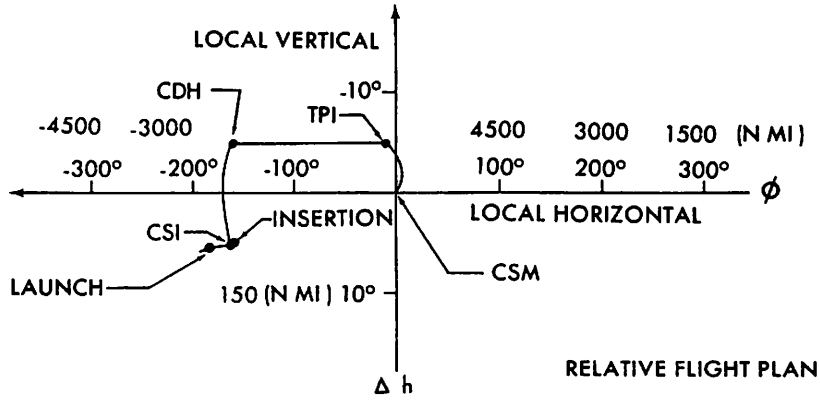
RELATIVE FLIGHT PLAN

Fig. 23

NASA-S-66-6576 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (-160°)

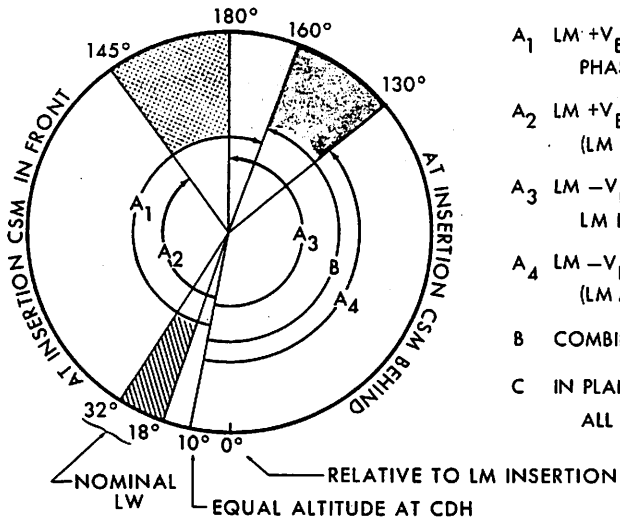


RELATIVE FLIGHT PLAN

Fig. 24

NASA-S-66-6569 JUN

RENDEZVOUS CATEGORIES
ACCORDING TO PHASING



A_1 LM $+V_E$ CATCHUP WITH TRANSFER FROM PHASING ALTITUDE (LM LIFETIME ONLY)

A_2 LM $+V_E$ CATCHUP AND ΔH = 15 N MI (LM LIFETIME ONLY 11.3 HR)

A_3 LM $-V_E$ CATCHUP USING TOTAL ΔV AND LM LIFETIME (INPLANE)

A_4 LM $-V_E$ CATCHUP 0.5° WEDGE ANGLE (LM ΔV TOTAL AND LIFETIME)

B COMBINED MANEUVERS WITH CSM

C IN PLANE CSM RESCUE POSSIBLE FROM ALL PHASE ANGLES

Fig. 25

NASA-S-66-6562 JUN

RENDEZVOUS LOGIC

CONSTRAINT:
- ASCENT STAGE LIFETIME
- LM ΔV AVAILABLE
- CLEAR PERICYNTHION TRANSFERS FOR BOTH LM AND CSM

DESIRABLE CONDITION:
- SATISFACTORY LIGHTING DURING TERMINAL PHASE

A) LM ACTIVE

1ST PREFERENCE
- (1) FIRST ORBIT
  - 1.1 NON TIME CRITICAL; CONCENTRIC
  - 1.2 TIME CRITICAL EQUIVALENT DIRECT ASCENT
- (2) MULTIPLE ORBITS

TERMINAL THRUSTINGS BY LM

B) COMBINED MANEUVERS

2ND PREFERENCE
- MINIMUM AMOUNT OF ENERGY EXPENDED BY CSM, MULTIPLE ORBITS BY LM IF NECESSARY

TERMINAL THRUSTINGS BY LM

C) CSM ACTIVE ONLY

3RD PREFERENCE
- (MAXIMUM CAPABILITY OF LM PROPULSION) WAS TO INSERT IN MINIMUM SAFE ORBIT

TERMINAL THRUSTINGS BY CSM

Fig. 26

NASA-S-66-6545 JUN

SUMMARY

- PLANS FOR
  - EVERY PHASE; ALL PRACTICAL OUT OF PLANE CONDITIONS
- NORMAL CASES
  - CONCENTRIC- STANDARD PATTERN OF PROCEDURES
- PLAN FOR TIME CRITICAL SITUATIONS
- REDUNDANCY ON THE LM TO RELIEVE THE CSM
- NON TIME CRITICAL
  - JOINT DECISIONS DURING INTERCEPT TRAJECTORY

Fig. 27

TRANSEARTH INJECTION THROUGH REENTRY

By

M. P. Frank

TRANSEARTH INJECTION THROUGH REENTRY

In this section we will finish off the mission planning considerations and the affects of constraints on trajectory shaping of the nominal mission. All we have to discuss on this topic are the problems of getting the spacecraft back from the moon through the reentry corridor to the recovery area. This section is divided into two phases, transearth and reentry.

Transearth

The geometric restrictions on transearth injection are somewhat similar to those of the translunar injection. However, proper allowances must be made for the fact that the spacecraft trajectory relative to the moon is hyperbolic instead of elliptic and the moon is orbiting about the earth. In the transearth case the target body is relatively stationary and the spacecraft is leaving the body which is in orbit. The spacecraft must break out of the lunar gravitational sphere and fall back to earth. The velocity relative to the moon must be increased in order to escape but the spacecraft inertial velocity must be decreased in order to return to earth.

This is illustrated in Fig. 1. At a distance from the earth equal to the distance to the moon's sphere of influence the earth relative velocity vector required to obtain a given perigee altitude is illustrated by the vector V_{re}. In this figure two cases are shown; a high energy short time trajectory and a low energy long transit time trajectory. There is a continuum of safe return trajectories between there two extremes. The moon's orbital velocity is shown as the vector V_m. The spacecraft must obtain a velocity vector relative to the moon V_{rm} that results in a velocity relative to the earth V_{re}. The minimum velocity relative to the moon occurs when the velocity relative to the earth is a minimum or at apogee of the earth return trajectory. This is not necessarily the longest flight time because the spacecraft could leave the moon on such a trajectory as to have a positive flight-path angle relative to the earth. This type of trajectory is of no interest, however, because the increase of transearth flight time does not allow a decrease in the injection velocity requirements. The conclusions that can be drawn from this figure are that faster return times require larger exit velocities and hence larger transearth injection velocities. It can also be seen that the lower injection velocities require a more nearly retrograde motion upon leaving the moon's sphere of influence.

Figure 2 illustrates the trajectory geometry inside the moon - sphere of influence relative to the exit conditions. In this figure the moon's sphere of influence is approximated by the large diameter circle. The velocity of the moon relative to the earth is given by the vector V_m. Since the sphere of influence moves with the moon any point on this sphere would also have the velocity relative to the earth of V_m. The trajectories inside this sphere are hyperbolic relative to the moon and the energy of these trajectories determines the magnitude of the exit velocity V_{rm}. The direction of this exit velocity is primarily controlled by the exit position which in turn is determined by the longitude at which the injection is performed. To obtain a more retrograde direction of the exit velocity, the injection must occur around on the leading edge of the moon as illustrated by the upper trajectory in Figure 2 To obtain a high energy, short time, earth return trajectory the injection maneuver must be performed on the trailing edge of the moon. This results in exit velocities directed more toward the earth.

It can be seen that once a desired transearth flight time has been selected the exit velocity requirement can be determined and in turn these exit velocity requirements will specify the energy of the transearth injection maneuver and also the location of the transearth injection maneuver. The shaded area in Figure 3 shows the region of longitudes where the transearth injection could be performed. It extends from about 140 degrees west longitude to about 140 degrees east longitude on the far side of the moon.

The affect of the transearth transit time on the injection velocity requirements can be obtained from Figure 4. In this figure the transearth injection ΔV is shown as a function of transit time for two different moon positions in its orbit. The trend of generally decreasing injection velocity requirements with increasing transit time is clearly illustrated. The fact that these curves cross over indicates that there are other factors at work in determining the actual injection velocity required. However, these other factors do not change the basic conclusions.

Now, lets consider how the required transearth flight time is determined. Figure 5 illustrates the transearth trajectory drawn in the moon orbit plane. The return perigee location is shown relative to the moon's antipode. In this case the moon's antipode is drawn at the time of the transearth injection. This relative angle between the return perigee and the antipode has very little variation with the return transit time. Therefore, the approximate inertial position of the return perigee is a function only of the moon's position. The landing location relative to the return perigee also has very little variation so that in affect the inertial position of the landing is known well in advance of the actual transearth injection.

The recovery forces, of course, are fixed to the earth and hence are rotating around with the earth. This leads to the rather interesting situation in which the inertial location of the landing position is well known. However, the inertial position of the recovery forces is highly dependent on the time at which landing occurs. The determination of the transearth transit time is based on the location of the recovery forces. That is, the reentry and landing must occur when the recovery forces are in the proper position. This occurs once every 24 hours, and within any 24 hour range of return transit times one time can be found which allows rendezvous with the recovery forces.

Entry Phase

For the lunar mission the entry corridor is defined by the variation in flight-path angle at the entry interface altitude of 400,000 feet. Strictly speaking, this allowable variation is a function of the velocity at entry, but there is such a small variation for a nominal lunar mission that the velocity effect is generally omitted. However, for abort returns to earth, this is not the case, and the entry corridor is defined as a function of entry velocity.

The maximum entry angle is defined by the maximum allowable aerodynamic decelleration. Aerodynamic loads encountered during entry increase rapidly with increasingly negative flight-path angles. The high-"g" side of the corridor is called the undershoot boundary. The minimum entry angle, called the overshoot boundary, is defined by the Command Module's capability to prevent an uncontrolled skipout of the atmosphere. In addition to being a function of entry velocity, these corridor boundaries are strongly dependent of the L/D ratio of the entry vehicle.

Figure 6 shows the entry corridor for the nominal L/D of .34 of the Command Module. In this figure, entry flight-path angle is plotted as a function of range from the entry point to the landing point in order to combine the corridor and maneuvering capability information. The skipout limit is shown here to be a flight-path angle slightly less than 5 degrees. In other words, any entry conditions at shallower angles than this would result in an uncontrolled skip. The 10 g undershoot limit is shown to be about 7.3 degrees and any flight-path angle steeper than this would result in the g loads exceeding 10. The emergency limit is defined as a limit below which the aerodynamic load factor would exceed 20 g. This 20 represents the structural design limit of the spacecraft.

The minimum guided range represents the limit to which the reentry range can be controlled with a guidance system. For the overshoot boundary this corresponds to about 1500 miles. The maximum range is limited by the lifting capability of the vehicle. At the steep entry angles the spacecraft could not fly further than this range with maximum positive lift. The maximum range is arbitrarily cut off at 2500 n.m. to comply with the limitations

imposed by the heat shield. The heat shield is designed to tolerate a 3500 n.m. reentry. There is an entry monitoring system onboard the spacecraft which gives warning when excessive skip is to be encountered. However, the tolerance on this entry monitoring system is about 1000 miles. This is made large so that it will not unnecessarily take over a trajectory or give warning that an excessive skip is going to occur. To allow for this 1000 mile tolerance the mission can not be planned with reentry ranges exceeding 2500 miles. The inplane maneuvering capability that can be used which would be independent of the entry ocrridor position is given by the 1500-mile limit and the 2500-mile limit so that ± 500 miles of down range maneuver capability is available. The nominal aim point represents the conditions that will be targeted to for the transearth trajectory. That is, the return trajectory will be planned to have a flight-path angle at entry of 6.2 degrees and to have the entry point located some 2000 miles away from the landing point. This gives maximum maneuvering capability and allows maximum tolerance of dispersions both in flight-path angle and in range at entry. The primary purpose of the maneuvering capability is to allow a change in the landing site after the transearth injection has been performed. If bad weather were to develop in the area of the recovery forces such that a landing there was undesirable, the spacecraft would have the capability of going 500 n.m. to either side of this position. Figure 7 shows the total maneuver footprint, both the downrange and crossrange plotted on a map of the Pacific Ocean hemisphere. The reentry point is some 2000 miles away from the nominal touchdown position. The crossrange capability is about 440 miles at the base and some 660 miles at the toe of this footprint.

Now let's examine the considerations involved in locating the recovery areas. Figure 8 shows the inplane geometry of the location of significant points of the reentry trajectory. The angular relations shown between the entry point, the perigee of the return trajectory and the landing point have only a slight variation. The location of the return perigee relative to the antipode also has only a slight variation. The result is that the landing point on the earth's surface will be very near the moon's antipode. All of the earth return trajectories will pass through this antipode regardless of the return inclination. Because of the relatively small angle between the antipode and the landing position, very little latitudinal control of the landing can be obtain by varying the inclination of the earth return trajectory. This is illustrated in Figure 9. In this figure, two typical earth return trajectories are shown. These two inclinations would be obtained by performing pland changes or performing an azimuth change at the transearth injection maneuver. This plan results in a different return inclination for the two trajectories. However, since the antipode is in

effect a node through which all of these return trajectories pass and since the landing is fairly close to this node, it can be seen in Figure 9 that only a very small amount of latitude variation can be obtained. The amount of latitude control is further restricted by the fact that the return inclination is limited to 40 degrees or less relative to the earth's equator. The landing position control that is available then can be summarized as one in which very fine control is available for the landing latitude. The longitude can be controlled exactly by merely varying the transit time of the transearth trajectory. However, the latitude will be a function of the moon's declination at the time of the transearth injection maneuver, and only small variations about this latitude are available through the means of changing the return trajectory inclination. These landing point control characteristics have led to the definition of the recovery zones shown in Figure 10.

There are two zones in which recovery from the lunar landing mission may occur. The northern zone extends from a latitude of 35 degrees north to 35 degrees south along a longitude of approximately 160 degrees west. The southern zone extends from five degrees south latitude to 35 degrees south along a longitude of approximately 167 degrees west. The northern zone would be supported by recovery forces staged from Hawaii. The staging base for the southern recovery zone would be Pago Pago in the Samoa Islands. The northern zone extends into the south latitude because of the preference to stage the recovery forces from the Hawaiian vase. The shape of these recovery zones reflects the landing area control capability of the return trajectory. The zones need not extend over a large range of longitudes because the longitude can be controlled precisely by the variation in return time. They do extend over a wide range of longitudes because the longitude can be controlled precisely by the variation in return time. They do extend over a wide range in latitudes since very little latitude control of the landing point is available. The latitude range of these recovery areas is a function of the maximum northern and southern declinations of the moon during any given month.

For any specific mission, of course, it will not be necessary to deploy forces to cover the entire recovery zones. The latitude of the antipode, or the moon's declination, will be confined to a narrow region of latitudes so that the recovery forces for any specific mission will be confined to a narrow region in one of these recovery zones.

Questions and Answers

TRANSEARTH INJECTION THROUGH ENTRY

Speaker: M. P. Frank

1. Dr. Von Braun - Can the same recovery ships be used throughout the launch window?

   ANSWER - Yes. The recovery ships are capable of achieving 30 knots, which is more than adequate to keep up with the recovery area changes.

2. Mr. Green - What is the longest time practical between getting back into lunar orbit and transearth injection?

   ANSWER - The CSM will have sufficient expendables aboard for approximately 14 days. Therefore, the longest time after LEM separation would be approximately 7 days. The trajectory changes required to return to earth during this period are not significant.

NASA-S-66-6465 JUN

EFFECT OF LUNAR SPHERE EXIT VELOCITY ON RETURN TRAJECTORIES

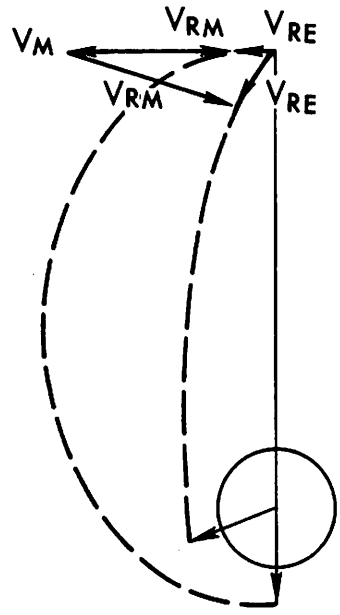


Figure 1

NASA-S-66-6413 JUN

LUNAR SPHERE EXIT HYPERBOLAS THAT RETURN TO EARTH

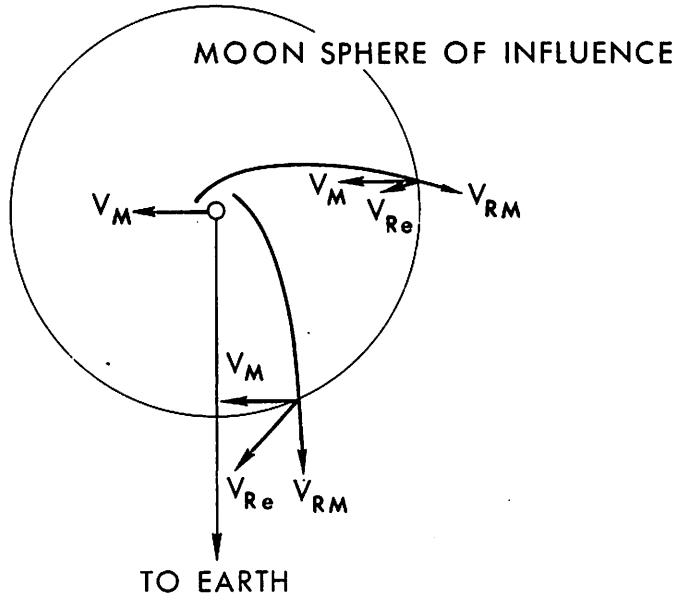


Figure 2

NASA-S-66-5019 MAY 31

EFFECT OF TRANSEARTH FLIGHT TIME ON INJECTION POSITION

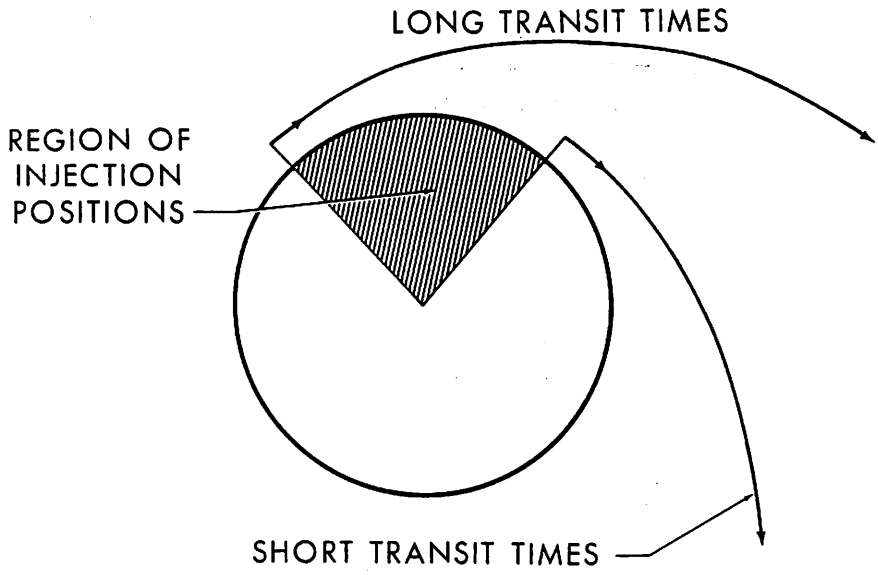


Figure 3

NASA-S-66-5020 MAY 31

EFFECT OF TRANSEARTH FLIGHT TIME ON INJECTION VELOCITY REQUIREMENTS

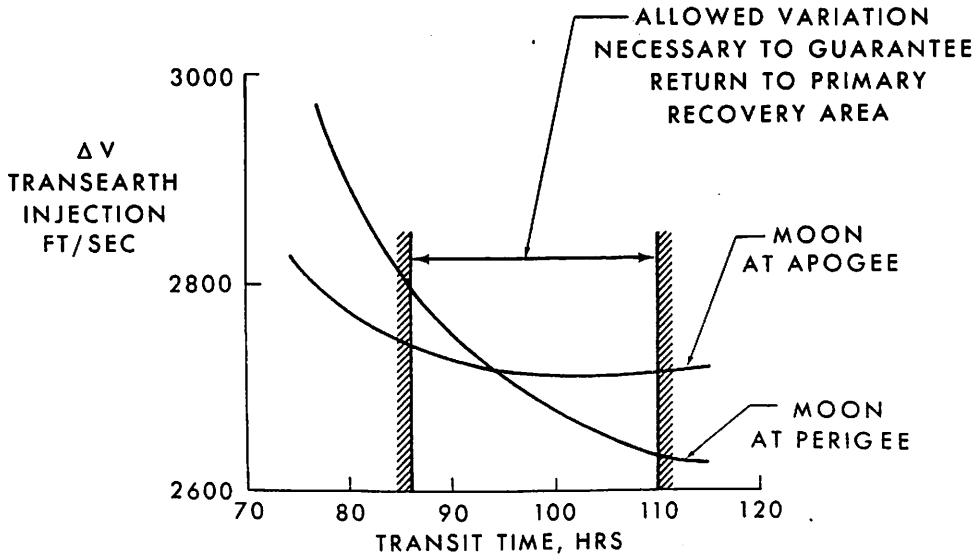


Figure 4

NASA-S-66-6051-JUN

TRANSEARTH TRAJECTORY PERIGEE LOCATION

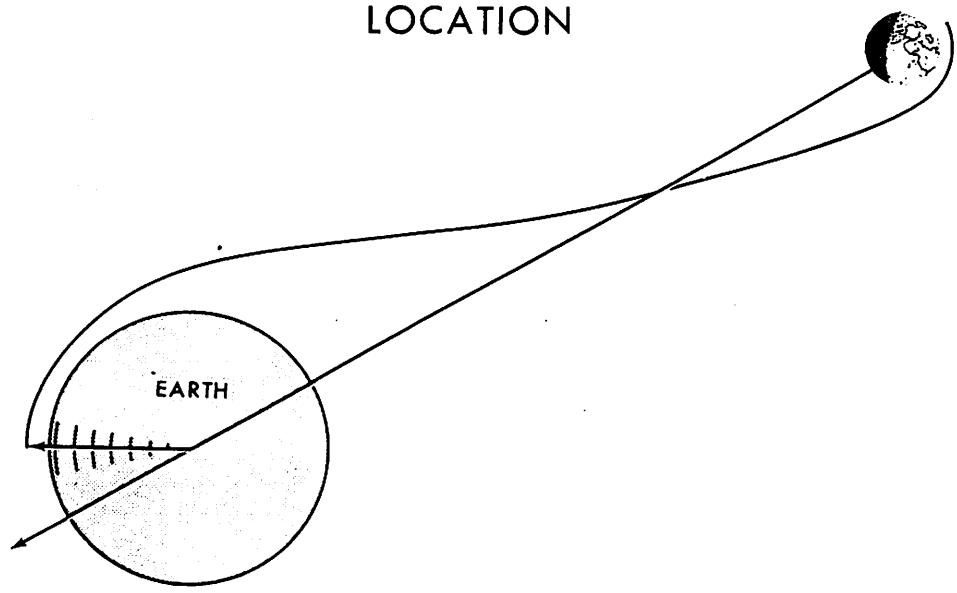


Figure 5

NASA-S-66-5009 MAY 31

ENTRY CORRIDOR

ENTRY SPEED = 36,000 FPS

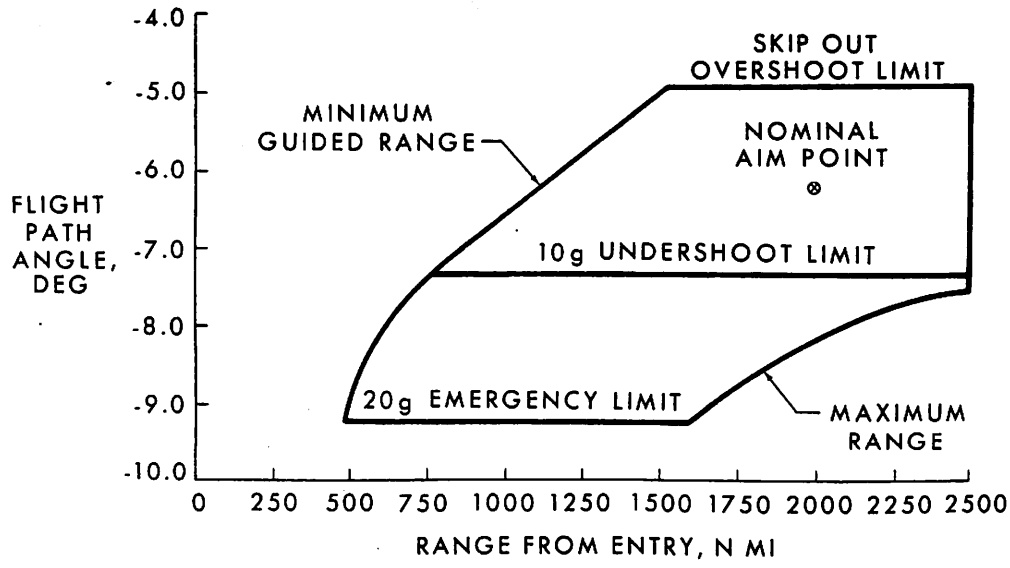


Figure 6

NASA-S-66-6041 MAY

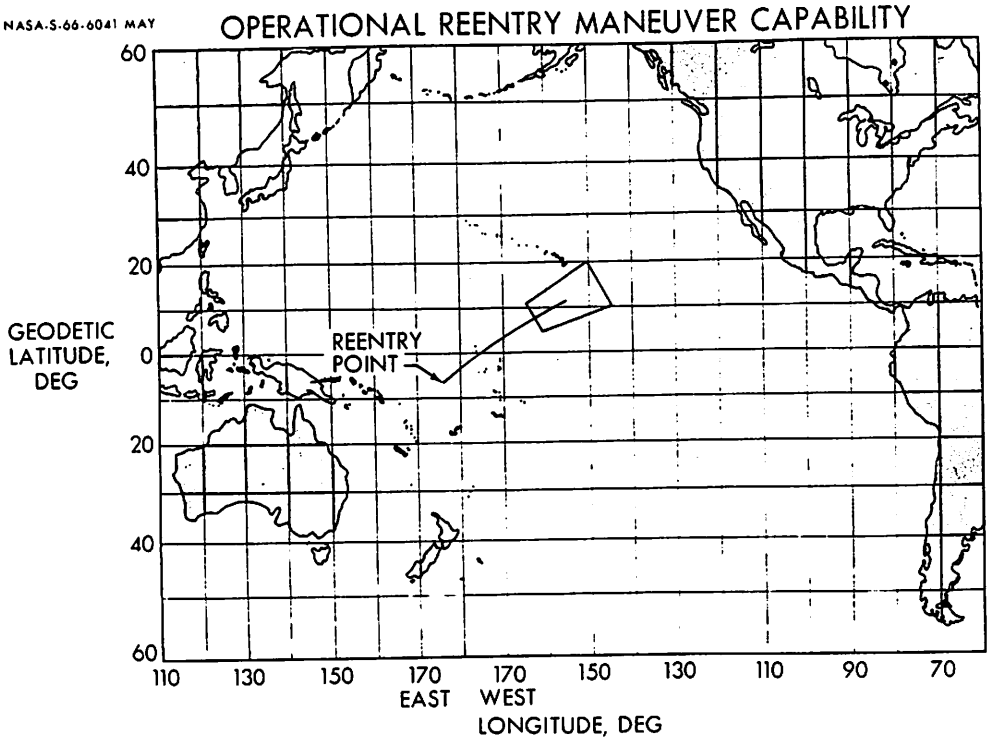


Figure 7

NASA-S-66-6464 JUN

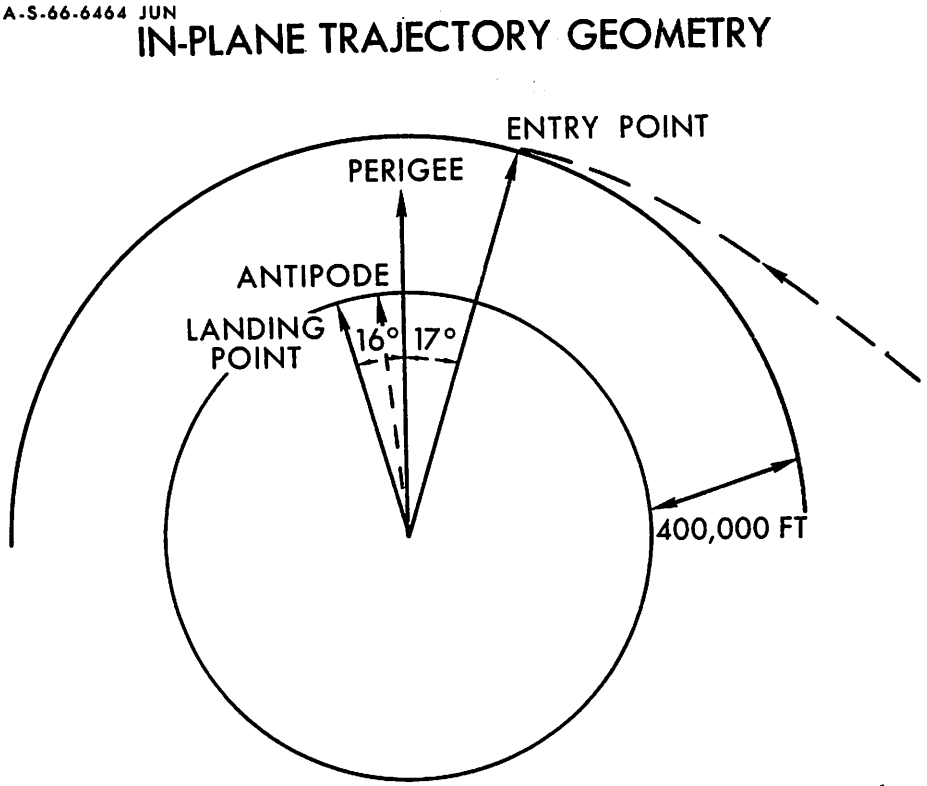


Figure 8

NASA-S-66-6392 JUN

EFFECT OF PLANE CHANGE ON CHANGING THE LANDING POINT

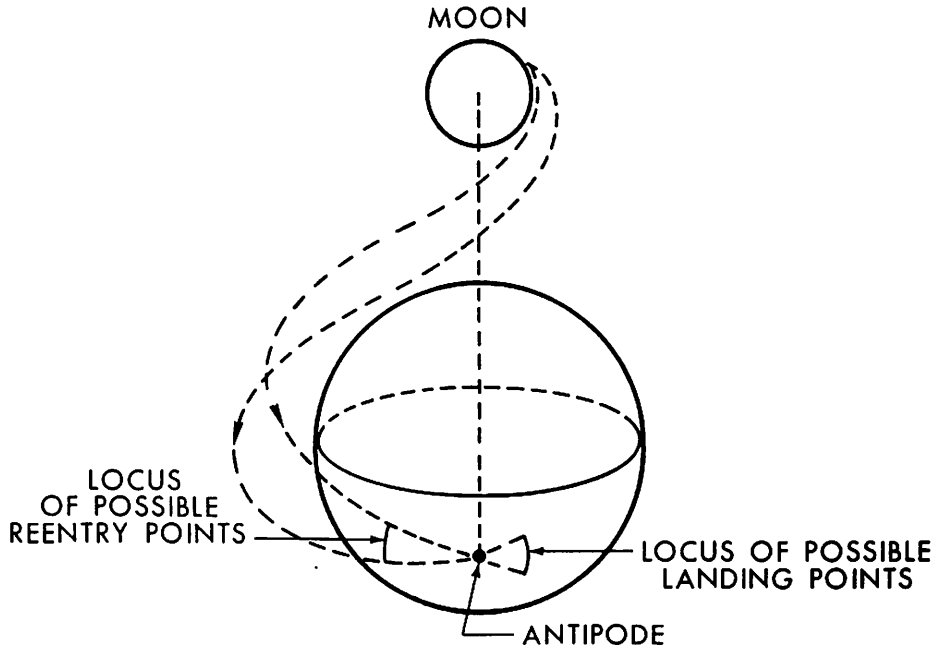


Figure 9

NASA-S-66-5018 MAY 31

NOMINAL RECOVERY AREAS FOR MISSION PLANNING

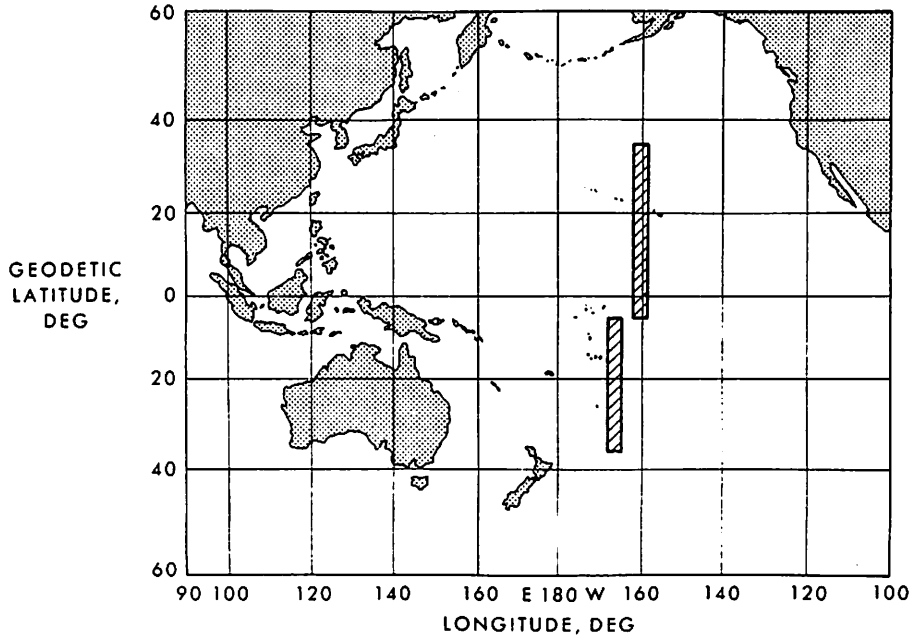


Figure 10

APOLLO EARTH RETURN ABORT CAPABILITIES

By

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APOLLO EARTH RETURN ABORT CAPABILITIES

I.0 INTRODUCTION

It has always been a stated desire, as regards Project Apollo, to have continuous abort-to-earth capability throughout the entire lunar landing mission. This paper will examine the capability of the spacecraft to satisfy this objective through each mission phase of the lunar landing mission. The abort capability will be discussed primarily from a performance standpoint. In other words, examining whether or not the spacecraft has the necessary performance required for continuous abort capability throughout the mission. Portions of the mission where redundancy exists will be pointed out as well as portions of the mission which are critical or marginal as regards abort capability. The primary ground rules for this discussion are as follows:

a. The only objective considered as regards earth return aborts is the safe return of the crew. No alternate mission objectives are considered.

b. Only spacecraft abort capability after an abort decision has been made will be discussed. In other words, an assessment of the spacecraft capability to recognize an abort situation will not be included.

c. Only one, or at the most two, burn abort maneuvers will be considered since more sophisticated multiple burn maneuver sequences are not required to provide adequate abort capability.

The items which will be discussed for each mission phase are as follows:

a. The characteristics of the trajectories from which aborts could be required will be described. This includes any trajectory which might result from an underburn or an overburn during any of the major power flight maneuvers.

b. The basic abort modes or procedures will also be briefly described. Where possible, the mode considered prime will be pointed out along with the modes which are optional or backup. The computer logic required to provide all of these abort modes and procedures are currently being implemented into the real time ground computer system.

c. The capability of the spacecraft of actually performing abort manuevers will be discussed. Included here will

be a description of the propulsion systems available, the ΔV available from each, and the ΔV required for abort.

d. The significant characteristics of abort trajectories will also be discussed. Included will be such things as orientation with respect to the earth and moon, relation between return time ΔV required, and delay time, etc.

The major constraints which shape all abort trajectories are as follows:

a. Reentry corridor - In other words, a velocity/flight-path angle relation at the beginning of the atmosphere (400,000 feet). For the purposes of this paper, only abort trajectories targeted to the center of the reentry corridor will be considered.

b. Maximum reentry speed (less than 36,000 fps) - This constraint is due to heat shield limitations.

c. Return inclination (less than 40°) - This constraint assures landings in temperate zones even in the presence of large reentry dispersions and also reduces heat shield requirements.

d. ΔV available - In other words, the ΔV required for an abort must be within the ΔV capability of the spacecraft. This value will probably be set at some amount less than the total ΔV available to allow a pad for midcourse and possible contingencies.

e. Return time - This constraint must always be less than the lifetime of the spacecraft systems such as power supply, life support, etc.

The mission phases considered in this discussion will be (1) launch-to-earth parking orbit, (2) earth parking orbit coast, (3) translunar injection, (4) translunar coast, (5) lunar orbit insertion, (6) lunar orbit coast, (7) transearth injection, and (8) transearth coast. Note that aborts during LM maneuvers are not considered since the abort would be to rendezvous.

2.0 ABORTS DURING LAUNCH PHASE

This phase consists of the launch vehicle three-stage burn from the pad-to-earth parking orbit insertion, a burn duration of approximately 700 seconds. An abort during this phase requires immediate action since the spacecraft trajectory is suborbital for the majority of the burn.

The propulsion systems available for abort during the launch phase are as follows:

a. LES - Launch escape system propulsion systems mounted on the tower atop the spacecraft.

b. SPS - The service propulsion system of the CSM.

c. S-IVB - The third stage of the launch vehicle.

Figure 1 shows a summary of the abort modes for the launch phase and through what region of the launch burn they apply. These modes are shown as a function of the burn time as well as the respective launch vehicle stage. The black bar designates the prime mode while the striped bar represents the optional or backup modes of abort.

Note that the first mode is the LES or launch escape system mode which is prime from the pad throughout the S-I stage and extending on for a few seconds into the S-II burn before LES jettison. As shown in figure 2, a LES abort consists of the LES propulsion system separating the Command Module from the stacked launch vehicle configuration and providing an adequate altitude and downrange translation. This is followed by the orientation of the Command Module with heat shield forward for reentry. Landing occurs in a continuous Atlantic recovery area along the flight azimuth up to a maximum downrange of approximately 400 nautical miles.

The next mode is the suborbital free-fall abort. This mode, as shown in the summary chart, begins where the LES is jettisoned and remains available until approximately halfway through the S-IVB burn. Note that this mode is prime through approximately the first half of the S-II burn and through half of the S-IVB burn. It is considered an optional mode through the second half of the S-II burn because of the availability of a contingency; orbit insertion with the S-IVB, which will be discussed below.

As shown in figure 3, the suborbital free-fall mode consists of CSM separation from the launch vehicle using Service Module RCS, a 10-second SPS burn to gain further separation from the launch vehicle, Service Module jettison, and Command Module orientation (heat shield forward) for reentry. Landing would be in the continuous Atlantic recovery area along the flight azimuth up to 3,200 nautical miles downrange.

An extension of the suborbital abort mode can be achieved by addition of another SPS burn for landing area control, as shown in the summary of abort modes in figure 1. This mode is available as an option during the second half of the S-IVB burn when the suborbital free-fall mode is no longer available.

Figure 4 shows how the suborbital mode with SPS landing control differs from the free-fall mode. Note that the procedure is identical except for an additional retrograde SPS burn. This burn is a variable length depending on the time of abort and causes landing to be at the end of the continuous Atlantic recovery area, approximately 3,200 nautical miles downrange.

The next mode of interest, as shown in figure 1, is the S-IVB contingency orbit insertion followed by an SPS deorbit to reentry. This mode, as shown, is available and prime for approximately the second half of the S-II burn. The reason that this mode is considered prime over the suborbital free-fall mode is that it allows landing to be pinpointed precisely to a given recovery force and would allow additional "thinking" time to consider alternate missions.

The next mode is similar in nature to the S-IVB contingency orbit insertion, except the SPS is used for both the COI and the deorbit burn. This mode, as shown in figure 1, is only possible during approximately the latter half of the S-IVB burn, but it is the prime mode for this time period.

Figure 5 shows the basic features of these latter two abort modes. As shown, insertion into earth parking orbit is completed by either the S-IVB or the SPS. After a certain coast time in earth parking orbit, during which CSM S-IVB separation occurs (if it has not already), an SPS coplanar deorbit burn is performed to return the spacecraft to reentry. The time of deorbit is chosen so as to result in landing at a discrete recovery area, as done in Projects Mercury and Gemini. The SPS deorbit burn is targeted so as to place the earth's horizon at a specified point in the spacecraft window for monitoring purposes.

In summary, then, as regards launch aborts, the main things to remember are: (1) abort capability in one mode or another is available throughout the entire launch phase on a continuous basis; (2) contingency orbit insertion followed by deorbit is always prime when it is available.

3.0 ABORTS FROM EARTH PARKING ORBIT COAST

This mission phase consists of coasting in a circular earth parking orbit from earth orbit insertion to the initiation of translunar injection, a duration which is usually one orbit or longer. As might be expected for this phase, the abort procedures being planned are very similar to those used in Projects Mercury and Gemini.

The propulsion systems available and capable of performing an abort during this phase with their respective ΔV capabilities are shown in Table I. Note that the Service Module RCS propulsion system does not appear on this table. The reason for this is that when considering only abort trajectories targeted to the center of the reentry corridor, the Service Module RCS does not have sufficient ΔV capability to perform an abort maneuver from the nominally planned earth parking orbit. However, a procedure is currently being evaluated where the abort maneuver is targeted for very near the overshoot boundary of the reentry corridor. Preliminary indications are that an abort using this technique will be available using the Service Module RCS propulsion system, although it will be marginal. As shown in Table I, the propulsion systems which are available and capable of performing an abort are the SPS, the LM propulsion systems, and the S-IVB.

Figure 6 summarizes the abort modes for earth parking orbit. The first and primary mode of interest, as shown, is a single coplanar deorbit burn targeted to provide horizon monitoring and which results in landing at a discrete area. The initiation time of this type of abort is carefully selected to provide the landing area control. This mode is, of course, very similar to that planned for Projects Mercury and Gemini. As shown in the chart of figure 6, the abort can be performed by either the SPS or S-IVB throughout the entire phase with, of course, the SPS being the prime propulsion system. Use of the S-IVB is <u>not</u> desirable due to possible recontact problems during reentry and, thus, would never be considered as an abort mode unless there had been a definite indication by the instrumentation that an SPS failure had occurred prior to CSM separation from the S-IVB. Also, if use of the Service Module RCS to deorbit proves feasible by targeting reentry near the overshoot boundary, the possible use of the S-IVB as an abort propulsion system would seem even more remote. Thus, use of the S-IVB as an abort propulsion system during this phase, seems very improbable even though it is available.

Figure 7 shows the major features of the SPS and S-IVB abort mode during this phase. Note that the transfer angle from the abort maneuver point to reentry is much less than 90°, which means the time from abort to reentry is on the order of 15 to 20 minutes. Command Module/Service Module separation occurs during the coast period from abort to reentry followed by the Command Module orienting itself for reentry. The burn attitude is such that the maneuver is coplanar and such that the earth's horizon remains at a fixed position in the Command Module window for crew monitoring purposes. The ΔV required for abort is approximately 500 fps, which is, of course, well within the SPS capability. The time of deorbit is selected so as to cause landing in a discrete recovery area, as mentioned previously.

The other abort mode possible for this phase is a DPS coplanar deorbit burn. This mode, as shown in figure 6, is available throughout the entire phase, but obviously requires transposition and docking in earth parking orbit.

Figure 8 shows the basic difference between this mode and the previously described one. Note that the deorbit maneuver is such that the spacecraft passes through apogee in order to provide enough time for the crew to transfer from the LM to the CSM prior to reentry. Also, the horizon cannot be easily monitored during the abort burn due to the spacecraft docked configuration. Thus, this attitude restriction is deleted for this mode. As for the S-IVB mode, the use of this mode is very improbable if the Service Module RCS deorbit proves feasible due to the same type of recontact problems upon reentry as would be experienced with the S-IVB deorbit mode.

An alternate DPS abort mode not shown on the summary chart is currently being investigated. This new DPS mode would consist of using the DPS to lower perigee to very near the atmosphere so that the resulting spacecraft trajectory would then be within Service Module RCS capability to deorbit. In other words, the procedure would require maneuvers by both the DPS and the Service Module RCS. This procedure would eliminate recontact problems during reentry, since the LM would be jettisoned between the DPS burn and the Service Module RCS burn.

Summarizing for this phase, one can say that more than adequate abort capability exists from strictly a performance standpoint with essentially three independent propulsion systems capable of providing the abort ΔV required continuously through the mission phase. However, the use of the S-IVB and DPS, as described in this section, would be very undesirable due to the recontact problems during reentry. If use of the Service Module RCS system to deorbit proves feasible, then the abort modes using the S-IVB or the DPS can essentially be eliminated from consideration.

Since the SPS mode is very similar to that planned for Projects Mercury, Gemini, and the early Apollo orbital flights, the detailed procedures and computer programs are already available and checked out for the ground computers. The SPS deorbit modes could be executed using either the onboard G&N system, the SCS system, or a strictly manual-type abort using visual attitude reference. The DPS abort mode would normally be executed using the onboard G&N system, although it could also be executed using the backup AGS system.

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Since the SPS mode is very similar to that planned for Projects Mercury, Gemini, and the early Apollo orbital flights, the detailed procedures and computer programs are already available and checked out for the ground computers. The SPS deorbit modes could be executed using either the onboard G&N system, the SCS system, or a strictly manual-type abort using visual attitude reference. The DPS abort mode would normally be executed using the onboard G&N system, although it could also be executed using the backup AGS system.

4.0 ABORTS DURING TRANSLUNAR INJECTION PHASE

This phase consists of the S-IVB burn which injects the spacecraft on a highly elliptical trajectory to rendezvous with the moon. The duration of this phase is approximately 340 seconds for a typical lunar mission. What is of interest here is to consider abort capability from orbits which would result from a premature or early burnout of the S-IVB stage.

Figure 9 shows the type of preabort orbits which would result from an S-IVB underburn during this phase of the mission. Note that the family of orbits are elliptical, having very nearly coincident lines of apsides with ever increasing apogee altitude up to and beyond lunar distance. The perigee altitude, however, remains relatively fixed, very near that of the original circular earth parking orbit altitude. The periods of these orbits, as shown in figure 10, vary all the way from $1\frac{1}{2}$ hours to approximately 400 hours as burn time increases. Actually, for free-return translunar profiles, the moon's gravitation perturbs the trajectory resulting from nominal burnout such that return to earth requires much less than 400 hours. Note also in figure 10 that the period remains relatively small (less than 10 hours) for more than three quarters of the way through the burn.

Table II shows the propulsion systems available which are capable of performing abort maneuvers during one portion or another of this phase as well as the subsequent phase, translunar coast. Note that there is a large ΔV capability with the service propulsion system even when the LM is attached. Also, there is moderate capability available with the LM propulsion systems, although use of them requires transposition and docking prior to abort. Note, however, that very little capability is available with the Service Module RCS. Because of this, aborts using the Service Module RCS are marginal at best, as will be shown later.

Figure 11 presents a summary of the abort modes available for translunar injection. Note that redundant abort capability exists throughout the entire phase and even double redundancy for the latter part of the burn. The first and primary abort mode shown in the summary chart consists of a single burn to return the spacecraft directly to reentry, as shown sketched in figure 12. This mode is available throughout the phase with either the SPS or DPS propulsion systems. Note also that the burn attitude is not constrained to be either coplanar or to enable horizon monitoring. A constrained attitude for aborts from this family of ellipses could result in excessive ΔV penalties and could also prevent landing at a desired recovery area.

Seven sub-modes exist which would be available from the ground for abort in this one basic mode. In other words, there exists seven different classes of abort trajectories which will be possible with a single unconstrained attitude burn. There was really no requirement for this kind of flexibility in the previous phase--in other words, during the earth parking orbit phase--because the spacecraft was always in the close proximity of the earth, and there was really no significantly different ways to return to reentry. Beginning with this phase, however, the spacecraft could be in a preabort orbit which extends a considerable distance from the earth. This fact, combined with the large spacecraft performance capability, means that abort trajectories having significantly different characteristics as regards ΔV, return time, and landing point are possible. Therefore, plans are being made to take advantage of this situation to provide greater flexibility in real time planning.

The seven sub-modes available are as follows:

a. Returns to primary recovery sites - This class of abort trajectory returns the spacecraft to a specific primary recovery site within the major constraints mentioned previously. For premature S-IVB shutdown during all but the very last few seconds of the nominal burn, this class of abort trajectories requires that the abort initiation time must be selected very carefully to cause landing at a discrete site, i.e., there will exist only small regions of abort initiation time which will allow returns to a discrete site. For late premature S-IVB shutdowns, however, reasonably placed recovery sites can be reached for any abort initiation time. This sub-mode also required, in general, that a plane change be made in order to reach a primary recovery site, unless there exists sufficient time for the spacecraft to remain in the preabort orbit until the primary recovery site rotates into the proper position for a coplanar abort maneuver. The delay time to abort, however, is usually limited if the apogee of the spacecraft preabort orbit extends into the radiation belts.

b. Time critical returns to a contingency recovery area - This class of abort returns the spacecraft to a contingency recovery area in the quickest possible return time consistent with the constraints of the situation. A contingency recovery area is a continuous line extending from a far northerly latitude, usually 30°, to a far southerly latitude, also usually 30°. These lines will approximately follow the contours of major continents such as the North American and South American continents and, thus, will be

very close to being a fixed longitudinal line. Landings on or near these defined lines can be supported by contingency recovery forces, resulting in recovery times nearly as fast, if not as fast, as those associated with the primary recovery sites. The expected existence of four or five of these contingency areas or lines placed equidistant around the world means that no long delay times will be required before the abort can be initiated if a time critical situation exists. The existence of these recovery lines also means that plane change manuevers will never buy a great amount of return time. For these reasons, this sub-mode is defined to be a coplanar maneuver with the abort initiation time selected to cause landing at the first accessible contingency recovery area or line.

c. Fuel critical returns to contingency recovery areas - This class of abort returns the spacecraft to the same contingency recovery areas or lines as described above, on a trajectory which requires the minimum possible ΔV or fuel. This class of abort return would only be used in a situation where there had been a serious degradation in the propulsion system. The very nature of the sub-mode limits or restricts the region of abort initiation to the vicinity of apogee of the preabort orbit.

d. Time critical returns to a water landing - This class of abort returns the spacecraft to a water landing in the quickest possible time. Note that the only restriction on the landing point is that it be on water. The landing is not restricted to a particular latitude, longitude, or combination thereof. It is assumed in this sub-mode, as it was in the previous time critical sub-mode, that the savings in return time made possible by plane changes were not worth the added logic complexity. Therefore, returns in this abort sub-mode are restricted to coplanar maneuvers. The selection of the abort initiation time is not as critical for this sub-mode as for those described above, and for a large percentage of the cases, the abort maneuver can be executed as soon as possible, i.e., as soon as the crew and systems are prepared and checked out. The only instances when the abort initiation time must be be delayed is to prevent a land landing. It should be pointed out that the use of this sub-mode is very improbable due to the small savings in return time possible when compared to the time critical returns to a contingency recovery area, as described above.

e. Fuel critical returns to a water landing - This class of returns is restricted to a water landing, as above, but provides the minimum ΔV solution rather than the quickest return time. Again, as in the fuel critical sub-mode described above, this class of return trajectory would

only be used in the situation where a serious degradation has occurred in the propulsion system. As would be expected, the region of abort initiation times for this sub-mode is restricted to the vicinity of apogee of the pre-abort orbit. The use of this sub-mode is more probable than its time critical counterpart due to the fact that a return to a contingency recovery area or line may not be possible when the abort initiation is restricted to the vicinity of apogee. Thus, a fuel critical return to a water landing would be the next best possible way to return.

f. Time critical returns to unspecified landing area - This class of abort returns the spacecraft in the quickest possible time with no restriction on the point of landing, i.e., the landing could be on land or water. These returns, by their very nature, are required to be coplanar and are initiated as soon as possible, i.e., as soon as the crew and systems are prepared and checked out. The use of this type of return is even more improbable than the time critical water landing sub-mode due to the small savings in return time as compared to the other possible sub-modes, and due to the possible degradation of crew safety associated with a land landing. Only a very extreme contingency situation would justify such drastic action.

g. Fuel critical returns to an unspecified area - This class of abort returns the spacecraft to reentry with the minimum possible ΔV required. The definition of this sub-mode requires that the maneuver be performed coplanar at apogee of the preabort orbit. This sub-mode would be used when only the Service Module RCS system was available and when the abort initiation time could not be delayed to some subsequent apogee passage due to radiation dosage or a preabort reentry condition, i.e., when the perigee of the preabort orbit is so low as to cause reentry during that particular orbit.

A straightforward procedure for selection of the best sub-mode is virtually impossible due to the complexity of the spacecraft systems involved, the possible failures or combination of failures which could have occurred, and the wide range of possible preabort orbits including the wide range of possible geographic orientations possible. For these reasons, flight controller judgment in real time must be heavily relied upon to make the sub-mode selection based on the exact situation at hand. The flight controllers will, of course, develop many mission rules to aid him in this decision. The current plan is to display the significant characteristics of abort trajectories in these various categories or sub-modes to the flight controller after a decision to abort has been made. These characteristics would include such items as (1) ΔV required,

(2) return time, (3) landing location, and (4) whether landing is in daylight or darkness. The flight controller would then select the one which, in his judgment, provides the greatest margin of crew safety. It is obvious that, all things being equal, an abort to a primary recovery site would be selected if at all possible. In order of decreasing priority, then, would be aborts to a contingency recovery area, aborts to a water landing, and finally, for an extreme contingency situation, aborts to unspecified areas. However, all things may not be equal for a given abort situation, and a tradeoff may be required between the possible ways of returning in order to maximize crew safety.

Note that the above described sub-modes would be available from the ground not only for this phase but for all the remaining mission phases to reentry. The onboard backup capability will be restricted to the latter two sub-modes (f and g) due to computer storage limitations, but certain manual iterations can be performed by the crew to provide some measure of control over longitude of landing.

As shown in figure 11, the only other basic abort mode for this phase is the Service Module RCS coplanar deorbit returning the spacecraft directly to reentry using the minimum possible ΔV. This mode of abort is available in one of three possible sub-modes, as described below.

a. Fuel critical returns to contingency recovery areas.

b. Fuel critical returns to water landing.

c. Fuel critical returns to an unspecified area.

The definitions of these sub-modes are identical to those described previously for the SPS and DPS propulsion systems. Note from figure 11 that this mode of abort is available only during the latter part of the TLI burn. Again, it should be emphasized that this latter statement assumes that targeting for the abort maneuver would be for the middle of the reentry corridor. Targeting for near the overshoot boundary could conceivably make this made of abort available throughout the entire phase.

Another abort mode not shown in the summary chart is currently under investigation and evaluation. This new mode would consist of two burns and would be considered only if the SPS were available. The first burn would be a circularization maneuver at perigee of the preabort ellipse. The second burn would be a standard deorbit maneuver as described in the earth parking orbit phase, i.e., a coplanar burn targeted so as to provide the crew with horizon monitoring. This procedure would allow the crew to retain the simplified abort procedures of the

an SPS abort in the same situation. Note also that the switchover line dividing direct aborts from post-pericynthion aborts is now back to approximately 22 hours after translunar injection due to the much smaller ΔV capability associated with the DPS propulsion system.

Figure 29 shows another significant characteristic of translunar coast aborts as regards landing accessibility. Note that the abort return geometry requires that all possible abort trajectories from a given abort position must pass through the antipode or negative of the abort position vector. Since the transfer angle from abort to reentry is always approximately 170 to 180 degrees, then the reentry positions must be in the immediate vecinity of the antipode. The fact that the reentry ranging capability is relatively limited together with the constraint on return inclination forces the landing point to a relatively restricted band of possible latitudes.

Figure 30 shows typical landing area accessibility for aborts from a point midway along the translunar coast trajectory. The shaded region represents the possible geographic landing areas for aborts from this one position vector. Note that the region of possible landing latitudes is relatively restricted whereas any longitude is accessible. The widening of the accessible area from east to west is associated with slower and slower return trajectories. A slower return means that more energy can be utilized for plane changes which widens the possible latitude band. The particular case shown is an abort position vector at a southerly latitude or declination; therefore, the abort antipode, as well as all possible landing points, are notherly in latitude or declination. Thus, recovery sites should be placed at northerly latitudes when the lunar mission is planned for a southerly lunar declination and vice versa. Proper positioning of recovery sites is represented in figure 30 by the "X" marks.

Summarizing, then, for the translunar coast phase: (a.) Redundant abort capability exists throughout the entire phase. (b.) Spacecraft ΔV capability is large compared to the minimum required due to the free-return profile. (c.) This large margin of capability can be utilized to speed up returns and/or to control the landing area. (d.) No portion of this mission phase is critical or marginal unless both SPS and DPS failures occur.

6.0 ABORTS DURING LUNAR ORBIT INSERTION PHASE

This phase consists of the SPS maneuver which transfers the spacecraft from the approach hyperbola at the moon into a circular parking orbit about the moon. The

prior to the time of pericynthion were computed in the direct abort mode. Note that the time of landing for a particular landing site is relatively insensitive to the time of abort until a limit is reached (Δ V required or reentry velocity). At this point, aborts to that same landing site are forced to a time of landing 24 hours later where it remains fixed with delay time until one or the other of the limiting constraints is reached again. Note, however, that if a different landing site is considered when a limit is reached, the jump or discontinuity in time of landing would be something less than 24 hours. Therefore, if the recovery sites were placed approximately equidistant around the world, the incremental increases in the time of landing to a specific site would always be some fraction of 24 hours. Another interesting feature of figure 27 is that, after an abort time of approximately 57 to 58 hours, an abort delayed to the vicinity of pericynthion or beyond will result in the same time of landing as direct aborts prior to pericynthion. Also, since aborts performed following pericynthion will always require much less ΔV than direct aborts prior to pericynthion, the obvious course to follow is to delay abort to post-pericynthion after this switchover line is reached on the translunar coast trajectory. It should be pointed out that this switchover line from direct abort to delay abort to post-pericynthion is a strong function of ΔV required for abort. The particular switchover line shown in this figure assumes 90% of the entire SPS capability (9,000 fps) available for abort.

Another interesting characteristic to note from this figure is that returns from the vicinity of pericynthion are limited to approximately 40 to 50 hours return time due to the reentry velocity constraint. Without this constraint, aborts could be performed post-pericynthion with return times of approximately 25 to 35 hours, assuming 9,000 fps available.

Figure 28 shows the same type of data for DPS aborts from the same translunar coast trajectory as used for figure 27. Note that although the times of landing still demonstrate the 24-hour effect, the times are much later than those for SPS aborts. For example, consider an abort performed at 20 hours after translunar injection targeted to return to Hawaii. Figure 28 shows that a DPS abort in this situation would result in a time of landing of approximately 120 hours after translunar injection. However, figure 27 shows that a SPS abort in this same situation would result in a landing at Hawaii at approximately 48 to 50 hours after translunar injection. Thus, in this particular case, a DPS backup abort would return three days later than

length of the lunar orbit insertion burn is approximately 380 seconds for a typical mission. Of interest here is the abort capability from preabort trajectories resulting from premature or late SPS burnout during the lunar orbit insertion burn. As will be described later, the abort procedures are dependent upon the time of SPS burnout during this phase.

Figures 31 and 32 show the different kinds or classes of preabort trajectories which could result from an underburn or an overburn during lunar orbit insertion and the associated burnout times at which they would occur. For early premature burnout, approximately 0 to 130 seconds, the resulting trajectories escape the influence of the moon and return into the earth's influence. These escape trajectories are either hyperbolic or extremely elliptical relative to the moon.

For premature burnouts occuring from approximately 130 to 145 seconds, the resulting preabort trajectories are unstable in character. The apogees of this class of preabort orbit extend to the vicinity of the lunar sphere of influence where the trajectory loses its central force field or two-body type of motion due to the large perturbations from the earth. These perturbations are so severe as to sometimes cause the motion to reverse from retrograde to posigrade with respect to the moon, as shown in fugure 31. The periods of these preabort trajectories, if such can be defined, are generally greater than 100 hours.

For premature shutdowns from approximately 145 to 160 seconds, the resulting trajectories are relatively stable ellipses but are pertubed sufficiently by the earth to cause impact after one revolution. Periods for this class of trajectories range from approximately 35 to 100 hours.

For burnouts occurring from approximately 160 seconds to 10 seconds past the nominal burnout, the resulting trajectories are non-impact stable ellipses with periods ranging from approximately 35 to 2 hours.

Premature burnouts occurring 10 seconds or more after nominal burnout result in impact ellipses after less than one-half revolution. This class of preabort trajectory is quite obviously the most critical from an abort standpoint, but also is the most improbable one to occur due to the redundant monitoring devices available to the crew onboard the spacecraft.

Before discussing the various abort modes and procedures for thses preabort trajectories, it would seem worth while

The last major abort mode associated with this phase is a two-burn abort procedure for LOI overburns which result in preabort ellipses which impact in less than one-half revolution. Use of either the SPS or the LM propulsion systems can be considered with this mode although use of the LM propulsion systems is marginal. Figure 38 shows that the first burn of this two-burn procedure is executed as soon as possible to clear perilune or pericynthion altitude to an acceptable value. The second burn is executed on some later orbit in the approximate quadrant shown to return the spacecraft to earth. It again should be emphasized that it is very improbable that such an overburn could occur due to the redundant monitoring capabilities available to the crew during the lunar orbit insertion burn.

Figure 39 shows the effect of delay time on abort V required for preabort escape trajectories. Shown plotted is the minimum ΔV required to abort as a function of the time of abort measured from lunar orbit insertion burnout Data for two different escape trajectories are shown--one associated with a lunar orbit insertion premature burnout after 40 seconds and the other after 80 seconds. Note that the ΔV requirements tend to flatten out after the first one or two hours with the later burnout producing the steepest initial slope. Although the abort requirements for the two trajectories shown are well within the LM propulsion system capability, the trend should be obvious that later and later premature burnouts will eventually cause the LM propulsion system capability to be exceeded for direct-type ASAP aborts.

Figure 40 shows the same type data for a typical preabort ellipse having a period of approximately 4 hours. Note that, although the initial slope of the abort ΔV required/delay time curve is extremely steep, the ΔV required is cyclical with delay time. As shown, this effect permits aborts with the LM propulsion system if the proper time of abort is chosen. This proper time of abort corresponds to the vicinity of perilune of the preabort ellipse.

Figure 41 displays the abort capability using the LM propulsion systems throughout a typical lunar orbit insertion phase. Shown plotted is the ΔV required to abort as a function of the time of premature lunar orbit insertion burnout for several delay times. Also shown on the plot are two limit lines associated with the LM propulsion system capability. First notice that the minimum ΔV required for an immediate abort (essentially zero delay time) is always within the capability of the LM propulsion systems throughout the entire lunar orbit insertion phase. Note, however, that for an abort delay

performed. Thus, precious "thinking" and preparation time can be had to obtain a better assessment of the situation before action is taken. Note from the summary chart that this mode using the LM propulsion systems is shown available through the first 200 seconds of burn time. This assumes that the LM can be prepared for an abort maneuver in approximately one-half hour. For longer preparation times, use of the LM in this mode would be eliminated at an earlier LOI burn time.

The next mode of interest is a two-burn procedure for use when the spacecraft is on a preabort ellipse which impacts the moon after one revolution. As shown in the summary chart, this mode is available with either the SPS or the LM propulsion systems for a period of approximately 15 to 20 seconds near the middle of the lunar orbit insertion burn. This time period, of course, corresponds to the premature burnout times which would result in impact ellipses. Figure 36 shows the basic features of this abort mode. Note that the first burn is made in the vicinity of apogee to raise perilune to an acceptable value with the second burn being executed on the back side of the moon in the vicinity of perilune to return the spacecraft to earth. This mode is considered prime only during the latter part of its region of availability due to the excessive delay times associated with the preabort ellipses during the earlier portion. A very significant feature of this abort mode, as shown in the summary chart, is that it overlaps the immediate direct abort mode even when considering use of the LM propulsion systems. The amount of this overlap, however, is dependent upon the LM preparation time required in the direct abort mode. It should be recalled that the LM propulsion system direct abort mode shown in the summary chart assumes a LM preparation time of one-half hour. Longer preparation times would reduce this degree of overlap and, as will be shown later in more detail, could even eliminate the overlap between the two modes.

The next abort mode to be considered consists of delaying one or more orbits to abort when premature SPS burnout results in a stable non-impact ellipse. As shown, this mode is available using either the SPS or LM propulsion systems throughout the entire latter half of the burn phase. The basic features of this abort mode are shown sketched in figure 37. As shown, the maneuvers are always delayed at least one orbit with execution occurring in the vicinity of perilune to return the spacecraft to earth. The longest delay time which would ever be encountered before the abort maneuver could be executed is approximately one and a half days corresponding to the maximum period of the possible non-impact ellipses.

to describe the two basic ways of returning from the moon and why one can be ruled out. Figure 33 shows the two basic ways of returning in a simplified "patched conic" model. As shown, after exiting the moon's sphere of influence, the spacecraft will either pass through apogee along the earth phase conic or it will not pass through apogee. In order to return post-apogee, the spacecraft must exit the moon's sphere of influence in the quadrant shown such that the exit velocity with respect to the moon will combine with the velocity of the moon with respect to the earth to result in the required velocity of the vehicle with respect to the earth. Similarly, if the spacecraft is to return pre-apogee in the earth phase conic, the exit point must be on the far side of the moon in order that the velocity of the spacecraft relative to the moon can combine with the velocity of the moon relative to the earth to result in the required spacecraft velocity relative to the earth. This latter class of return trajectories which pass through apogee along its earth phase conic can be ruled out from consideration due to prohibitive return time and very difficult targeting requirements.

The propulsion systems available for abort during the lunar orbit insertion phase are shown in table III along with their respective ΔV capabilities. Note that there are basically two independent propulsion systems available during this phase--the SPS system and the LM propulsion systems. The ΔV capabilities vary through the phase due to the change in mass brought about by SPS fuel expenditure.

Figure 34 summarizes the abort modes for the lunar orbit insertion phase. As shown, redundant abort capability exists continuously throughout the entire phase, if the proper abort modes are chosen.

The first abort mode listed is an immediate (as soon as possible) direct abort, as shown sketched in figure 35. This mode is available with either the SPS, the LM propulsion systems, or for a very small region, the Service Module RCS System. This mode consists of a single burn performed as soon as preparation is completed which returns the spacecraft directly to earth. As shown in the summary chart, this mode is available throughout the entire phase when the SPS is available and is shown as prime for approximately the first 150 seconds of lunar orbit insertion burn. This mode is not prime, though available, for the latter portion of the lunar orbit insertion burn due to the fact that the preabort trajectories are elliptical with periods short enough to reasonably allow one complete revolution before the abort maneuver is

time of one-half hour, direct aborts cannot be considered using the LM propulsion systems for premature burnout after approximately 190 seconds. The cutoff for delays of one hour and two hours occurs at approximately 160 and 135 seconds, respectively. However, as shown, LM DPS aborts using the delay-one-orbit mode for burnouts in the vicinity of 140 to 150 seconds result in extremely long delay times before the abort maneuver to return the spacecraft to earth can be executed. These delay times are on the order of approximately 100 hours. Thus, in order to eliminate the possibility of being forced to these extremely long delay times, the LM preparation time should be minimized to produce as large an overlap as possible between the direct abort mode and the delay-one-orbit abort mode. Under investigation now is the possibility of checking out at least some of the LM systems prior to lunar orbit insertions to expedite abort preparation time. Other studies now being performed in an attempt to alleviate this problem are:

a. Evaluation of a two-burn abort procedure where the first burn would shorten the period of the preabort ellipse with the second burn being executed in the vicinity of perilune to return the spacecraft to earth.

b. Evaluation of the use of the Service Module RCS to continue the lunar orbit insertion burn in the event of a SPS premature burnout as long as possible in order to shorten the period of the preabort ellipse.

Figure 42 presents abort performance data for returns to primary recovery sites from a preabort ellipse about the moon, having a period of approximately 19 hours. Shown plotted is the time of landing as measured from lunar orbit insertion burnout as a function of the time of abort measured from lunar orbit insertion burnout. Again, as for translunar coast, three recovery sites are considered--Indian Ocean, Hawaii, and Bermuda. Note the the 24-hour effect appears here as it did in the translunar coast phase; and also note that the times of landing are cyclical (repeatable after one orbit).

Summarizing, then, for the lunar orbit insertion phase it can be seen that:

(a.) Redundant abort capability exists throughout the entire lunar orbit insertion phase if the proper abort mode is chosen.

(b.) The only critical regions are:

1. Premature burnouts in the vicinity of 150 seconds where the delay-one-orbit mode would result in extremely delay times before an abort maneuver could be executed.

2. Overburns which would result in impact ellipse in less than one-half revolution.

However, different techniques are being studied to eliminate the possibility of the long delay times associated with critical region "a" and critical region "b" is very improbable due to crew monitoring techniques.

7.0 ABORTS DURING LUNAR ORBIT COAST PHASE

This phase of the lunar landing mission consists of the spacecraft coast in lunar orbit from lunar orbit insertion to initiation of transearth injection. As would be expected, the abort procedures are very similar to the normal transearth injection procedures.

The propulsion systems available for aborts during the lunar orbit coast phase are shown in table IV with their ΔV capabilities. Again, two independent systems are available--the SPS and the LM propulsion systems.

Figure 43 summarizes the abort modes for the lunar orbit coast phase. As shown, continuous abort capability exists using the SPS, but redundant abort capability exists only for the early portion of the phase prior to initiation of LM descent. The abort mode for this mission phase consists of a single abort burn on the far side of the moon to return the spacecraft to reentry in a manner similar to that of earth injection, as shown in figure 44.

Figure 45 shows the affect of delay time on abort ΔV required for aborts for the lunar orbit coast phase. Plotted is the minimum ΔV required for abort as a function of the time of abort and the longitude of abort for a typical landing mission. Although the slopes are steep, the ΔV requirements are cyclical in phase with the period of the orbit. Note also that the minimum ΔV required occurs at a longitude just prior to 180°.

Summarizing the abort capability from the lunar orbit coast phase, then, one can see that continuous abort capability exists throughout the entire phase only if the SPS is available.

8.0 ABORTS DURING TRANSEARTH INJECTION PHASE

This phase consists of the SPS burn which injects the spacecraft from lunar orbit onto a return trajectory to earth. The length of this phase for a typical lunar landing mission is approximately 120 seconds. The aborts of interest are those performed from trajectories

resulting from premature SPS burnout during this phase. As would be expected, the abort procedures are very similar to those used for the lunar orbit insertion phase.

Figures 46 and 47 show the different classes of preabort trajectories and their respective times of occurrence for the transearth injection phase. As shown, the same classes of trajectories exist as for the lunar orbit insertion phase with the exception of the overburn impact ellipse class.

Table V shows the only propulsion system available for the transearth injection phase with its ΔV capability. As for the lunar orbit insertion phase, the ΔV capability varies due to the change in mass through the TEI burn.

Figure 48 summarizes the abort modes for the transearth injection phase. Note that continuous abort capability exists if the SPS can be restarted. As shown, the modes are very similar to those planned for lunar orbit insertion. The delay-one-orbit mode is prime during the earlier portion of the transearth injection burn to allow additional delay time for a better assessment of the situation. However, as shown, this abort mode is not considered prime after approximately 70 seconds due to the exessively long delay times required before the abort maneuver could be initiated.

Likewise, the two-burn, delay-one-orbit procedure is not considered prime, even though it is available for a small region, due to the long delay times required before the abort could be initiated. There is more reason to avoid these long delay time regions during this phase than for the lunar orbit insertion phase, due to the lifetime of the spacecraft's systems becoming more critical as the length of the of the mission increases.

The immediate direct abort is available throughout the entire phase and is considered prime during approximately the last half of the phase to avoid excessive delay times associated with the delay-one-orbit modes.

Figure 49 shows the SPS abort capability during a typical transearth injection phase. Shown plotted is the minimum ΔV required for abort as a function of the time of premature transearth injection burnout. Shown plotted are the ΔV requirements for an immediate abort, an abort delayed one hour, and an abort delayed two hours. Also shown are the abort requirements for a delay-one-orbit abort which happened to be identical to the immediate abort ΔV requirements for that portion of the mission phase. Shown as a

limit line is the curve representing SPS capability for this particular mission. It should be recognized that this limit line is mission dependent as the amount of SPS fuel reserve will vary from mission to mission. As shown, aborts delayed two hours or longer will completely eliminate the overlap between the direct abort mode and the delay-one-orbit to abort mode.

Summarizing the abort capabilities for the transearth injection phase, then:

a. Continuous abort capability exists only if the SPS can be restarted.

b. The only critical region exists for premature burnout in the vicinity of 80 seconds where allowable delay time before the SPS must be restarted is on the order of two hours. Premature burnout before or after this critical region allows delay times greater then two hours before restart is required.

9.0 ABORTS DURING TRANSEARTH COAST PHASE

This mission phase consists of the coast trajectory from transearth injection to reentry. The nominal transearth coast results in landing at prime recovery site. An abort could (1) speed up the return to the same site or to an alternate site or (2) change the landing site to avoid bad weather.

Since the transearth coast is already returning to a prime recovery site, however, the probability of ever aborting during this phase is small.

Table VI lists the propulsion system available and capable for abort during this mission phase and its ΔV capability. Note that the only system available is the SPS with a typical ΔV capability of 1,500 fps. This ΔV capability is, of course, very mission dependent but will usually be of this order of magnitude, at least for the earlier missions.

Figure 50 is a summary of the abort modes for the transearth coast phase. Note that only one mode is available, that being a single SPS burn returning the spacecraft directly to reentry, as shown sketched in figure 51. The mode is available continuously throughout the entire phase, although it is obvious that the closer to reentry, the less likely an abort would be executed.

Figure 52 presents actual abort performance data for aborts to primary recovery sites for a typical transearth coast trajectory. Shown plotted is the time of landing as measured from transearth injection as a function of the time of abort measured from transearth injection. Data is shown for three primary recovery sites--Indian Ocean, Hawaii, and Bermuda. As shown, the nominal transearth trajectory is targeted to return to Hawaii at the time of landing of 90 hours as measured from transearth injection. Aborts with a time of landing 24 hours earlier to Hawaii can be performed during the first 10 hours of transearth coast. Note also that the faster returns are available to alternate landing sites during the first 45 hours of transearth coast. Also shown is the availability of returns to an alternate (Indian Ocean) site with a later time of landing up through approximately the first 55 hours of transearth coast. These data, then, show considerable flexibility or abort capability to speed up the return and/or to change the point of landing. It is also of interest that returns to primary recovery sites still display the 24-hour effect, as in the previous phases.

10.0 CONCLUSIONS

A. Continuous abort capability exists for all mission phases under the following conditions:

   1. For the lunar orbit insertion and transearth injection phases, the selection of the proper abort mode as a function of the time of premature burnout is a requirement.

   2. Following the initiation of LM descent in lunar orbit, the SPS must be operable to provide abort capability.

B. Continuous redundant abort capability exists for all mission phases prior to the initiation of LM descent with the exception of the launch phase.

Questions and Answers

EARTH RETURN ABORT CAPABILITIES

Speaker: Ronald L. Berry

1. Dr. Rees - Why not use the APS for a backup propulsion system?

   ANSWER - Since the ascent engine is not gimballed, the possible c.g. offset effects cannot be controlled.

2. What is the RCS engine burn time limitation?

   ANSWER - Specifications limit is 1000 seconds.

3. Dr. Mueller - Have methods to restart the SPS engine in the event of an early shutdown during transearth injection burn been investigated?

   ANSWER - If the problem is a guidance or control problem, the engine can be restarted and controlled manually. If the problem is with the SPS engine itself, nothing can be done. This is one of the accepted risks in the program.

4. Dr. Haeussermann - Have we looked into using a DPS-SPS combination?

   ANSWER - No. The main reason for using the DPS as backup is because of an SPS failure.

5. Mr. Richter - Can't we use ADS some of the time - isn't it load dependent?

   ANSWER - The control authority is marginal in most cases for the APD since the APS engine is not gimballed.

6. Mr. Richter - Can we use two SPS burns for fast earth return transfer?

   ANSWER - This is being looked into. This may not be a desirable way to achieve a gain in return time, since it could be dangerous. If the SPS did not fire the second time, you might exceed acceptable entry velocity or conditions.

7. Mr. Green - When the DPS is used for backup propulsion, do we jettison the SPS propellants?

   ANSWER - There is no capability to jettison SPS propellants.

8. Dr. Mueller - Have the procedures, etc., have been worked out for all of the abort possibilities?

   ANSWER - No.

9. Dr. Von Braun - Why the difference in the translunar and transearth transit times?

   ANSWER - The translunar phase uses a free return trajectory which limits the transfer time to a narrow band. The transearth time is primarily limited by the energy available and the amount of consumables remaining.

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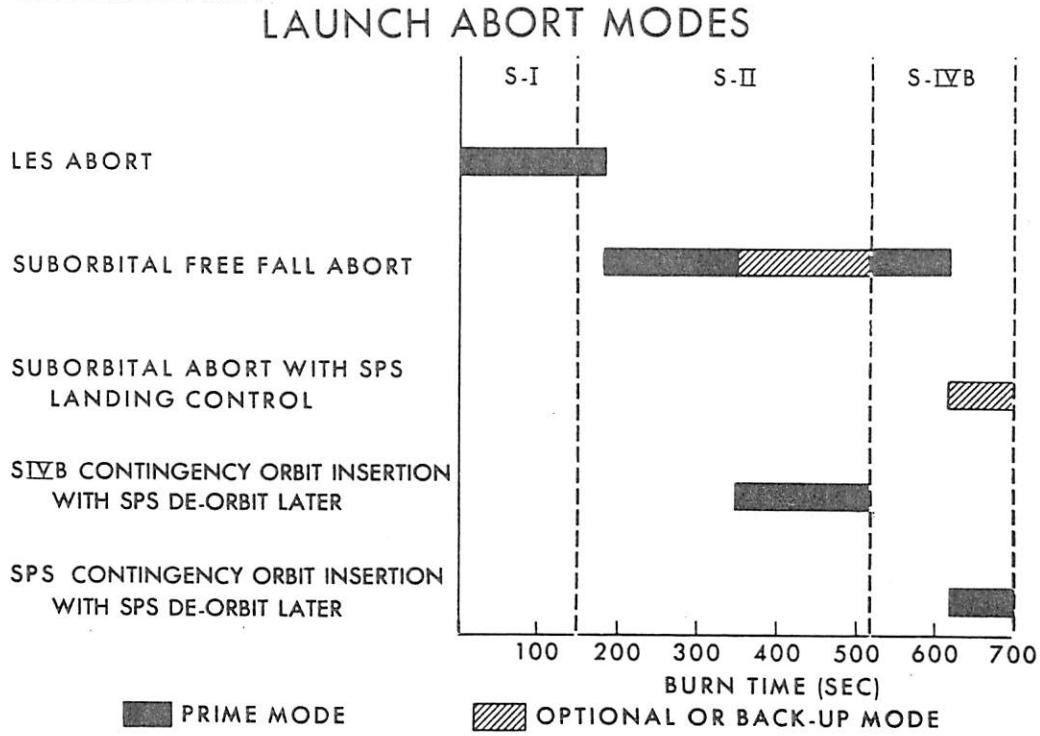

Fig. 1

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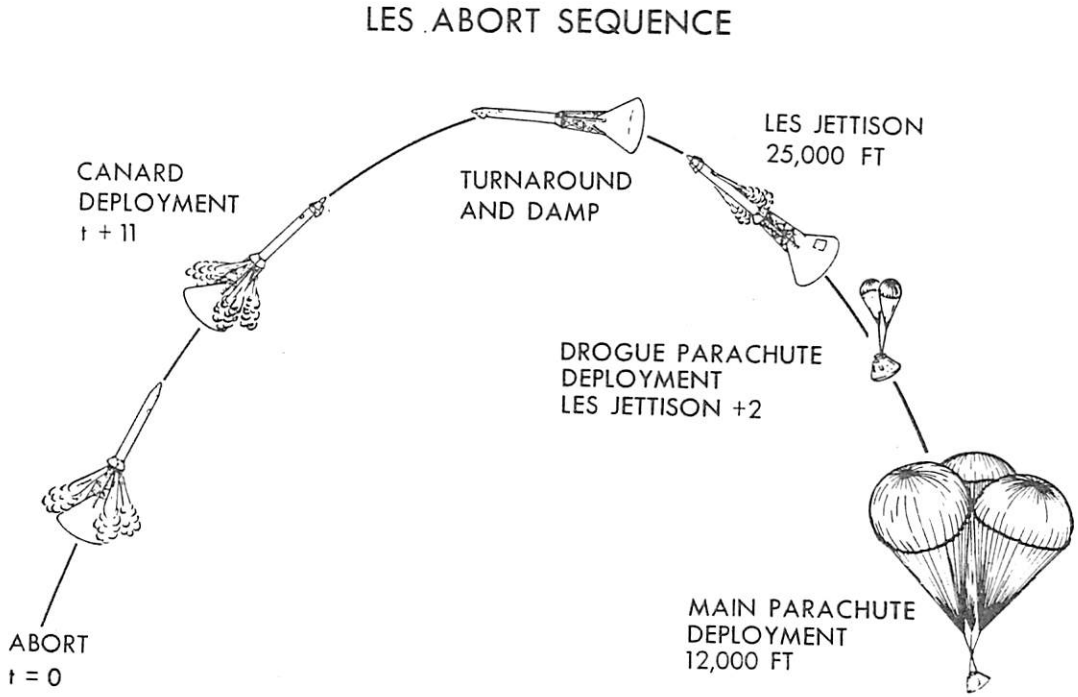

Fig. 2

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SUBORBITAL ABORT WITH NO SPS LANDING AREA CONTROL

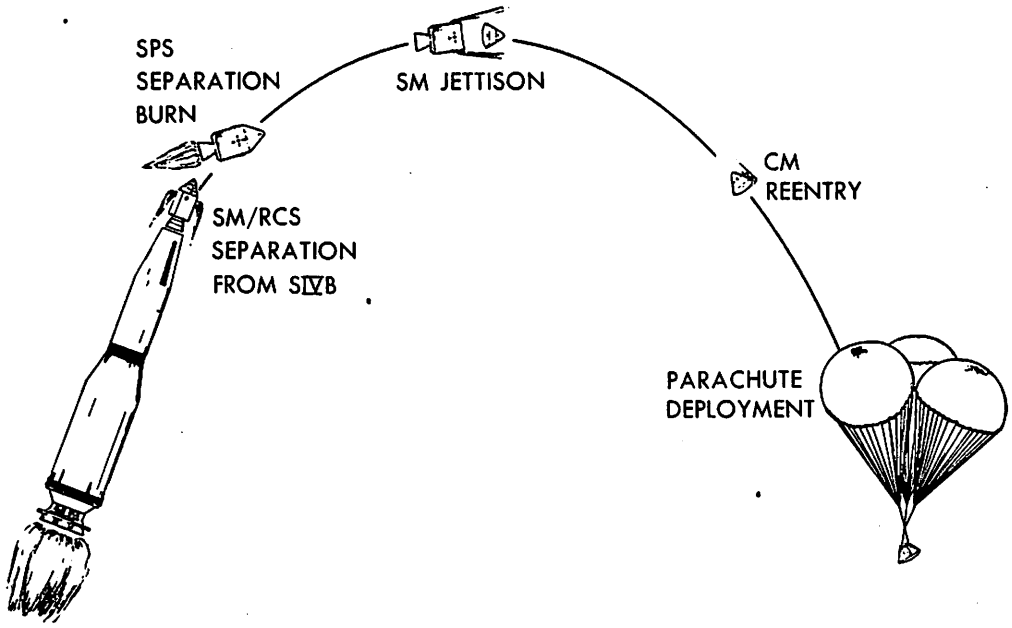

Fig. 3

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SUBORBITAL ABORT WITH SPS LANDING AREA CONTROL

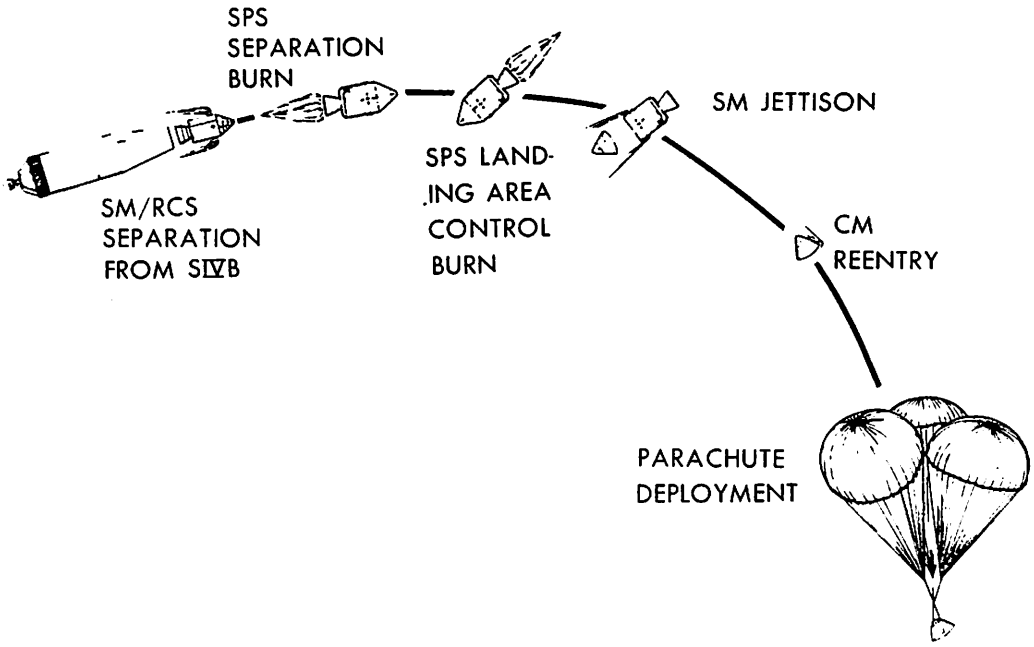

Fig. 4