NASA-S-66-6486 JUN

S-IVB OR SPS CONTINGENCY ORBIT INSERTION FOLLOWED BY SPS DE-ORBIT TO REENTRY

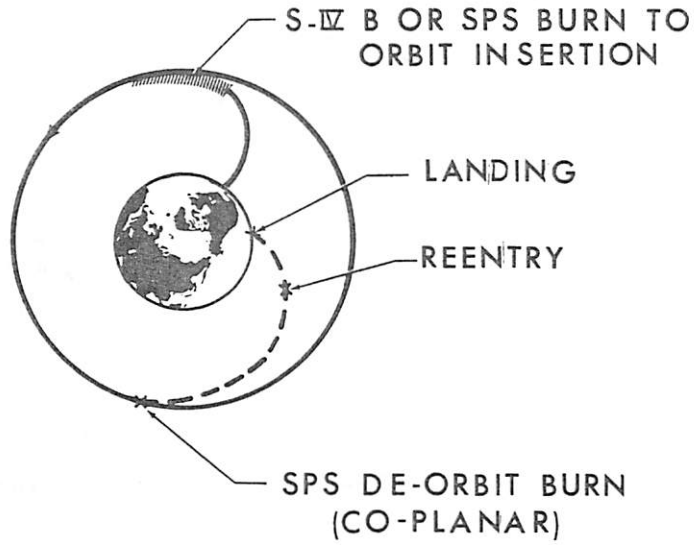


Fig. 5

NASA-S-66-6415 JUN

EARTH PARKING ORBIT ABORT MODES

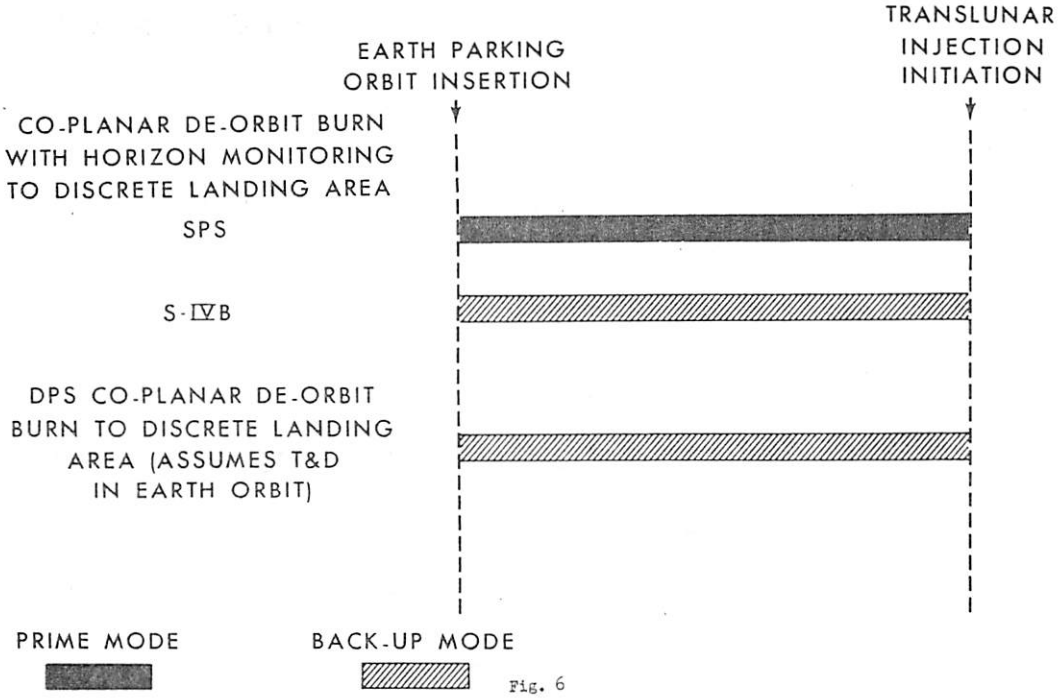


Fig. 6

NASA-S-66-4987 MAY 26

SPS OR S-IVB DE-ORBIT ABORT FROM EARTH PARKING ORBIT

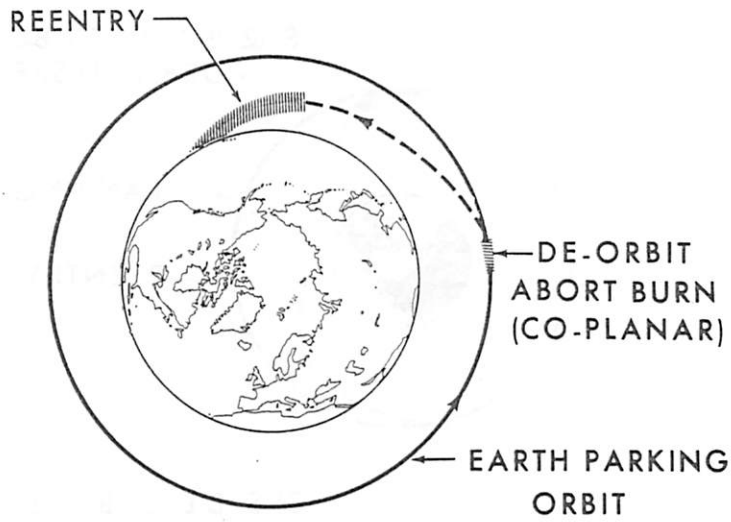


Fig. 7

NASA-S-66-6499 JUN

LM DPS DE-ORBIT FROM EARTH PARKING ORBIT

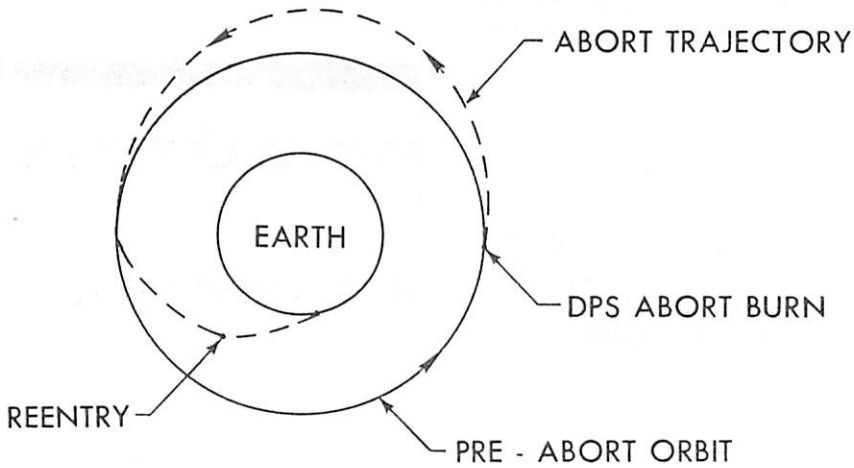


Fig. 8

NASA-S-66-4977 MAY 25

PRE-ABORT ORBITS DUE TO PREMATURE BURN-OUT DURING TRANSLUNAR INJECTION

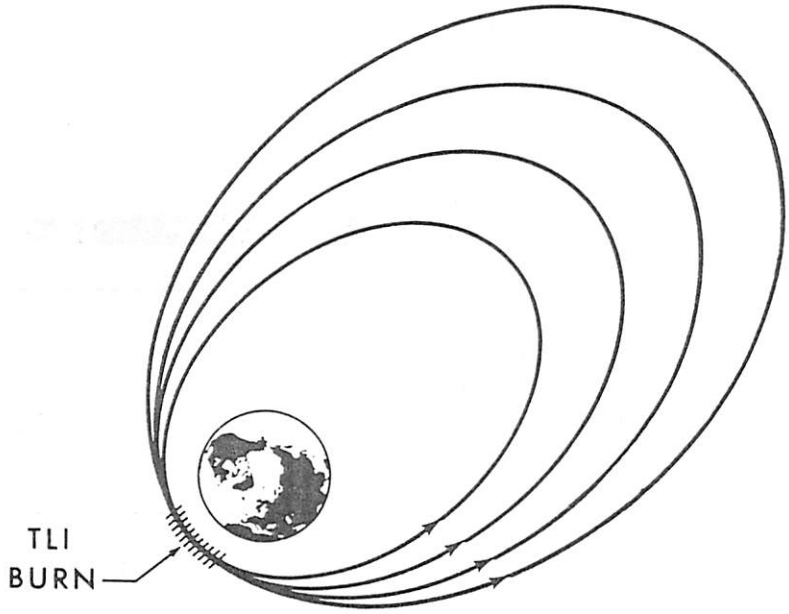


Fig. 9

NASA-S-66-6397 JUN

THE ORBITAL PERIOD AS A FUNCTION OF S-IVB BURN TIME

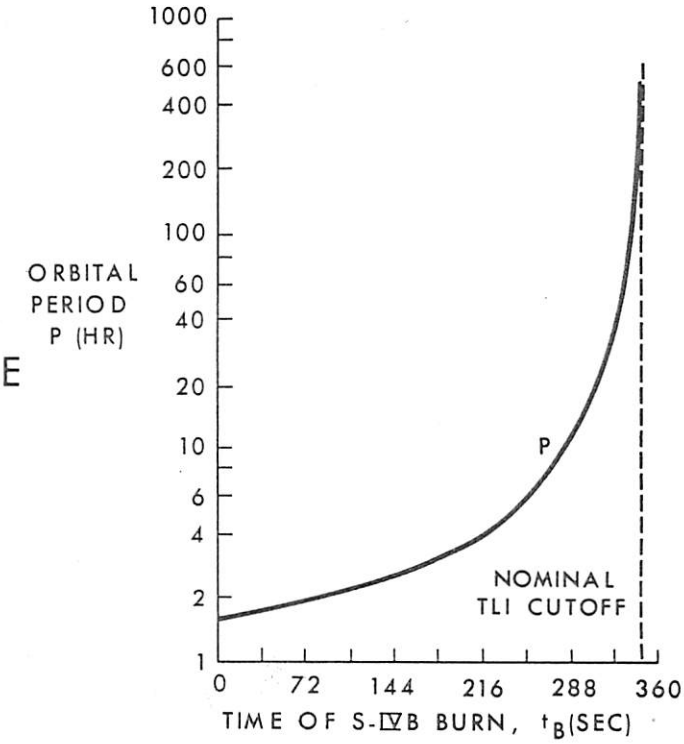


Fig. 10

NASA-S-66-6410 JUN

TRANSLUNAR INJECTION ABORT MODES

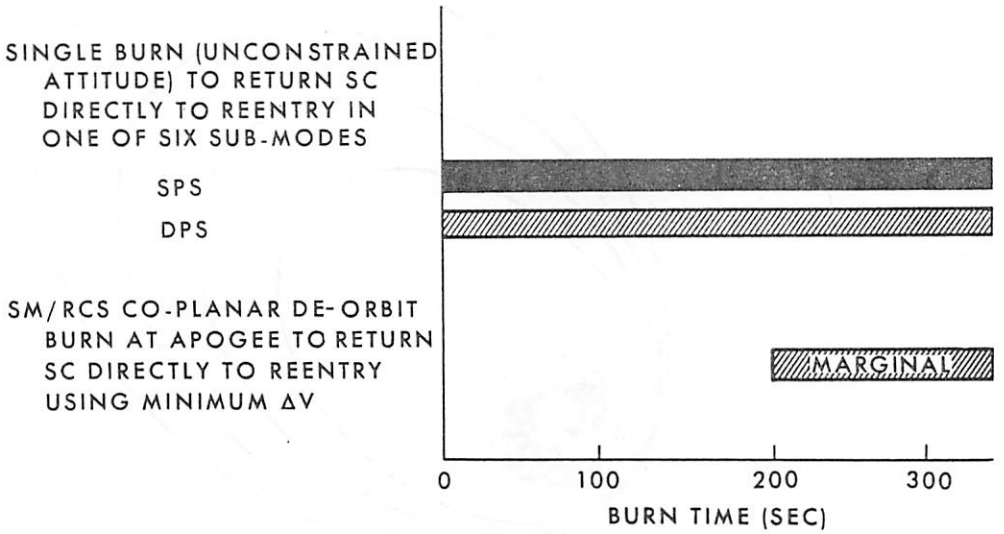


Fig. 11

NASA-S-66-6500 JUN

ABORTS FROM TRAJECTORIES RESULTING FROM TLI FAILURE

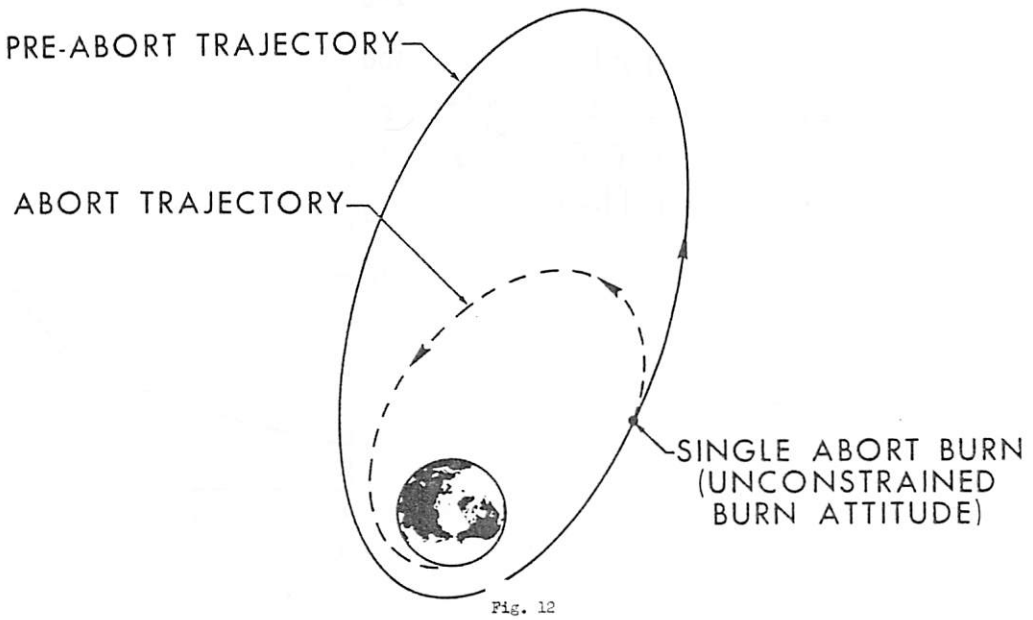


Fig. 12

NASA-S-66-4994 MAY 26

FUEL CRITICAL ABORT FOLLOWING TLI FAILURE

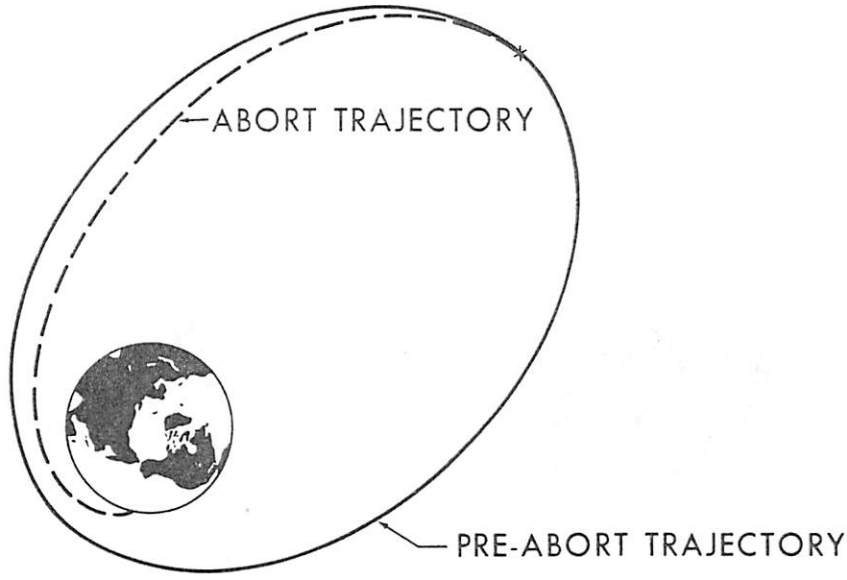


Fig. 13

NASA-S-66-4982 MAY 25

FUEL CRITICAL
ΔV REQUIREMENTS
FOR ABORTS
FOLLOWING
PREMATURE
SHUTDOWN
DURING TLI

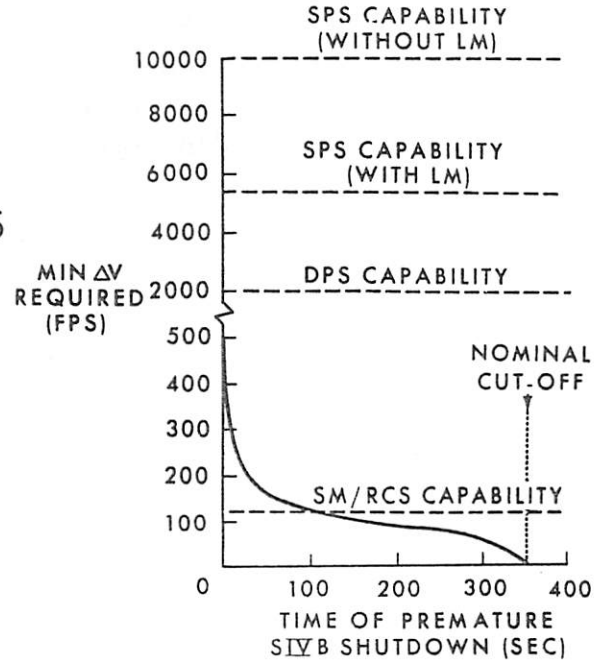


Fig. 14

NASA-S-66-4984 MAY 25

TIME CRITICAL ABORT FOLLOWING TLI FAILURE

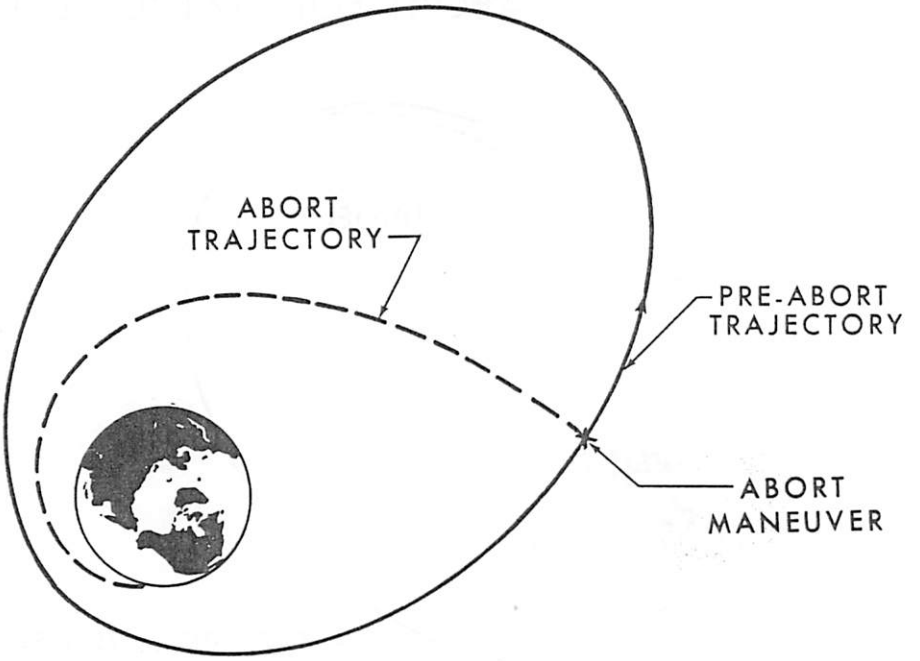


Fig. 15

NASA-S-66-4975 MAY 25

TIME CRITICAL RETURN TIMES FOR ABORTS FOLLOWING PREMATURE SHUTDOWN DURING TLI & ASSUMING SPS AVAILIBLE ($\Delta V = 10{,}000$ FPS)

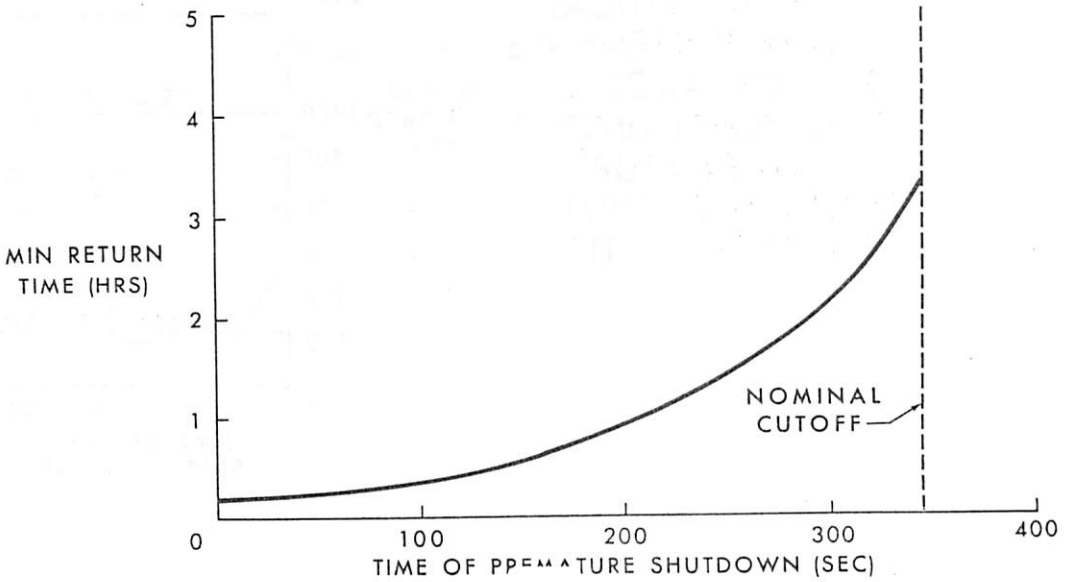


Fig. 16

NASA-S-66-4990 MAY 26

FUEL CRITICAL ABORT VELOCITY INCREMENTS, ΔV(FT/SEC) - BERMUDA

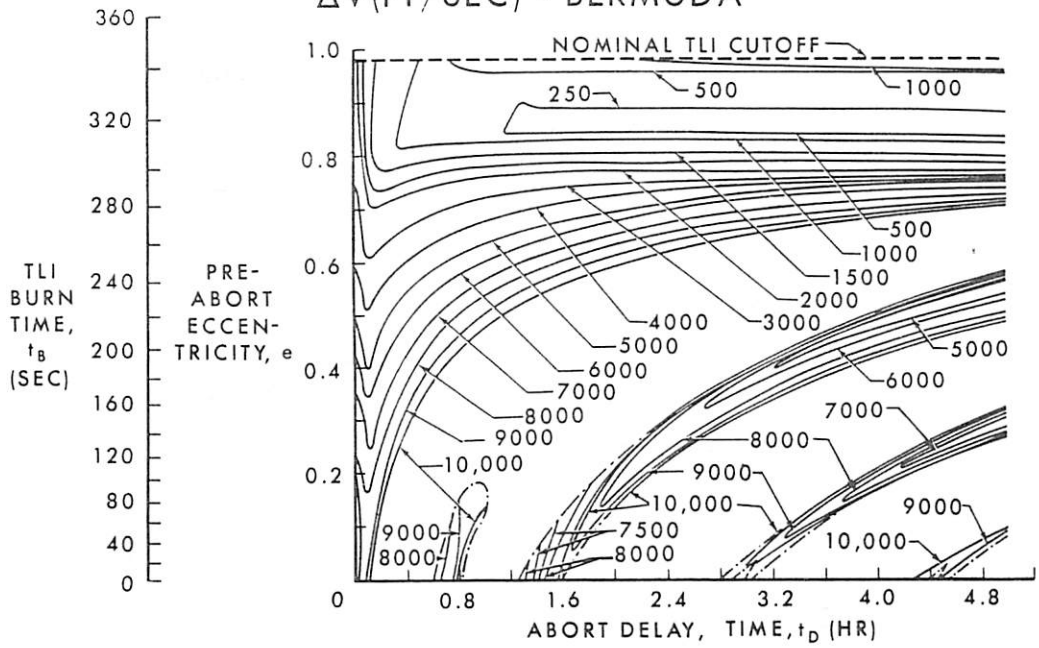


Fig. 17

NASA-S-66-5000 MAY 26

TIME CRITICAL TOTAL TRIP TIME REGIONS

BERMUDA – ΔV MAX = 10,000 FPS (ENTIRE SPS CAPABILITY)

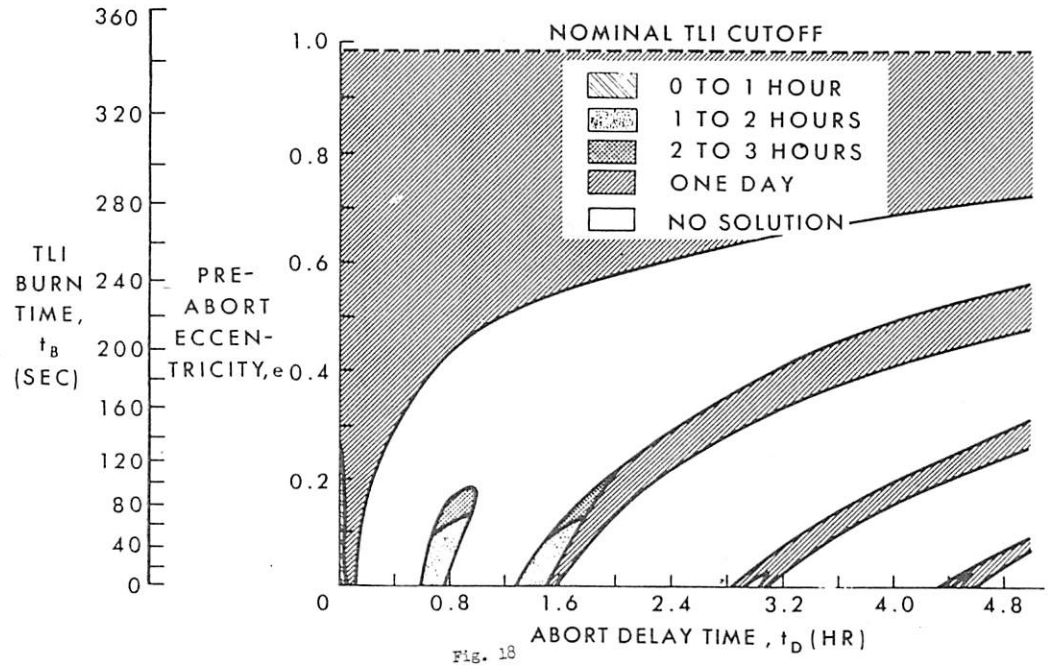


Fig. 18

NASA-S-66-5156 JUN

TRANSLUNAR COAST ABORT MODES

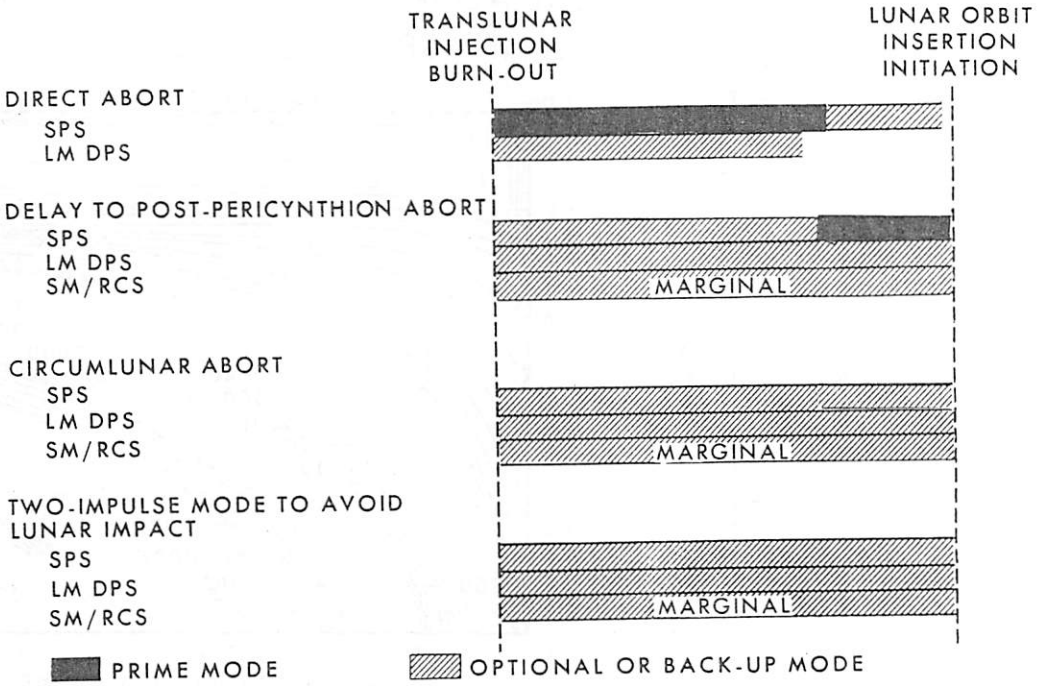


Fig. 19

NASA-S-66-4979 MAY 25

DIRECT RETURN MODE FOR TL COAST

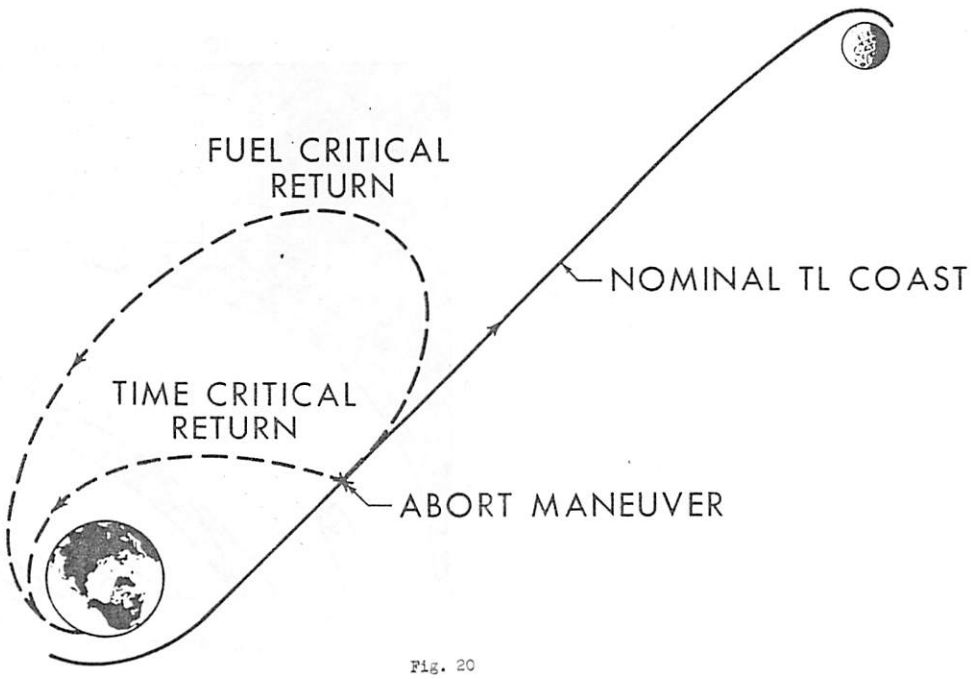


Fig. 20

NASA-S-66-4993 MAY 26

DELAY ABORT TO POST-PERICYNTHION MODE FOR TL COAST

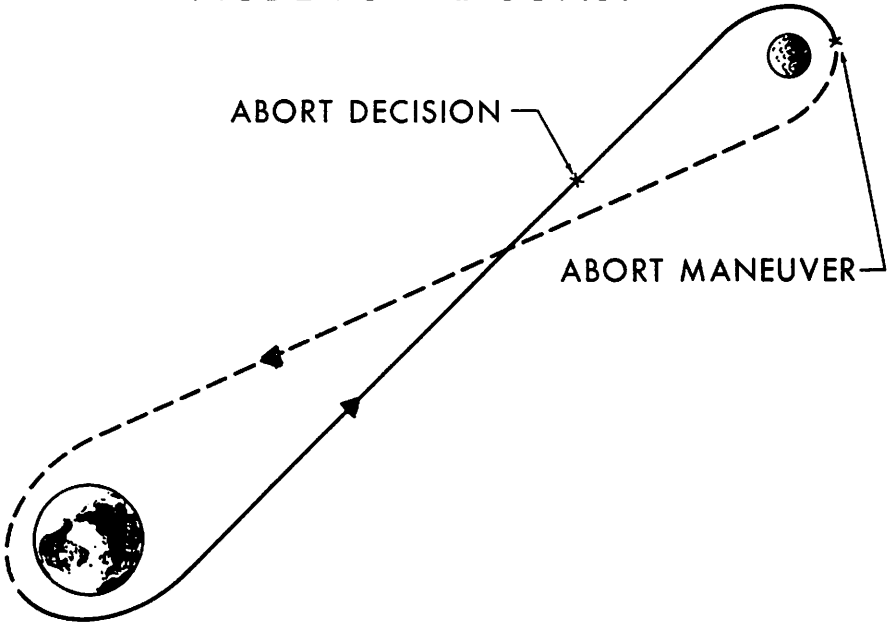


Fig. 21

NASA-S-66-4996 MAY 26

CIRCUMLUNAR ABORT MODE FOR TL COAST

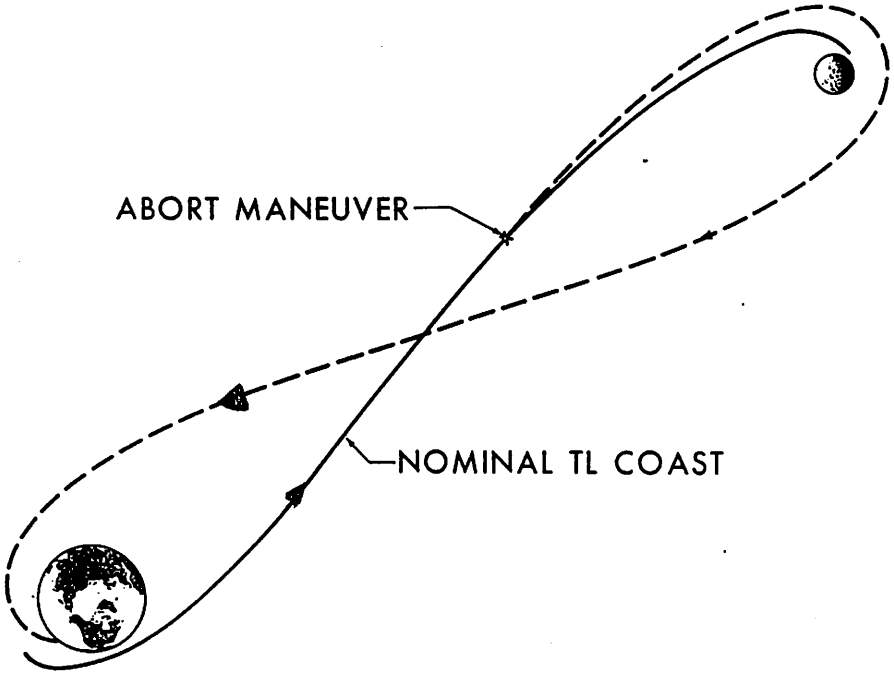


Fig. 22

NASA-S-66-5001 MAY 26

CORRECT BACK TO NOMINAL FREE RETURN MODE

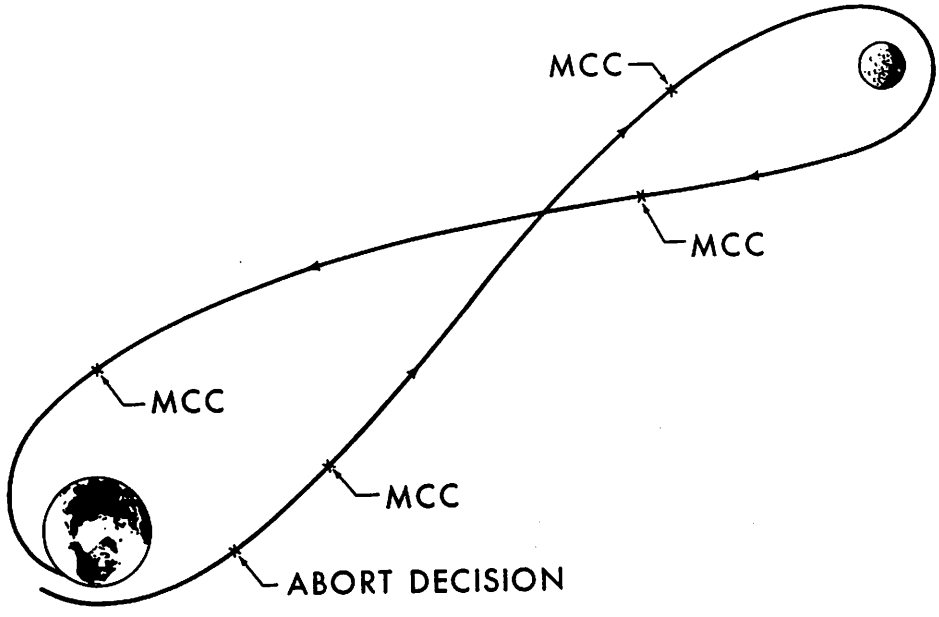


Fig. 23

NASA-S-66-5002 MAY 26

TWO-BURN MODE TO AVOID LUNAR IMPACT

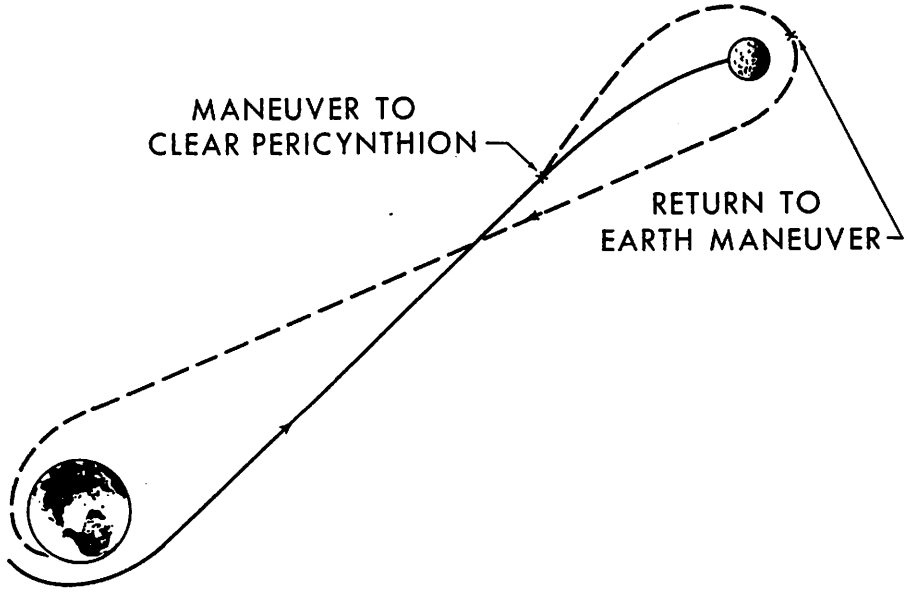


Fig. 24

NASA-S-66-4997 MAY 26

ABORT TRANSFER ANGLE FOR TL COAST

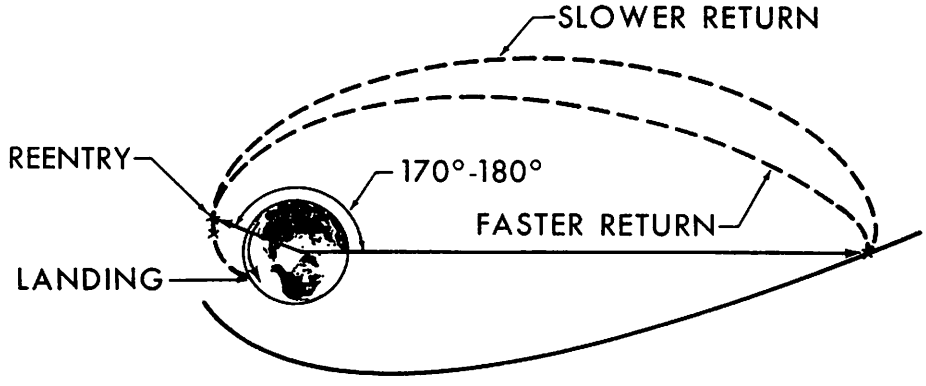


Fig. 25

NASA-S-66-4998 MAY 26

'24 HR' EFFECT FOR ABORTS TO SPECIFIC LANDING SITES

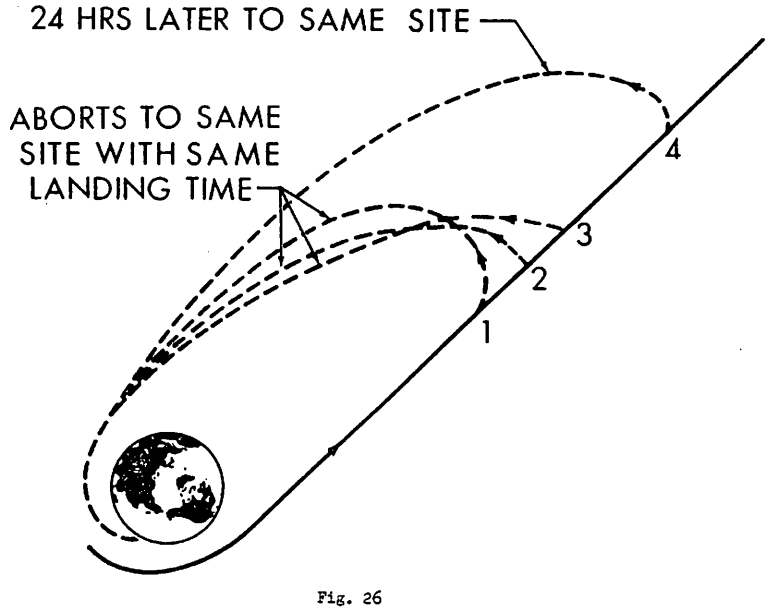


Fig. 26

NASA-S-66-6568 JUN

SPS ABORTS TO SPECIFIC SITES DURING TRANSLUNAR COAST

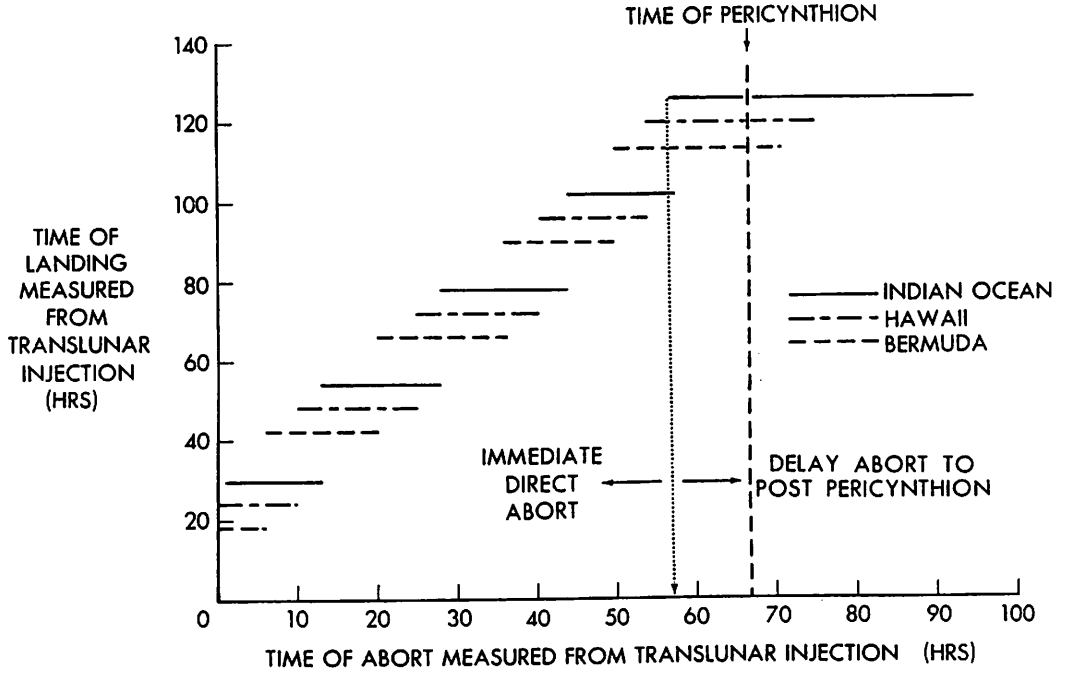


Fig. 27

NASA-S-66-6532 JUN

DPS ABORTS TO SPECIFIC SITES DURING TRANSLUNAR COAST

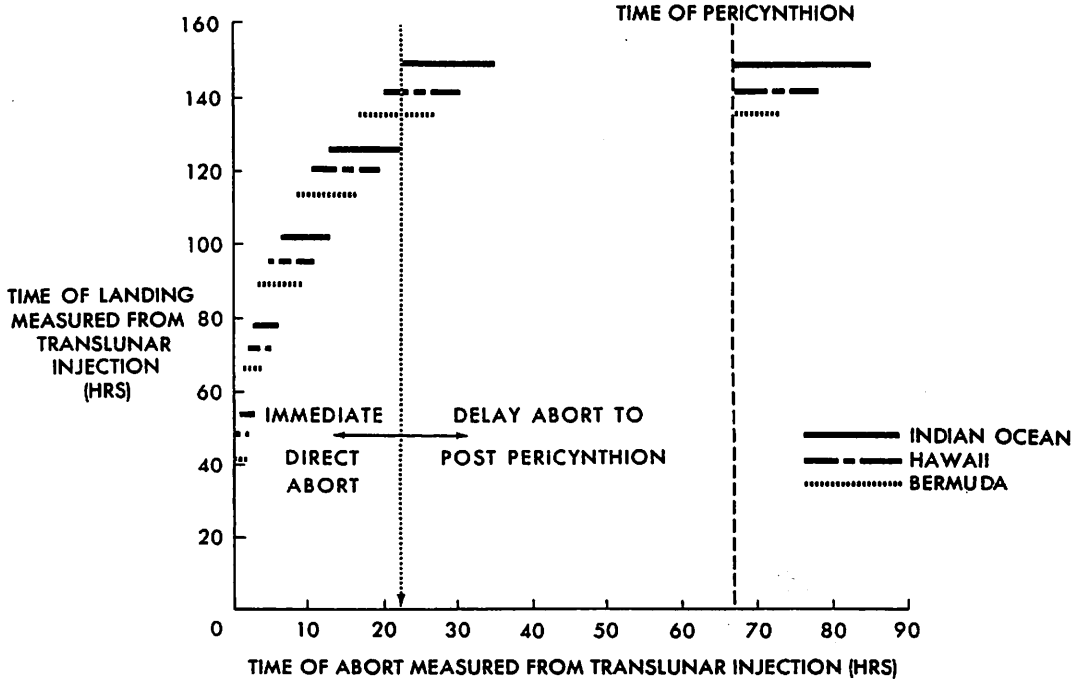


Fig. 28

NASA-S-66-6501 JUN

ABORT RETURN GEOMETRY

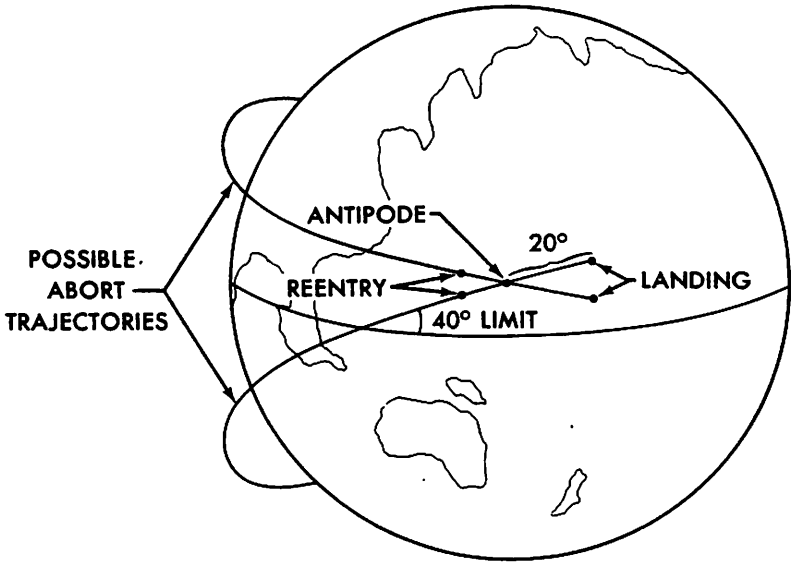


Fig. 29

NASA-S-66-4999 MAY 26

TYPICAL LANDING AREA ACCESSIBILITY FOR ABORTS FROM TL COAST

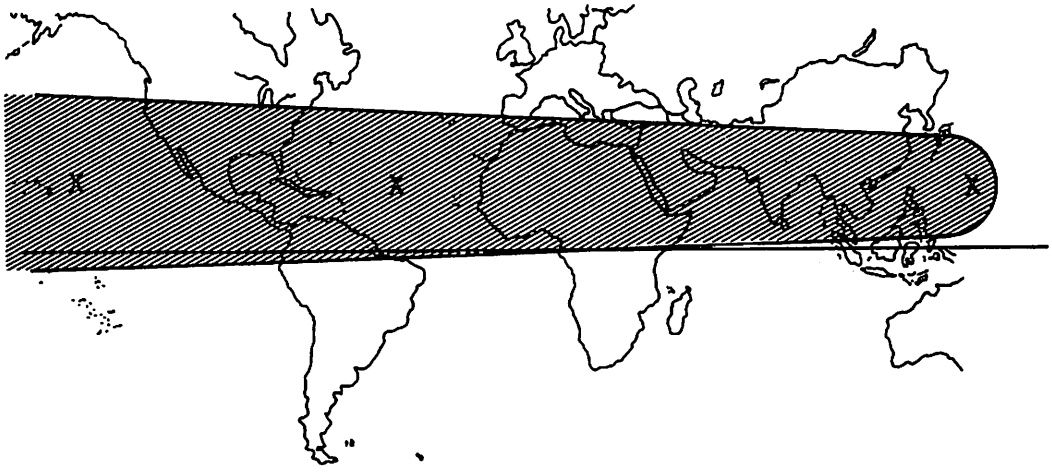


Fig. 30

NASA-S-66-4983 MAY 25

LOI PREMATURE SHUTDOWN TRAJECTORIES

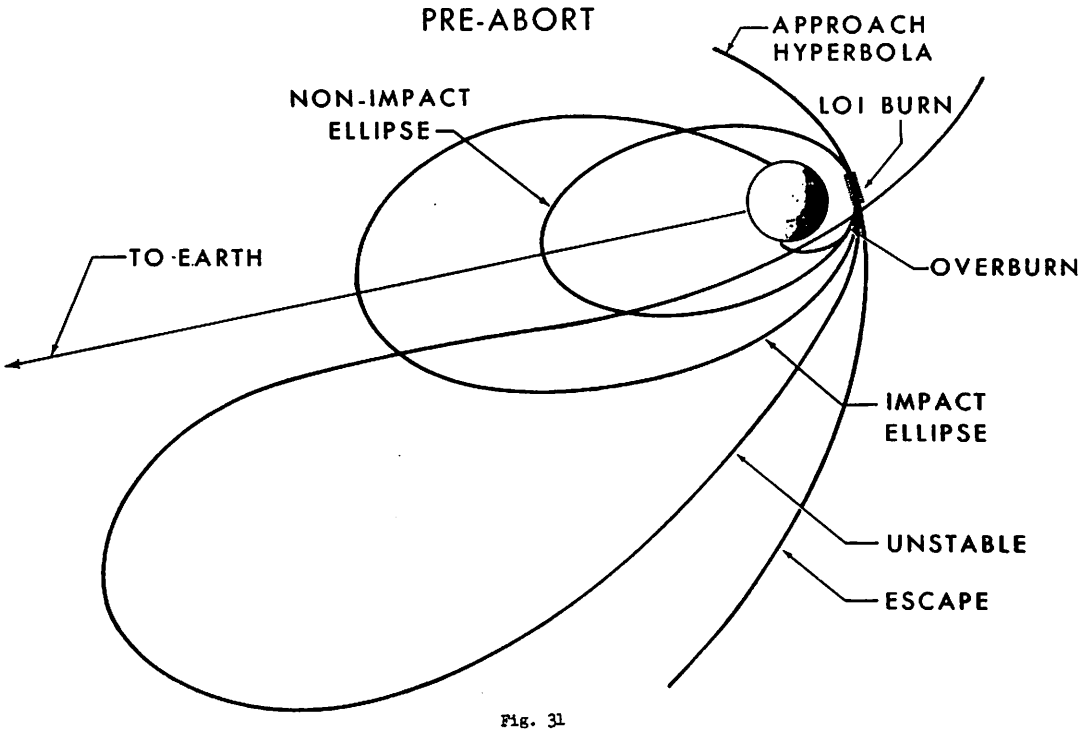


Fig. 31

NASA-S-66-6395 JUN

PRE-ABORT TRAJECTORY FAMILIES DURING LOI

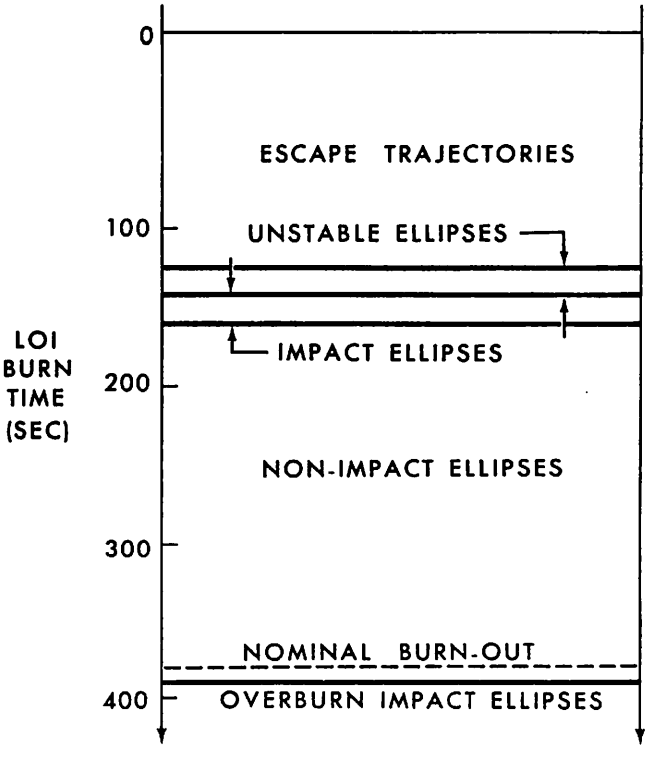


Fig. 32

NASA-S-66-6502 JUN

TYPES OF ABORT RETURNS FROM MOON - CENTERED ELLIPSE

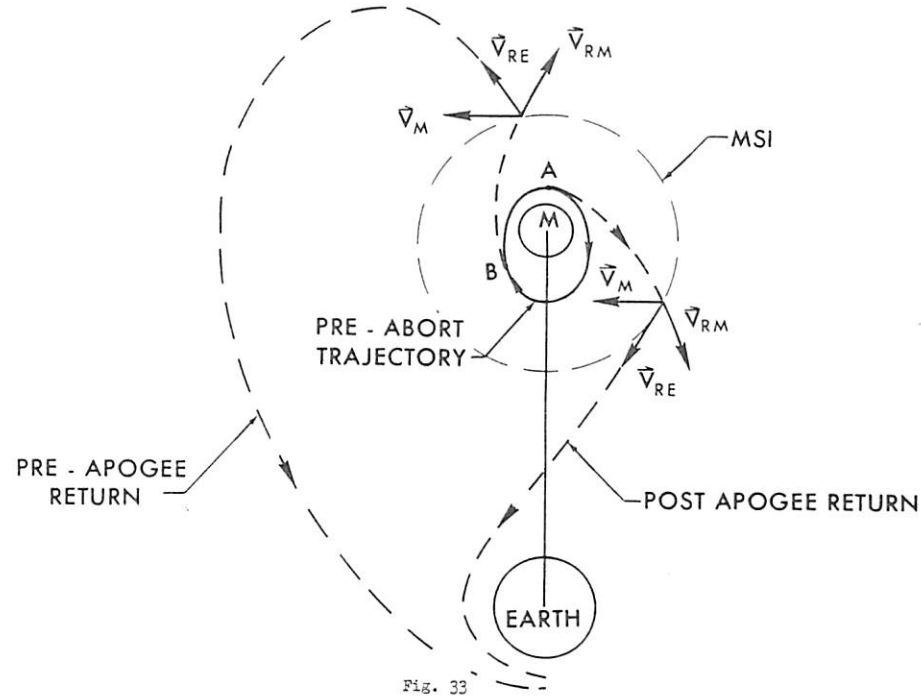


Fig. 33

NASA-S-66-5152 JUN 8

LUNAR ORBIT INSERTION ABORT MODES

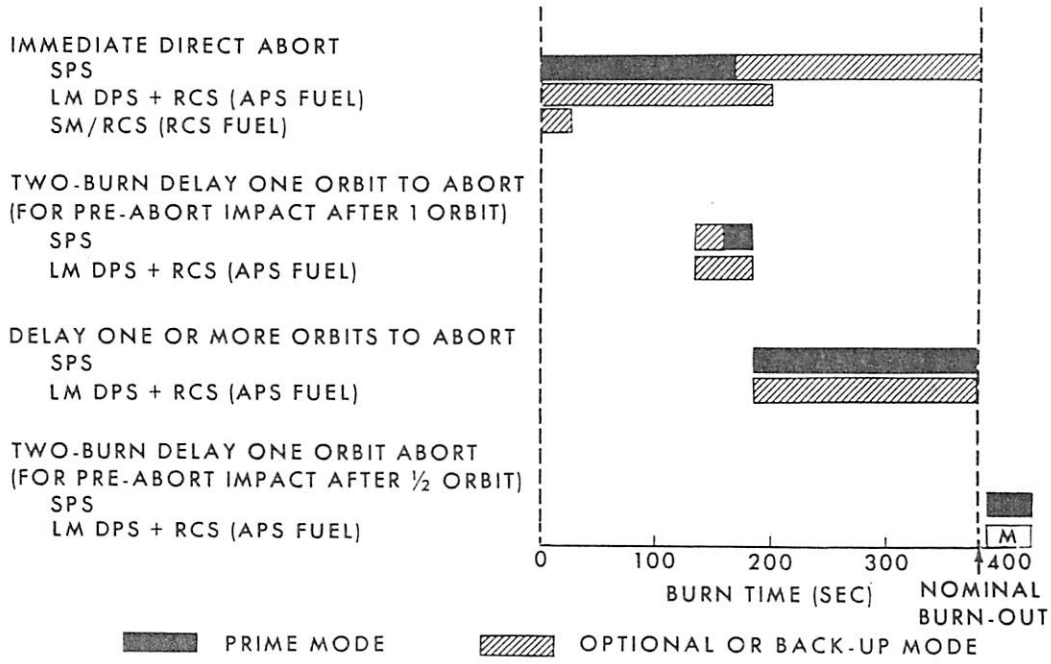


Fig. 34

NASA-S-66-4988 MAY 26

IMMEDIATE DIRECT ABORT MODE

MANDATORY FOR ESCAPE OR UNSTABLE PRE-ABORT TRAJECTORIES OPTIONAL FOR OTHERS

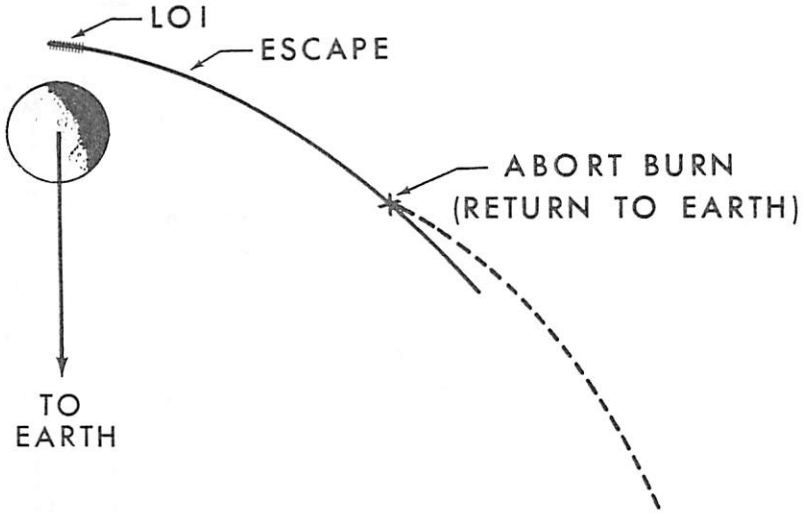


Fig. 35

NASA-S-66-4991 MAY 26

TWO-BURN DELAY ONE ORBIT MODE

FOR PRE-ABORT IMPACT ELLIPSES

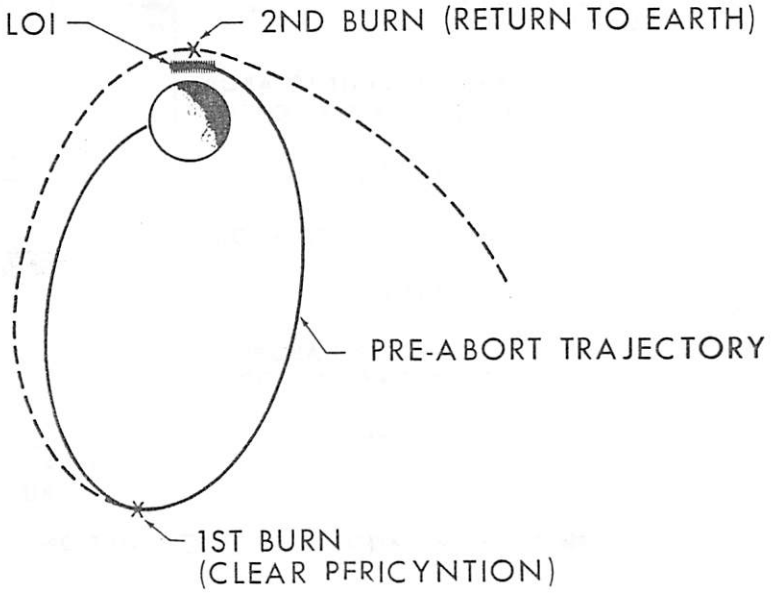


Fig. 36

NASA-S-66-4980 MAY 25

DELAY ONE ORBIT TO ABORT MODE

NON-IMPACT, SHORT PERIOD PRE-ABORT ELLIPSES

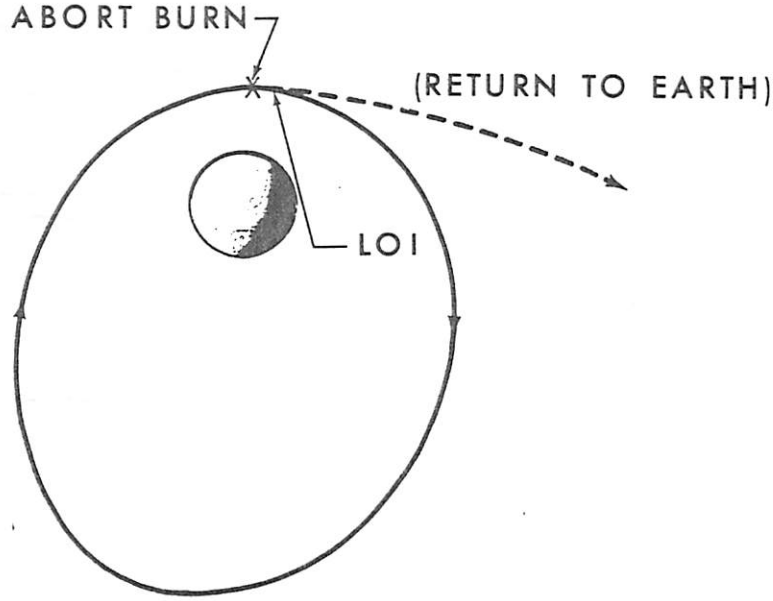


Fig. 37

NASA-S-66-4989 MAY 26

TWO-BURN DELAY ONE ORBIT MODE

OVERBURN CASES

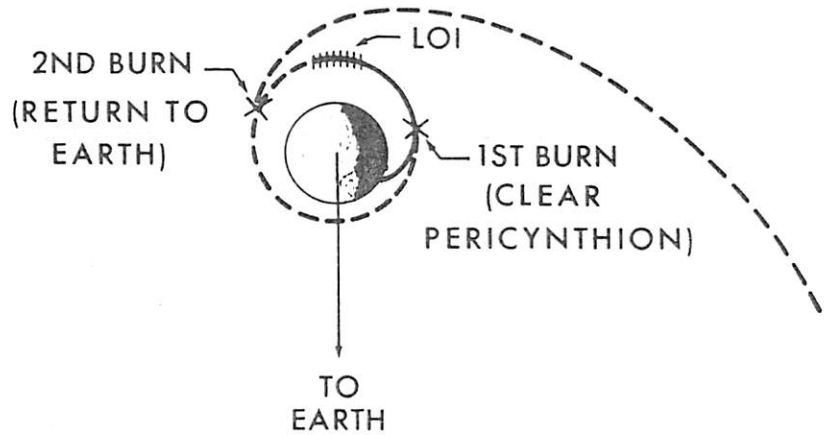


Fig. 38

NASA-S-66-6400 JUN

EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR PRE-ABORT ESCAPE TRAJECTORIES

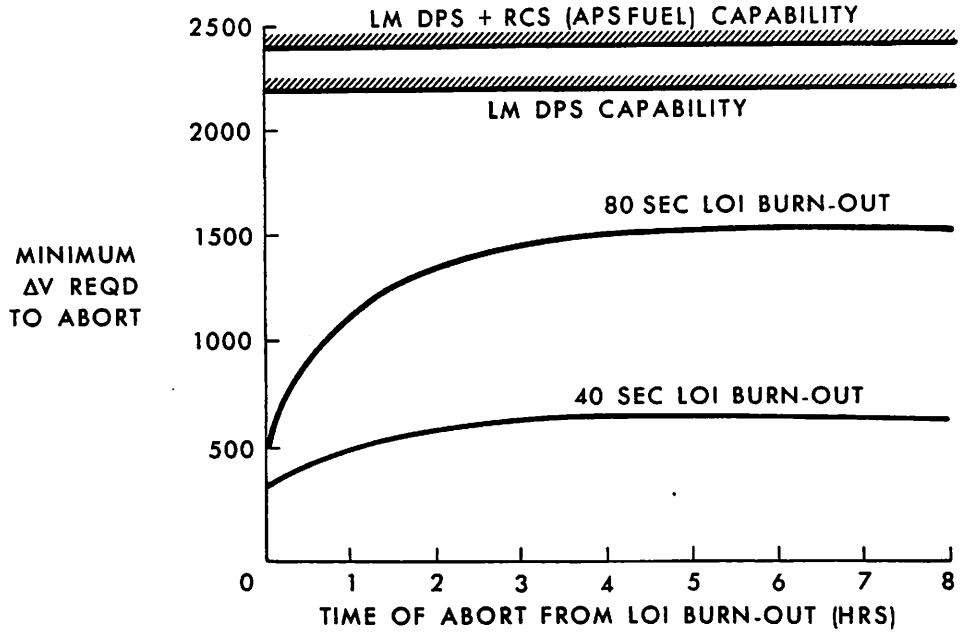


Fig. 39

NASA-S-66-6399 JUN

EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR PRE-ABORT ELLIPSE

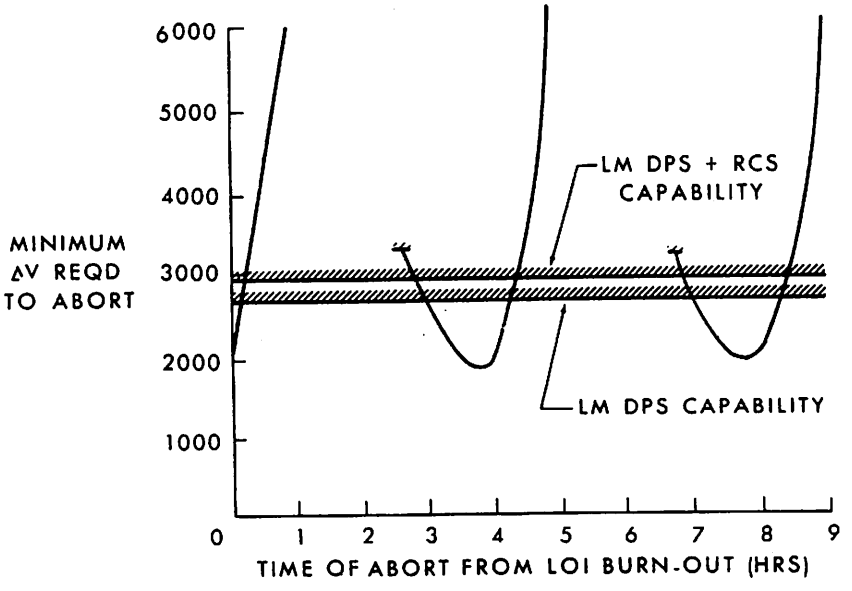


Fig. 40

NASA-S-66-6511 JUN

ABORT CAPABILITY USING LM PROPULSION SYSTEMS IN LUNAR ORBIT INSERTION PHASE

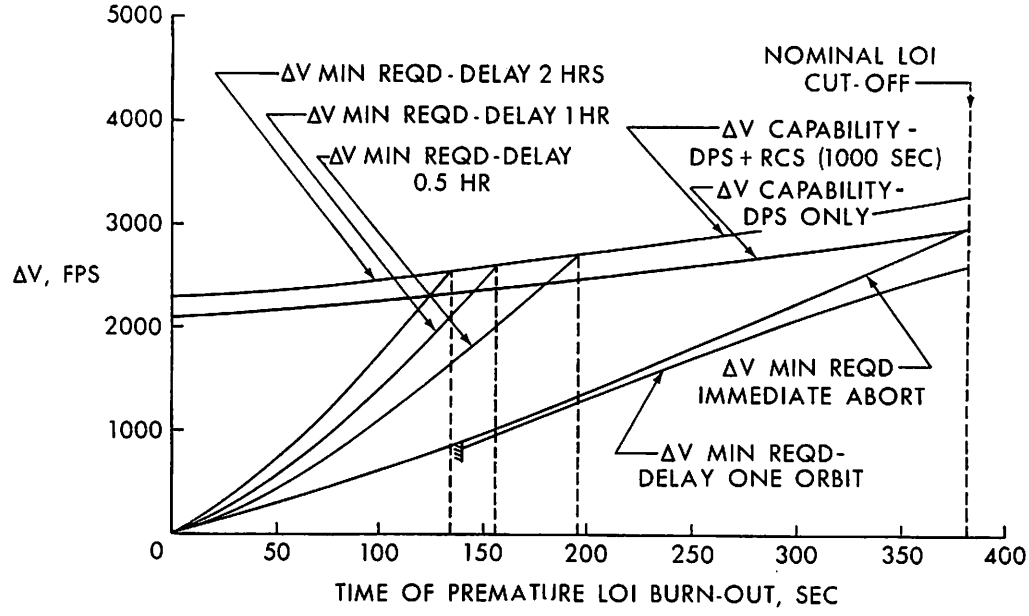


Fig. 41

NASA-S-66-6458 JUN

ABORTS TO SPECIFIC SITES FROM A PRE-ABORT ELLIPSE ABOUT THE MOON

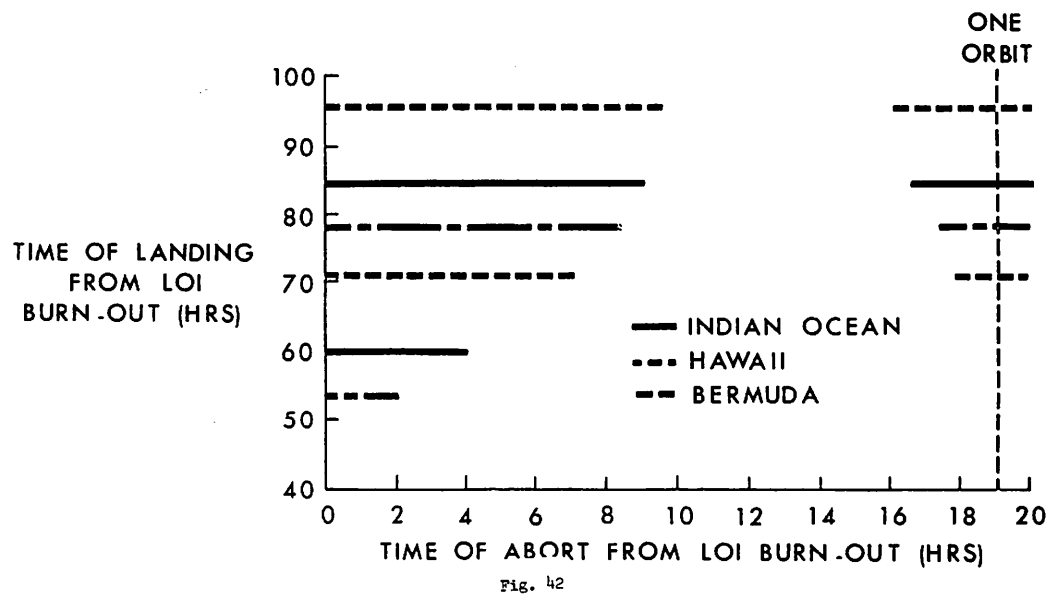


Fig. 42

NASA-S-66-5158 JUN 8

LUNAR ORBIT COAST ABORT MODES

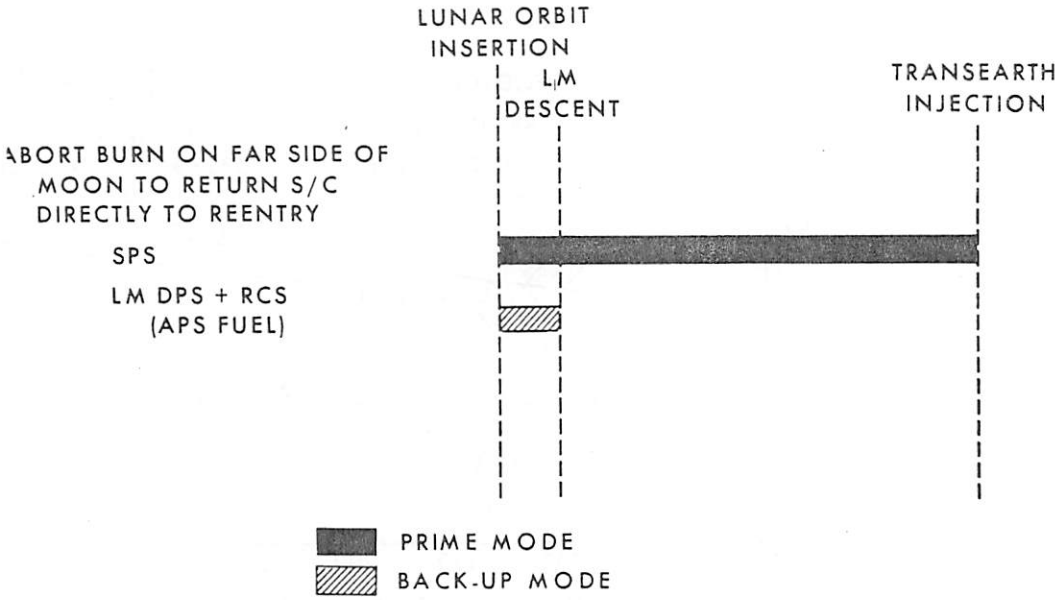


Fig. 43

NASA-S-66-4978 MAY 25

ABORT FROM LUNAR ORBIT COAST

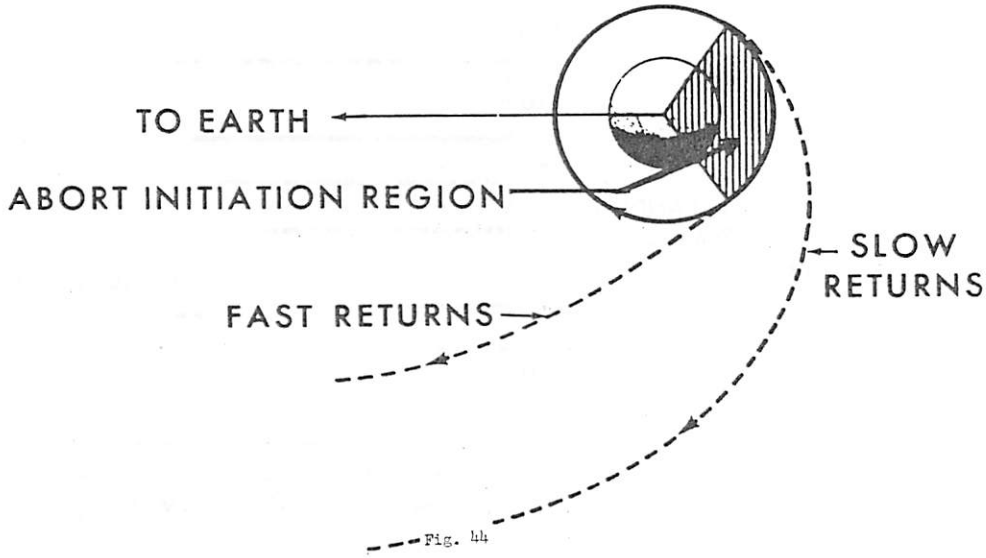


Fig. 44

NASA-S-66-6414 JUN

EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR LUNAR ORBIT COAST

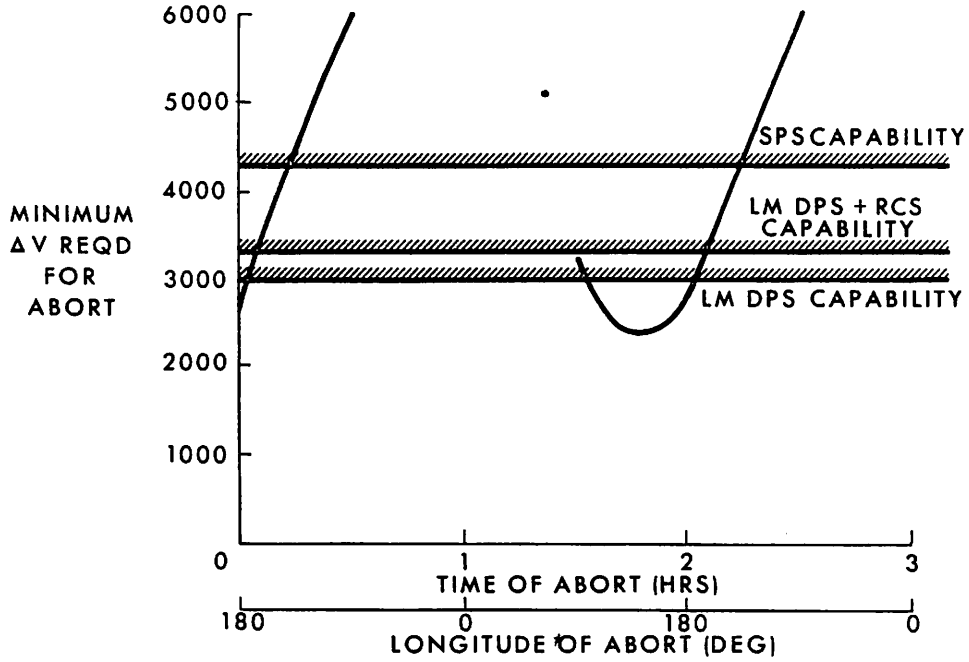


Fig. 45

NASA-S-66-4992 MAY 26

TEI PREMATURE SHUTDOWN TRAJECTORIES

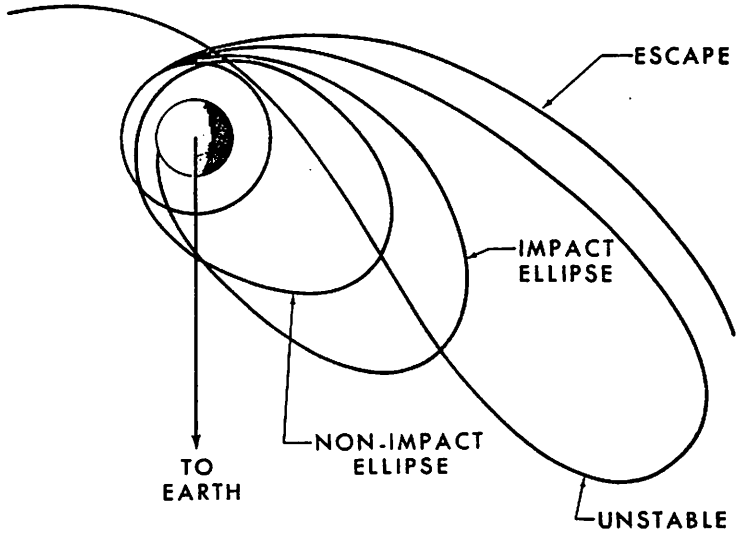


Fig. 46

NASA-S-66-6401 JUN

PRE-ABORT TRAJECTORY FAMILIES DURING TRANSEARTH INJECTION

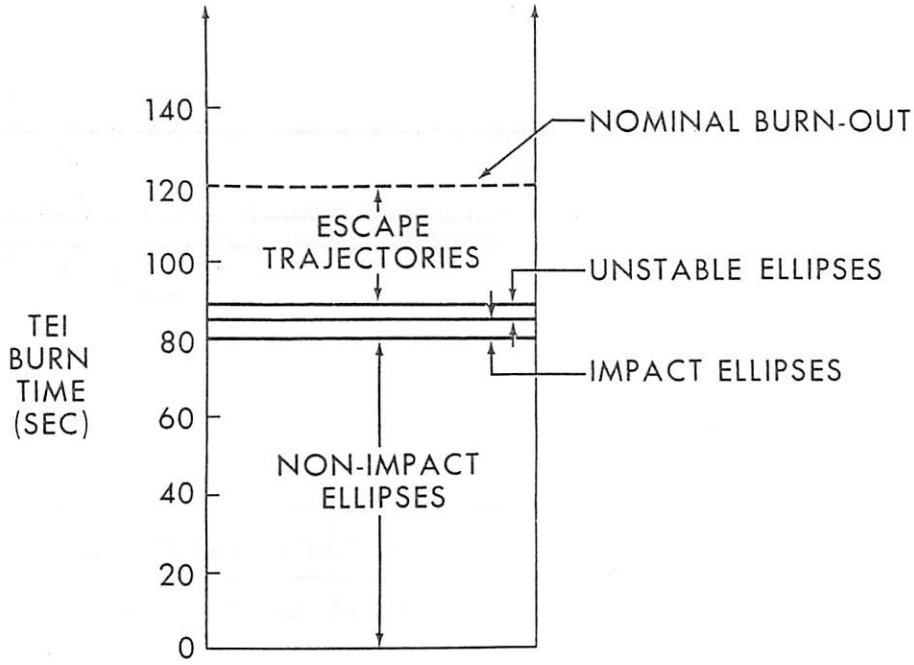


Fig. 47

NASA-S-66-5157 JUN 8

TRANSEARTH INJECTION ABORT MODES

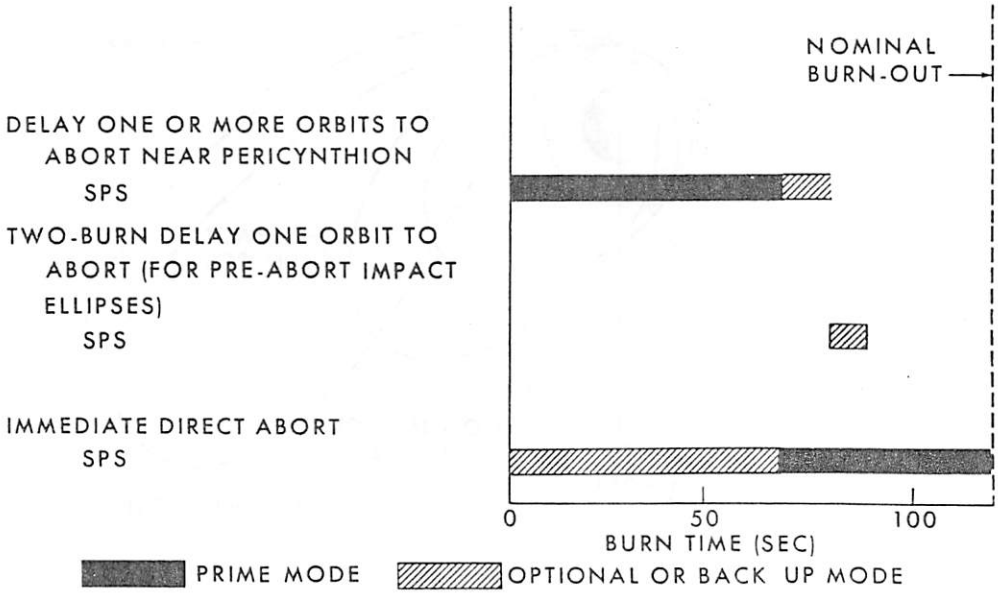


Fig. 48

NASA-S-66-6416 JUN

SPS ABORT CAPABILITIES DURING TRANSEARTH INJECTION PHASE

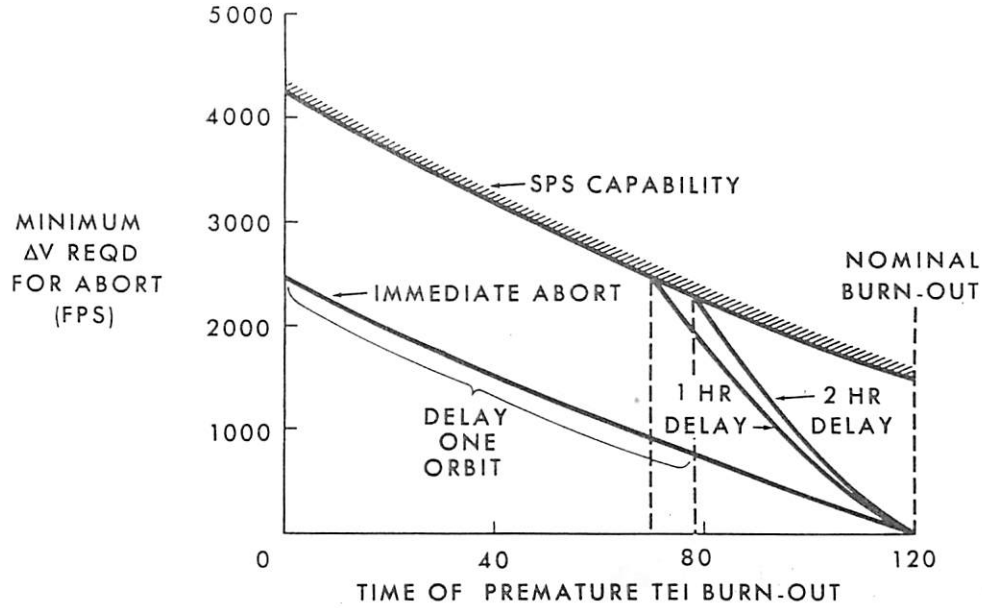


Fig. 49

NASA-S-66-5159 JUN 8

TRANSEARTH COAST ABORT MODES

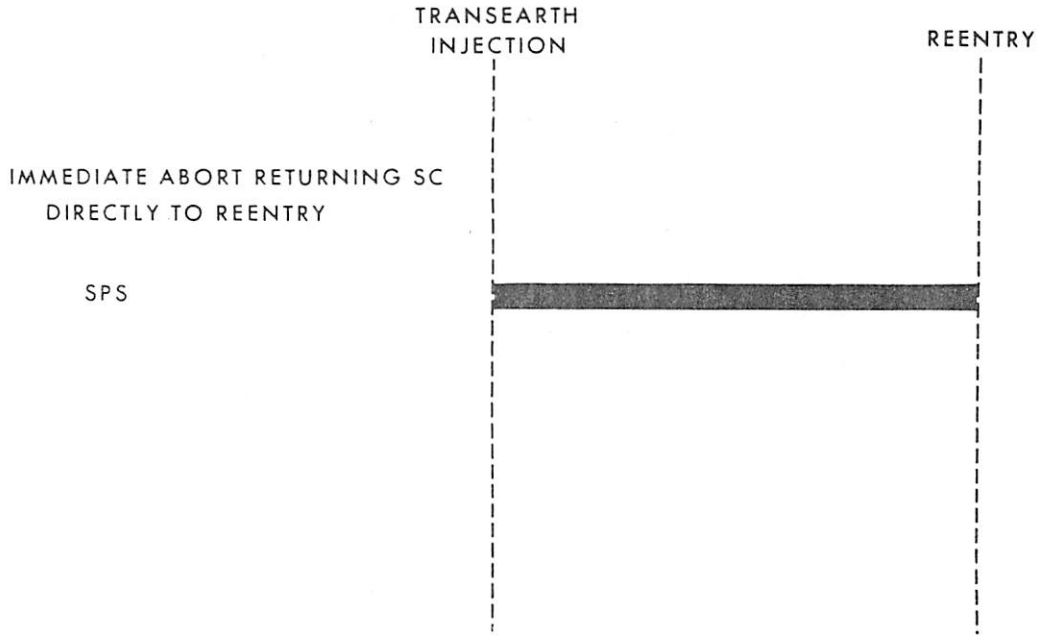


Fig. 50

NASA-S-66-4981 MAY 25

ABORT FROM TE COAST

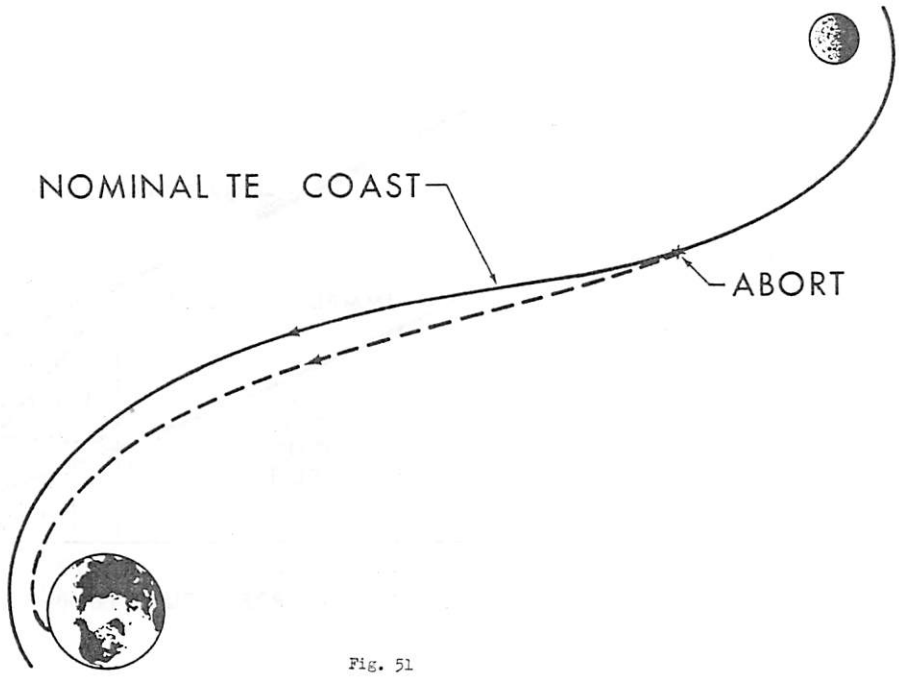


Fig. 51

NASA-S-66-6408 JUN

ABORTS TO SPECIFIC SITES DURING TRANSEARTH COAST

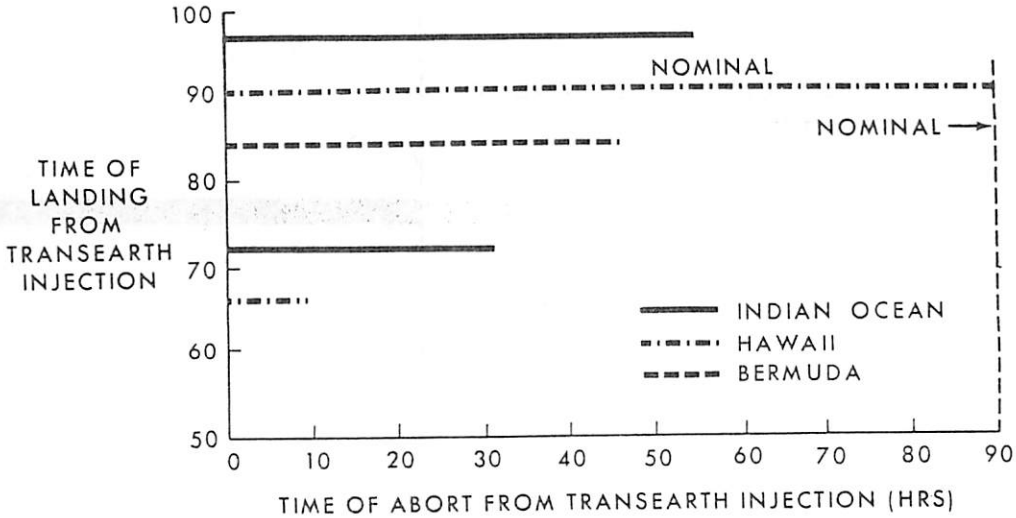


Fig. 52

NASA S 66 6477 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORT DURING EARTH PARKING ORBIT PHASE

PROPULSION SYSTEM	ΔV CAPABILITY
SPS (PRIME)	10,000fps
LM DPS	2000fps
LM DPS + RCS (APS FUEL)	2250fps
S-IVB	10,000fps +

Table
I

NASA-S-66-6436 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORT DURING TLI AND TL COAST PHASES

PROPULSION SYSTEM	ΔV CAPABILITY
SPS (PRIME)	10 000fps LM JETTISON 5300fps WITH LM
SM/RCS (RCS FUEL ONLY)	120fps LM JETTISON 90fps WITH LM
LM DPS	2000fps FULL SM
LM DPS + RCS (APS FUEL)	2250*fps FULL SM

*RCS BURN TIME LIMITED TO 1000 SEC
ASSUMES FOUR RCS THRUSTERS USED

Table
II

NASA-S-66-6475 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING LOI PHASE

PROPULSION SYSTEM	ΔV AVAILABLE
SPS (PRIME)	10 000 → 4300fps (LM JETTISON) 5300 → 2000fps (WITH LM)
LM DPS	2000 → 3000fps
*LM DPS + RCS (APS FUEL)	2250 → 3300fps

*ASSUMES DESCENT STAGING AFTER DPS BURN AND ALSO ASSUMES 1000 SEC LIMIT ON RCS THRUSTERS

Table III

NASA-S-66-6437 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING LO COAST PHASE

PROPULSION SYSTEM	ΔV CAPABILITY
SPS (PRIME)	4300 fps
LM DPS	3000 fps
*LM DPS + RCS(APS FUEL)	3300 fps

*ASSUMES DESCENT STAGING AND 1000 SEC LIMIT ON RCS THRUSTERS

Table IV

NASA-S-66-6496 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING TEI PHASE

PROPULSION SYSTEM	Δ V CAPABILITY
SPS (PRIME)	4300 → 1500fps

Table V

NASA-S-66-6444 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING TE COAST PHASE

PROPULSION SYSTEM	Δ V CAPABILITY
SPS (PRIME)	1500fps

Table VI

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

by

Morris V. Jenkins

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

The term software implies the programs essential to the onboard and ground computers necessary for guidance and navigation; it also takes into account the input and output interfaces, such as the mode of entry into the computer and the outputs driving the displays.

The points to be discussed in this paper are outlined below.

SOFTWARE CONSIDERATIONS (Figure 1)

The main consideration is to provide software adequacy in all mission phases to insure crew safety. The next consideration is to select programs which significantly enhance the probability of mission success.

In general, onboard adequacy insures crew safety, and in particular, attention is paid to an onboard independent capability to return.

Programs have been selected which significantly contribute to the probability of mission success. Two obvious examples are the programs which allow the spacecraft G&N system to provide an essential contribution to the capability to perform the translunar injection in the event of a Saturn inertial platform failure and another example is the program formulation to accept ground updates.

It is found in some cases that ground capability is essential.

All the time the ground capability is contributing to mission success with its active and monitoring guidance and navigation capability, together with the necessary provision of peripheral data.

Combined ground and onboard solutions further advance crew safety, provide an additional measure of optimization such as to produce the maximum probability of mission success.

The greatest factor contributing to our present day status of software capability can be attributed to the decision made in the initial stages of framing the G&N capability for Apollo. This decision was that of adopting the simple and reliable MIT targeting schemes. The principles of these schemes are being used, in many instances, both onboard and on the ground.

ADEQUACY OF ONBOARD SOFTWARE CAPABILITY (Figure 2)

The main features of the software which contribute to crew safety include the following:

1. An acceptable degree of independent capability to return.

2. The capability to accept ground updates either by the up-digital link in the case of the Command Module Computer (CMC), or by voice as in the case of the Lunar Module Guidance Computer (LGC) where the entry to the computer is made through crewman use of the Display Keyboard (DSKY).

3. Compatibility of software in the Prime Guidance and Navigation Control System (PGNCS) and the Abort Guidance System (AGS) and an obvious example of this is the equivalent targeting for the powered ascent.

4. The existence of a double check on the relative state between the Command Service Module (CSM) and the Lunar Module (LM) which is derived from the LM rendezvous radar observations of the CSM and the CSM sextant observations of the LM.

5. The LM can direct the CSM to do a concentric flight plan rendezvous, which is the preferred mode of rendezvous in non-time-critical situations. Briefly, this is achieved by the entry of the CSM state into the LGC, whereby the LGC will solve the concentric rendezvous solution for the CSM. Following this, the necessary information can be transmitted by voice to the CSM and the necessary targeting parameters are entered into the CSM DSKY by the crewman.

6. The software logic facilitates meaningful crew checks and this is achieved by the way the programs are framed such that data may be input in a form such that the crewman has an understanding of what he is doing and, in addition, the computer driven displays are also in meaningful form to the crewman. An example of this is where the crewman is allowed to input data referred to the local horizontal and the trajectory plane and, in addition, displays can be referred to the same reference.

The obvious examples of programs selected to promote mission success are:

1. The steering processors which constitute the main components of the "thrust programs."

2. The programs which allow the translunar injection to be accomplished in the event of a Saturn inertial platform failure. At a minimum, this capability will exist if the takeover decision is made prior to the burn.

3. The capability to accept alternative or modified targeting from the ground will facilitate the opportunity to use ground optimized solutions.

A summary of what makes these capabilities possible appears in the next section.

BASIC ONBOARD G&N SOFTWARE CAPABILITY (Figure 3)

The main factor in the capability of a manned vehicle to achieve its destination is the crew G&N management. This management is possible through the monitoring of computer driven displays. A familiar pattern of G&N management questions may still be recognized. In their basic form they are:

1. Where are we?

2. Where will we be?

3. What change of course is needed?

4. Is the prime G&N system controlling the propulsion correctly?

The corresponding respective activities are commonly recognized as:

1. Navigation

2. Dead reckoning

3. Change of course determination

4. Maneuver control

In current software terms, these same activities are respectively recognized as:

1. Trajectory determination processors

2. Trajectory prediction processors

3. Pre-thrust programs

4. Thrust programs

The function of the "trajectory determination processors" is to accept observations such as the range and range rate of the rendezvous radar, and to formulate a smoothed solution wherein the current position and velocity is determined. This is essentially achieved by the equivalent of fitting a trajectory in a least squares manner such as to minimize the residuals.

The function of the "trajectory prediction processors" is to accept a state vector from the trajectory determination process and with the ephemeris information available as an input. The trajectory prediction processors will produce the best estimate of position and velocity at any future required time, provided that free coast conditions exist. The ephemeris data includes oblateness terms and the position and magnitude of other perturbing gravitational sources, such as the moon if we are in a near earth phase.

The main function of the "pre-thrust programs" is to determine the targeting parameters for the "thrust programs."

The function of the "thrust programs," the main components of which are the steering equations, is of course to achieve desired cutoff conditions corresponding to a required trajectory condition.

Some amplification of this discussion occurs in the next section.

AMPLIFICATION OF THE BASIC PROGRAMS (Figure 4)

The main component of the "trajectory determination processors" is the "Kalman filter." The Kalman filter is essentially an efficient way of writing least squares fit equations. It is particularly adaptable to the processing of one observation at a time. In contrast, on the ground, data is processed in batches. This is for the convenience of manual editing and the detection and exclusion of a bad data source. In the case of the ground, the Bayes least squares method has been adopted.

The "trajectory prediction processors" have as their main component the Encke method. The stratagem that this method uses is to use the equations of motion in incremental form. The incremental form is achieved by subtracting the basic two body equation, i.e., the point mass - particle solution, away from the total equation of motion. Because the equations of motion have been reduced to incremental form in this manner, the integration routine can take larger steps for the same accuracy that would be accomplished by an integration of total quantities. Prolonged, extensive tests were carried out at Manned Spacecraft Center to verify this assertion.

The function of the "pre-thrust programs" is threefold: (1) to determine the targeting and put it in a form which the "thrust programs" will accept; (2) to determine the initial thrust command attitude; and (3) to determine the preferred platform alignment.

The function of the "thrust programs" is to provide state input, i.e., position and velocity data, to the steering laws which in turn will determine what the appropriate command quantities are. In general, the command quantities will determine the attitude and duration of the thrust. The steering laws normally strive to null out the discrepancy between command and response. The commonest form of steering used by the onboard programs is "X product steering." This is an old MIT steering law in which the velocity to be gained is implicitly constrained to be inertial during a continuing thrusting process, while the scalar value of it is shrunk to acceptable limits, approximately zero. "E guidance" is used for LM descent and ascent where additional control characteristics are required to fit the event peculiar to these special maneuvers. Another form of thrust which will be formed is where the thrust line is controlled in direction relative to "local vertical coordinates."

TARGETING

As was mentioned in the section on Software Considerations, the greatest factor contributing to our present day status of software capability can be attributed to the decision made in the initial stages of framing the G&N capability for Apollo. This decision was that of adopting the simple and reliable MIT targeting schemes.

In order that an insight into the potential of adopting these principles of targeting, together with a realization of the

flexibility that exists for the current stage of program formulation, all the major types of targeting are discussed, at least conceptually. It should not be assumed at this time, in terms of specifics, that all of these targeting forms will be employed on the lunar landing mission. However, minor modifications of the stratagems used will be employed for the lunar landing mission, both for onboard and ground G&N systems.

TARGETING - TYPE 1 - Translunar Injection (Figure 5)

The essential ingredients of this type of targeting are: (1) utilization of current $\vec{r}$; (2) specification of target vector; and (3) specification of energy. A sensitive parameter in the control of lunar trajectories is energy. Therefore, it is not surprising that this form of targeting was selected for "translunar injection."

Corresponding to the optimized lunar trajectory flight plan, there is a corresponding two body solution generated by the same $\vec{r}$ and $\vec{V}$ vector at the instance of translunar injection. The equivalent two body solution assuming a point mass gravity field only is commonly called the osculating conic, in this case an ellipse. The steering law is framed such that it is only cognizant of the two body situation. An $\overline{r}_t$ is selected on the osculating ellipse, a specified energy is stipulated for the ellipse, and this is achieved by specifying the semi major axis a. This, together with the current burn $\vec{r}$ instantaneously specifies the two body trajectory. This is done in a continuum of instances during a continuous burn wherein the velocity required and its derivative with time is continuously known such that cross product steering may be applied. The target $\overline{r}_t$ on the osculating ellipse is sometimes known as the phantom target. The additional perturbation gravity effects will cause the spacecraft to drift on to the actual trajectory which will go around the moon.

TARGETING - TYPE 1 - Lunar Orbit Insertion and Repeat for Plane Change (Figure 6)

In this instance the orbit is being controlled around the moon and the period of it is a parameter to control, but this is equivalent to saying the energy must be controlled, i.e., the semi major axis a will be specified. As will be guessed by the fact that we are classifying this scheme again as Targeting - Type 1, a target vector $\overline{r}_t$ must be

specified. However, the latter is done in a special way in order to insure circular conditions. The scalar value, $|r_T|$, of the vector $\bar{r}_t$ is specified to be the same length as the current burn $|r|$ and the semi major axis a is continuously commanded to be the same value as $|r_T|$ which in turn is the value of the current burn $|r|$.

<u>TARGETING - TYPE 2</u> - First Midcourse Correction and Initiation of Circumlunar Abort (Figure 7)

The principles used in this type of targeting are as follows:

1. Use an accurate trajectory predictor to determine what the state vector will be corresponding to some particular event in the future. The particular event could be specified by time or when pericynthion occurs, for example.

2. A two body conic trajectory solution is found which will satisfy the current state position vector and the future state. Then another body conic trajectory solution is found to satisfy the current state position vector and the future state.

3. The velocity vectorial difference between the two two body conic solutions is noted.

4. The velocity vectorial difference of the conics is assumed to be the precise velocity to be gained vector. It would be if the conic solutions were perfect simulations of trajectories going from the current state position to the future states and indeed because the differential of the conics is taken, the inaccuracy due to using imperfect simulations is almost completely nulled.

5. The vector velocity differential of the conics is added vectorially to the precise current state as determined by the "trajectory determination process" and the resultant is accepted as the precise vectorial velocity required.

This is the basis of the determination of Targeting - Type 2.

It is appropriate now to consider how these principles are applied to the "first midcourse correction." The original preflight optimized flight plan has on it a point where the translunar trajectory pierces the sphere of influence and the time (T_2) of the event is known. Let it be supposed

that the orbit determination process indicates a dispersed state subsequent to translunar injection and transposition and docking. At time T_1 it is decided to apply the first midcourse correction. The desired future state position at time T_2, as has already been mentioned, is known. The trajectory determination processors are used to find out where the spacecraft will actually be by continuing free coast until time T_2. Now by referring to Figure 7, it will be seen how the above described principles of Type 2 can now be applied to the circumstances at hand.

The first midcourse correction capability may also be used for the "initiation of circumlunar abort."

TARGETING - TYPE 2 - Second and Third Midcourse Corrections and Circumlunar Abort (Figure 8

It is now convenient to examine the circumstances for the second and third midcourse corrections. They are assumed to occur within the lunar sphere of influence and, hence, it is advisable to reference position and velocity to a lunar set of coordinates. It may be assumed that the stratagems used for the second midcourse correction will be used for the third midcourse correction. Let it be supposed that the trajectory position processors have determined that our current state within the sphere of influence will propagate to an unacceptable dispersion prior to lunar orbit insertion. More precisely, the activity would be as follows:

1. The trajectory determination processor predicts precisely ahead to find out where the "propagated pericynthion" will be. For simplification, let it be said that for fuel economy reasons, that the lunar orbit insertion must occur at a pericynthion point and therefore it is mandatory that this pericynthion point must be in the predetermined "desired orbit plane" which has been selected to be consistent with an overall optimization of flight plans including LM landing and subsequent rendezvous. However, when the current "propagated pericynthion" is determined, it is noted that it is not at the required altitude and furthermore, it is not in the "desired orbit plane."

2. The first action is to shrink the scalar value of the "propagated pericynthion" vector.

3. The next action is to rotate the shrunken "propagated pericynthion" vector into the "desired orbit plane." This is done by rotating about the current state position vector $\vec{r}$ holding the angle between this vector and the "propagated pericynthion" vector constant. In this way the "desired pericynthion" is formed.

Hence, the situation in which Targeting - Type 2 can be applied may be recognized. The original "propagated pericynthion" constitutes the future state generated by the "predicted actual" and the newly formed "desired pericynthion" constitutes the future desired state and, hence, two, two body hyperbolic conic solutions can be applied to find the required differential and thus the principles of Targeting - Type 2 are being applied. All of the above can be fairly easily followed by making reference to Figure 8.

The completion of the translunar leg is essential to a circumlunar abort capability and the principles of the "second and third midcourse corrections" can also be applied to "circumlunar abort."

<u>TARGETING - TYPE 3</u> - Transearth Injection, Transearth Abort, Transearth Midcourse (Figure 9)

Targeting - Type 3 is the targeting scheme found in the MIT "return to earth" program. A modified form of this program will be used by the ground computers for determining maneuvers corresponding to the mission phases indicated in the title of this section.

Basically, the required solution is scanned by approximating the return to earth trajectory with a hyperbola joined to an ellipse at the sphere of influence. This is then followed by another iteration, this time in a precision mode.

For convenience, "Targeting - Type 3" can be identified with the initial conic iteration mode wherein there is a matching of conics at the sphere of influence.

<u>Conic Iteration</u>

The main steps of the conic iteration mode are as follows:

1. An initial guess of a suitable juncture point on the sphere of influence (S of I). Typically this is done by taking a point 45° center angle from the moon in the

current hyperbolic trajectory plane and also meeting the condition that it is on the S of I. Thus, the initial sphere of influence point (S of I point) is identified.

2. A parabolic trajectory is run from this point back to the earth maintaining the same plane at the S of I.

3. In general, it will be found that the earth has not conveniently rotated the recovery point into the plane of the parabolic trajectory.

4. A transit time is now selected corresponding to an entry speed at 400,000 ft earth altitude less than parabolic.

5. By successive selection of transit times, a transit time will be found when the recovery point has conveniently rotated into the plane of the trajectory. Up and downrange requirements for the recovery point are taken care of by the freedom of "reentry maneuver range."

6. Unfortunately, the required total solution has not been achieved even in conic form. Tentatively the near earth conic trajectory has been found; however, it must pass a subsequent step which is found in the next step.

7. At the S of I point, the earth dominated trajectory will have an energy appropriate to the potential and kinetic energy referred to the earth coordinate system. The joining hyperbola at the S of I point must have a compatible energy when the conditions at the join are referenced to the moon coordinate system. Furthermore, the terminal velocity vector of the hyperbola at the S of I point must, when earth referenced, be in the same straight line as the terminal velocity vector of the ellipse at the S of I point.

8. When the conditions of the previous step have been satisfied at the S of I point, then the backtracked hyperbolic solution from the S of I point is explored to determine whether it intersects the desired "departure vector."

9. If the backtracked hyperbolic solution does not intersect the desired "departure vector" then a simultaneous iteration proceeds, in which the S of I point is matched and other Δt's of the earth dominated eliptical trajectory are carried out in a rational manner to achieve convergence on

all required conditions for the "conic iteration." The required conditions may be summarized as: a correct matching of the conics at the finally selected S of I point; the recovery point must have rotated into the plane of the earth dominated elliptical trajectory at the time of entry; and the backtracked hyperbola must intersect the desired "departure vector." When this is done the conic iteration is at an end; however, this solution is not sufficiently accurate, and it is necessary to proceed to a "precision iteration."

Precision Iteration

In this mode, a precision trajectory found by numerical integration is determined which will go from the desired "departure vector" to the recovery point. The steps to accomplish this are:

1. A precision trajectory is back integrated from the last found conic reentry condition and the pierce point at the S of I is noted.

2. A precision forward integration from the "departure vector" is done in order to note the corresponding pierce point at the S of I.

3. In general, it will be noted that there is a mis-match at the S of I.

4. A simultaneous "precision iteration," using the derivatives found in the conic iteration, must now proceed to achieve a match. This is achieved by again adjusting the S of I point and transit time, Δt, of the earth dominated trajectory.

5. Finally, all the necessary end conditions indicated in the "conic iteration" description are achieved for the "precision iteration."

6. When this has been done, then the necessary osculating hyperbolic conic corresponding to the established "departure vector" and the determined ΔV vector for it is established for targeting purposes.

In detailed specifics, the above may not be exactly correct; however, all the above principles are employed in intended "return to earth" programs for both onboard and ground computers.

DESCENT AND ASCENT TARGETING (Figure 10)

In the descent and ascent targeting additional conditions are required beyond that which could be achieved with MIT cross product steering. So called E steering, also developed at MIT, is used to achieve conditions required during descent and ascent. Briefly, the command thrust vector is derived from explicit expressions involving the differentials of current and target conditions, and the gains in the steering laws are influenced by the ΔT_{GO}. The referenced diagram indicated the conditions required by descent and ascent targeting. No further comments will be made in this discussion, as this would incur a level of detail not intended for this discussion.

CONCENTRIC FLIGHT PLAN (Figure 11)

The targeting for the concentric sequence is discussed in this section. A prerequisite of getting exactly nominal flight plan conditions is that the standard launch routine is initiated at the correct time. The concentric sequence objectives per se are are not identical with achievement of absolutely nominal flight plan conditions. For example, it is not an objective of the concentric sequence to achieve precisely a 15 mi differential altitude at terminal phase initiation, which might well be an objective of a reference nominal flight plan. Late liftoff and subsequent dispersions will prohibit absolute achievement of the reference flight plan conditions. The concentric sequence retains two main objectives, one is the establishment of a Constant Differential Height coast (CDH) prior to the Terminal Phase Initiation (TPI). The other objective is that TPI should occur at a particular time. The latter is a requirement aimed at achieving TPI such that mission planning purposes may be satisfied, e.g., the line of sight relationship with the sun direction. The steps taken to achieve the concentric sequence objectives are as follows:

1. The first thing is to determine the Concentric Sequence Initiation (CSI). This is done on a basis of instantaneous impulses simulating the CSI and CDH burns.

2. The CSI is specified as a horizontal burn and the next steps will indicate how its scalar value is determined in order to achieve the objectives of the concentric sequence.

3. An iteration is initiated by selecting a tentative scalar value of the CSI maneuver.

4. The resultant orbit is examined to determine: the position of the apsis point; the transit time (Δt_1) to achieve it; and the differential height of the apsis point from that of the target vehicle orbit opposite the apsis point.

5. The phase lead of the target vehicle at the time of the CDH maneuver, which occurs at the apsis point, is noted. In addition, the phase lead of the phase lead required at TPI is determined simply on a linear proportional basis, i.e., phase lead $\phi = K_1.(\Delta h)$, where K_1 is a known constant. Thus the net central angle catch up, i.e., net phase angle catch up, is known. This is a property of the concentric flight plan. Furthermore, the catch up rate is determined on a linear proportional basis, i.e., $\dot{\phi} = K_2.(\Delta h)$, where K_2 is a known constant. This is another property of the concentric flight plan, which simplifies the iterative solution.

6. Now if the net $\Delta \phi$ between CDH and TPI is divided by $\dot{\phi}$, then the Δt_2 between CDH and TPI is established since $\Delta t_2 = \Delta \phi / \dot{\phi}$.

7. Now it must be determined if $\Delta t + \Delta t_2$ = TPI time (T_2) - CSI time (T_o).

8. If the above condition is found not to be the case, then the whole process must be repeated and a new exploratory scalar value of the CSI maneuver is tried to initiate the second iteration. Following this iteration, a partial of ΔV of CSI with respect to time error at the desired time error referred to the desired TPI time (T_2), can be formed and used to converge with the third iteration which is initiated now with a calculated ΔV for CSI. It is just possible that even this third iteration has not achieved the objectives of the concentric sequence within acceptable limits; however, a highly convergent state exists and the objectives will surely be achieved with the next iteration.

With the required instantaneous ΔV established for CSI, the target orbit conditions for CSI are established such that a steering law can be applied to facilitate a finite burn. The final apsis point determination establishes the time at which the CDH maneuver will occur and in most cases, this will not be altered even with trajectory determination updating subsequent to the CSI maneuver.

The steps taken to establish the targeting of the CDH maneuver are as follows:

1. The value of $\dot{r}$ of the target vehicle orbit opposite the CDH apsis point is determined.

2. A requirement of the CDH maneuver cutoff is that it will have the same value of $\dot{r}$ as determined in the previous step.

3. In addition, the semi major axis a of the resultant orbit following the CDH cutoff is determined by subtracting (Δh) from the scalar value of r of the target vehicle orbit opposite the CDH apsis point. The last two steps describe the necessary cutoff conditions for the CDH maneuver and they will minimize the variation in differential height subsequent to the CDH maneuver. If the target vehicle orbit is circular, it is plain to see that the resultant orbit following the CDH maneuver will be concentric. It is from this property that the concentric flight plan derives its name.

It is certain that there will have been execution errors following the CSI maneuver and the CDH maneuver, and it is desirable that the resultant dispersions are absorbed in a manner to accomplish near minimum ΔV penalty while achieving rendezvous at approximately, but not constrained to, planned time. Another property of the concentric flight plan is that near optimum phase height relationships may be achieved by one single correlating parameter, and that is the elevation of the line of sight from the LM to the CSM. For example, let it be said that it is an elevation angle 26-1/2°. If this elevation angle is adhered to as a trigger for the TPI maneuver, then all subsequent dispersions will essentially be absorbed. The potential penalty is that TPI triggered in this manner will not occur at the selected time for mission planning purposes. However, this will not really be a problem in that the mission planning selected time for TPI will have been chosen knowing that dispersions of a known magnitude might occur and due allowance will have been made for it in the selection of TPI time. Therefore, although the occurrence of an elevation angle 26-1/2° may occur earlier corresponding to a lower altitude of CDH, there should not be any resultant problems.

The following steps are taken to achieve TPI:

1. The occurrence of an elevation angle of 26-1/2° for the line of sight is determined.

2. The position of the target vehicle following 140° of central angle travel subsequent to the newly determined TPI time is established. An equivalent of this is just to find the position of the target vehicle at a set elapsed time from the TPI time, supposedly corresponding to 140° travel of the target vehicle.

3. A Lambert solution for an intercept trajectory from the TPI LM vector to the intercept vector is now determined.

4. The orbit of the Lambert solution is the basis of the targeting to which cross product steering may be applied during a finite burn in which the Lambert solution will be repetively found in order to derive the velocity required.

The midcourse correction occurring at a fixed elapsed time after TPI will also be derived from the Lambert solution. Main braking may also be determined by solving Lambert's problem providing that a lsightly modified intercept point and associated time consistent with an approximately average closing rate which is specified from mission planning considerations. Subsequent braking will be carried out using the same principles until manual takeover occurs.

Before closing the section on concentric flight plan targeting, it should be mentioned that both the CSI and CDH maneuvers are constrained to burn parallel to the CSM orbit plane.

REENTRY TARGETING

Reentry targeting is achieved by creating a reference in real time such that subsequent to the reference gains on the lift control may be adjusted as functions of differentials between the reference and actual conditions. Hence, the discussion is centered on two main subjects, "creation of the reentry reference," and "real time iterative control."

CREATION OF THE REENTRY REFERENCE (Figure 12)

The reentry discussion which follows is to some extent conceptual; however, the description of the major principles employed is sufficient to give an insight to the onboard computer logic which will be formulated to provide control during reentry from a lunar mission.

The CSM will enter lift vector up. This is because the last midcourse correction will be aimed at a corridor height consistent with the following two conditions:

1. If the onset of g tests indicate a possibility of a near skip condition, then the situation may be safely corrected by rolling the lift vector down until g onset is satisfactory.

2. If g predicted is more than the preferred limit, say 8 g, then if the left vector control retains the lift vector upwards, then the 8 g will be exceeded, but not to an extent which will really trouble the crewmen and certainly the g reached will be well below that level which would give concern regarding structural integrity.

With this philosophy, vertical lift is maintained until the trigger for the "creation of the reentry reference" is activitated. This trigger is a radial rate of 700 ft/sec (see Figure 12), whereupon "constant drag control" is initiated, i.e., by controlling the roll angle of the lift vector, a control is exercised which strives to maintain constant drag.

A reentry reference is now calculated. This is preferable to using a preflight reentry reference, in that basic assumptions in any preflight nominal would be impossible to maintain. The reference trajectory created in real time attempts to take into account actual conditions being experienced such as the actual height, velocity, density relationships. The steps taken to find the reference in real time are briefly, conceptually described below:

1. The constant drag mode, together with simplified and linearized equations of motion referred to the trajectory plane, allow a prediction of the state conditions when the flight-path angle is zero, i.e., at the bottom of the pull up. Hence R_1 is known.

2. With an effective L/D = .2 from the bottom of the pull up, the state conditions when the acceleration falls to 6 ft/sec^2 may be predicted. When the acceleration, and to some the preferred term may be deceleration, has fallen to 6 ft/sec^2, than a space condition trajectory is assumed. However, at this point let us note that the state condition at this juncture is noted and, hence, R_2 is determined.

3. From this deceleration point until the acceleration has risen to 6 ft/sec^2, "two body space assumptions" are applied and, hence, R_3 may be calculated and the state

conditions when the acceleration has reached 6 ft/sec^2 may be determined. This corresponds to the definition of a second entry.

4. The second entry range is predicted by linearized assumptions assuming a constant effective L/D and, hence, the reentry range corresponding to second entry, R_4, is known.

5. R_1, R_2, R_3, R_4, are summed and compared with the desired range which corresponds to the downrange distance of the splash point relative to the initial entry point.

6. The error ΔR is noted and if it corresponds to an overshoot, the consequent logic is that the "constant drag control" which is being maintained during the "creation of the reentry reference" is extended for another cycle and if, after going through the whole process again, there is still an overshoot predicted, this time smaller because of the energy absorbtion, another cycle will be called for. If, however, there is an undershoot ΔR error, then the "constant drag control" mode will be terminated. If the predicted range is within a certain limit "real time iterative control" would commence immediately. If the undershoot is exceeding acceptable limits, then the "down control" of the "real time iterative control" would command lift vector up. In any case, the phase of iterating onto the reentry reference terminates.

Before leaving this section, it should be noted that if the velocity corresponding to 6 ft/sec^2 deceleration at the predicted end of the "exit" phase is more than circular velocity, then this in itself will call for a recycle while "constant drag control" is maintained, in order to absorb a sufficient amount of energy to reduce the exit velocity to below that of circular conditions.

REAL TIME ITERATIVE CONTROL (Figure 13)

When the mode in which the real time reference is being formulated terminates, a "down control" mode begins. The logical steps taken subsequent to the initiation of "down control" are conceptually described below:

1. The "down control" uses expressions formulated from simplified, linearized equations of motion. The differentials between the reference conditions and the corresponding actual values computed in real time and compared at the same velocity value as is being determined in real time

provide the commands for the "down control" (see Figure 13). This control attempts to achieve the same velocity magnitude at the bottom of the pull up, i.e., with flight-path angle (γ) = zero.

2. When it is determined that $\dot{r}$ = zero, i.e., $\gamma = 0$, then the logic will go over to employing "up control." In this mode again simplified, linearized equations of motion in the trajectory plane are used. With the equations in this form, V_{ref} and $\dot{r}_{ref}$ are derived for given drag values. The differentials between V_{ref} and actual V, $\dot{r}_{ref}$ and actual $\dot{r}$, determined for the same drag value which essentially is being measured, provide variable gains to increment effective L/D (see "up control" block on Figure 13). The "up control" mode will continue until the deceleration has fallen to 6 ft/sec^2.

3. The onboard computer, having sensed the deceleration has fallen to 6 ft/sec^2 will occur which by definition is the initiation of the "second entry phase."

4. During the "second entry phase," the second entry range is predicted empirically based on drag and velocity differentials referenced to preflight simulated data. Velocity is the correlating parameter establishing the differentials.

5. The range error is fed back and acts as a variable gain for effective L/D increment during the "second entry phase" (see "second entry phase" block on Figure 13). With this iterative control, downrange error is nulled.

6. During the "second entry phase," computer logic will also call for a prediction of max g and when the max g predicted is beyond the acceptable limit, then the normal "second entry phase" control logic is interrupted and lift vector up is commanded; however, the max g prediction continues and when the predicted max g gets within the acceptable limit, then normal "second entry phase" control logic continues to call for L/D increments which will null the downrange error.

In all atmospheric phases of the reentry control, lateral control is also exercised to null out crossrange errors. Essentially, it works on the following principles:

1. When max uplift is called for, the lift vector will retain a 15° roll angle in a direction such as to null out crossrange errors.

2. For all other roll angles used for controlling effective L/D in the plane of the trajectory, the roll angle is in the direction which will null out any existing predicted crossrange errors.

This concludes the conceptual description of the way in which downrange and crossrange errors are nulled out during reentry, together with an indication of the logic to avoid potential skip and unacceptable max g conditions.

ENTRY MONITORING SYSTEM LOGIC (Figure 14)

This section includes a brief reminder of the monitoring that goes on parallel to the operation of the prime guidance system. First of all, the onset of g is noted in the very initial phases of the first entry, and if the onset does not reach critical value within 10 sec, then lift vector down will be commanded. This is classified as "corridor verification" (see Figure 14).

"Corridor verification" is followed by "g, V" monitoring where the actual "g, V" that is being experienced is detected by backup integrating accelerometer and is shown as a trace on a display where rays are scribed and g boundaries corresponding to entry velocity converge on the asymptotic value of the acceptable g limit (see Figure 14) and the associated value of 9 g. If the slope of the trace is more negative than the appropriate g boundary corresponding to the actual entry velocity, then this is an indication that the g to be encountered will be unacceptable. If the prime guidance in this instance is not commanding lift vector up, then a serious excessive g situation is developing. If, during the "exit" phase of the first entry, the trace is parallel to the "skip" rays and the prime guidance is not commanding lift vector down, then a serious potential skip situation is developing. To insure safe reentry the crewman will probably resort to manual control of the lift vector; however, no longer will an accurate splashdown point be possible comparable with when the prime guidance is functioning properly.

Further development work is continuing as regards range control lines added to the "g, V" display. This latter development would reduce gross range errors during manual control

in circumstances when the prime guidance was not functioning correctly or capable of providing accurate displays. Reentry ship tracking of the exit phase, especially where effectively direct altitude rate is being measured, will also rapidly detect gross overshoots or undershoots.

SOFTWARE CAPABILITY NOT ONBOARD (Figure 15)

The title implies a list of capabilities not onboard, but it should be explained at this juncture that the ground capability in every case, will provide the capability which is stated does not exist for the onbaord capability.

1. On the ground, the capability exists to run ahead of real time and simulate an intended maneuver such that all conditions subsequent to the burn may be evaluated such as tracking acquisition, look angles, etc., and an accurate prediction of coast subsequent to the burn may be evaluated. The onboard capability is restricted to free coast prediction.

2. The ground control has the capability of predicting the characteristics of the total flight plan assuming present targeting values are retained. The fuel reserves may be evaluated after all the maneuvers have been made, together with all the appropriate operational information required to insure a mission success.

3. Ground control has the capability to determine a dispersed condition for the spacecraft, and from this dispersed state, can re-optimize the entire flight plan. This incurs sending new targeting data to the spacecraft which the onboard capability has been formulated to accept.

4. If the LM has to take off at a time when the CSM is out of communication with it, e.g., when the CSM is behind the moon, then the consequent rendezvous is excessively lengthy for an entirely LM active rendezvous with the CSM remaining passive. The ground control can design maneuvers for the CSM subsequent to which the LM active rendezvous is accomplished in a considerably shorter time than would be the case with the CSM remaining passive. There are cases with the CSM, say 160° phase behind the LM at insertion, where if an out of plane situation exists, and the CSM remains entirely passive, then the LM capability to rendezvous within its lifetime and delta V limits are in jeopardy. With the ground control in the picture, targeting advice would be issued to the CSM such that the subsequent LM active rendezvous could be accomplished in less time and for less delta V.

5. The onboard capability, in many instances, has not got the capability, and this is by intent as will be indicated later, to calculate peripheral data. Examples of peripheral data are: look angles, prediction of optical conditions, prediction of lighting conditions for rendezvous, docking, prediction of contrast and shadow conditions for descent, trajectory information for operational facilities such as monitoring aircraft and recovery ships, and differential radiation prediction.

6. The ground has the capability of using observations from mixed sources to provide optimum trajectory determination solutions. An example of the mixed sources would be ground radar and onboard radar. The observations receive appropriate weighting consistent with a knowledge of the accuracy existing for the circumstances incurred.

In the next section, it is intended to discuss the ground control G&N functions, not in the sense that it is filling gaps in the onboard capability, but in the sense of what the ground control provides as a service when the onboard G&N systems are functioning normally.

GROUND SUPPLEMENTARY AND BACKUP G&N FUNCTIONS (Figure 16)

1. In many circumstances during the lunar landing mission, the MSFN determination of position velocity will be more accurate than that which would be determined onboard. Furthermore, for mission planning purposes, the number of observations is being restricted in order to economize on RCS fuel, i.e., a change of attitude would, in all probability, be required prior to a sequence of onboard sightings. Consequently, it is the ground control which provides updated or modified targeting data.

2. The ground control provides monitoring information for manual maneuvers. Examples of this are launch abort, manual insertion into orbit using Stabilization Control System (SCS) mode, when the terminal phase initiation is performed manually, then the fround would provide monitoring information prior and subsequent to it. There is the distinct possibility that prior to the first braking of the terminal phase maneuvers when the AGS is being used to accomplish the rendezvous that the MSFN will supply the targeting information.

3. The ground control, with its extensive computer capability available to it, together with the presence of specialists in the particular area in which a problem may occur, has a capability for a more extensive exploration of the possibilities, especially in contingency situations.

4. The MSFN capability will, on occasions, perform an umpiring function. For example, the LM has two onboard guidance systems which will provide data. If a discrepancy occurs, then the MSFN will provide a significant clue as to which system is functioning correctly. When long tracking passes are possible, then the MSFN will provide a determination of the orbital elements or of the relative state between the two modules, when coasting conditions prevail. During short tracking passes, the MSFN can be relied upon to give extremely accurate data along the direction of the line of sight from the ground radar dish to the spacecraft. This is because the MSFN is relying on Doppler rate information. Even during LM launch, this particular measurement by the MSFN will provide an immediate clue as to which onboard guidance system is functioning correctly, should there be a serious deterioration in the performance of either one of the onboard systems.

In summary then, the MSFN will provide some assistance as regards "ascent switchover monitoring," "navigation", and determination of targeting.

AUSTERITY IN ONBOARD COMPUTER

During the current stage of the Apollo project, it has become increasingly clear that restraint must be used in the selection of programs to be formulated for the onboard computers. There are two important reasons for this and they are: (1) to avoid overflowing the capacity of the computer, and (2) maintenance of schedule.

1. Approximately 1-1$\frac{1}{2}$ years before launch, it is important that the summation of the estimates of the word capacity for each program leaves something of the order of a 15% margin relative to the ultimate capacity of the computers. There are two good reasons for this. The first one is that when a program is only partially formulated, then there is a degree of uncertainty as to the final capacity allocation needed for that program and also a nominal 15% margin at this stage is desirable because secondly, when the overall margin of the computer capacity is in jeopardy, inherently each program has to be formulated with additional care in order to restrict capacity demands. This means that there is a tendency for all programs to be formulated somewhat more slowly than they normally would than when it is known that the margin is not in jeopardy. This facet, in itself, causes a delay in schedule.

2. It is important that the first draft of each program is formulated at the earliest time possible. This is because it is important that each program is fitted into an

integrated formulation such that controlling and sequence selector programs can be tested in conjunction with the individual programs. Furthermore, a period is necessary for checking, testing, and verifying each program and the integrated formulation of all programs.

3. Recent surveys of the selection of programs for the onboard computers have revealed that further austerity in onboard computer capability is needed. It is now apparent that some main programs previously selected with the best intentions and there were particular reasons why each one in itself was desirable, however, not mandatory for onboard computer capability. These main omissions are:

(1) Self checking restricted: It was intended that there should be automatic testing for malfunction of the computer with diagnostic warnings displayed.

(2) Logic to avoid gimbal lock not included: This is a feature of the altitude change programs to insure that the rotations were such that gimbal lock was not incurred. Crew monitoring of the 8 Ball now becomes a more critical function to insure the same objective.

(3) CMS: No concentric sequence logic: In all non time critical situations, a concentric rendezvous flight plan is the preferred mode of rendezvous, however, the capability for this has had to be excluded from the CMC state to be entered into the LGC, which is then called upon to work out the concentric sequence for the CSM, and this is advised by voice to the CSM crewman. He in turn enters the required targeting through the CSM DSKY.

(4) Absence of flight plan prediction: Trajectory determination has always been, and is restricted to free coast prediction.

(5) Point return: This capability has been deleted from the "return to earth" program, however, the time critical mode may be manually iterated through the DSKY by entering delta V capabilities less than the maximum. This effectively produces longitudinal control of the splash point, in that convergence is easy enough if the variation of the longitude of the splash point is noted in its relationship with the "delta V capability" input to the DSKY. The iteration logic is described in Targeting - Type 3. When point return capability is deleted, and with this proviso it is not necessary that the nominal recovery point is contained within the return trajectory plane, then the iteration logic may be considerably simplified.

(6) MSFC iterative steering: In order to insure a lack of serious transients in the event of the spacecraft having to take over guidance of the Saturn, it was considered necessary that the CMC computer contained the MSFC iterative steering logic. This requirement has been deleted because: (1) it was not considered that crew safety was seriously impaired, in that safe aborts are still possible. This entails a CSM separation from the Saturn and therefore, in this case, the objectives would no longer be possible, however the probability is low. (2) The probability of detection of trouble with the Saturn platform during the translunar burn associated with a nondetection during parking orbit, is considered to be extremely low. If a Saturn platform failure is detected during the parking orbit and it must be remembered that it is only the Saturn platform which is being backed up, i.e., the Saturn computer is an essential component when the spacecraft has taken over, the spacecraft can use a simpler scheme without fear of transients. This scheme is described in Targeting - Type 1, and the simpler MIT cross product steering is applied. It must be remembered that the spacecraft only takes over in contingency circumstances.

TIME HISTORY OF AN ONBOARD PROGRAM (Figure 18)

Figure 18 indicates a typical time history of an onboard program. The initial functional "requirements" must be issued something like $18\frac{1}{2}$ months before launch. Manned Spacecraft Center issues the requirements to the contractor, in the case of the prime guidance systems, MIT. Subsequent to the issuance of the requirements, program modular formulation begins which is followed by unit checking and testing in each case. With the complexity of modern onboard integrated programs, this process of modular formulation and unit checking and testing will take on the order of $10\frac{1}{2}$ months. At the end of this time the "first draft of integrated formulation" is achieved. Now bit by bit testing of the whole integrated program begins, together with "hybrid simulations" involving the checking of warning lights, etc. These processes will take on the order of 3 months by the end of which time the "flight program release" may be issued to the contractor responsible for the manufacture of ropes. From the "flight program release" to the "delivery of flight ropes at Cape", will incur an additional $1\frac{1}{2}$ months. Subsequent to the delivery of the flight ropes at the Cape, a period of intense computer interface testing with other systems dependent upon the onboard computer, ensues. With one or more tests occurring daily, a busy schedule lasting an additional $3\frac{1}{2}$ months, is unavoidable. When the interface testing has been achieved then the onboard

computers are ready for launch activity.

One unescapable milestone should be recognized and that is that the entire flight program must have been verified at least 5 months before launch. This is the permanent logic in the computer. Of course, the quantitative values of guidance constants, etc., can be inserted into the erasable memory at a much later time before launch, facilitating alternative missions, etc. However, again it is significant that the permanent logic must be ready and verified 5 months prior to launch.

SUMMARY (Figure 19)

1. The onboard capability has been framed such that:

   (1) The selected programs provide a capability that insures crew safety.

   (2) The selected programs provide a capability which significantly enhances mission success, for example the spacecraft guidance capability for translunar injection, which would be used in contingency circumstances.

   (3) The onboard programs have been framed such that they can efficiently use ground information.

2. The ground capability is essential in some cases, particularly is this the case when real time flight planning is required involving both the CSM and LM. It will be remembered that this is incurred with rendezvous cases with abnormal initial phasing. In general, the ground capability contributes to mission success as has been noted in the manned space projects so far and the ground capability provides necessary useful peripheral data.

3. It is found that the combined capability further advances crew safety. This is evident in launch and abnormal rendezvous situations. The combined capability insures more optimum trajectories due in some cases to the increased accuracy and in others, to the increased flexibility of capability, and in general, it is found that the combined onboard and ground capability provides maximum probability of mission success.

Questions and Answers

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

Speaker: Morris V. Jenkins

1. Dr. Mueller - Is there a budget for the CSM and LM computers?

   ANSWER - Yes. There is now about a 3,000 word pad in the CSM and a 5,000 word pad in the LM.

2. Mr. North - Would you ever use remaining SPS propellant to reduce entry velocity?

   ANSWER - No.

NASA-S-66-6847 JUN

SOFTWARE CONSIDERATIONS

- ONBOARD ADEQUACY INSURES CREW SAFETY
  - INDEPENDENT CAPABILITY TO RETURN
- MISSION SUCCESS CONTRIBUTIONS
  - TLI CAPABILITY
  - ACCEPTS GROUND UPDATES
- GROUND CAPABILITY ESSENTIAL IN SOME CASES
- GROUND CONTRIBUTES TO MISSION SUCCESS AND PROVIDES NECESSARY PERIPHERAL DATA
- COMBINED SOLUTIONS:
  - ADVANCES CREW SAFETY
  - OPTIMIZED SOLUTIONS
  - MAXIMUM PROBABILITY OF MISSION SUCCESS
- ADOPTION OF SIMPLE MIT TARGETING SCHEMES HAS CONTRIBUTED

Fig. 1

NASA-S-66-6553 JUN

ADEQUACY OF ONBOARD SOFTWARE CAPABILITY

- CREW SAFETY
  - INDEPENDENT CAPABILITY TO RETURN
  - CAPABILITY TO ACCEPT GROUND UPDATES
  - COMPATIBILITY OF SOFTWARE IN PGNCS AND AGS
  - DOUBLE CHECK ON RELATIVE STATE
  - LM CAN DIRECT CMS ACTIVE RENDEZVOUS
  - SOFTWARE LOGIC FACILITATES MEANINGFUL CREW CHECKS
- MISSION SUCCESS
  - NECCESSARY STEERING
  - TRANSLUNAR INJECTION
  - ACCEPTANCE OF ALTERNATIVE TARGETING FROM THE GROUND

Fig. 2

NASA-S-66-6542 JUN

BASIC ONBOARD G AND N SOFTWARE CAPABILITY

CREW G AND N MANAGEMENT → DISPLAYS				
WHERE ARE WE ?	NAVIGA-TION	OBSERVA-TIONS →	TRAJECTORY DETERMINATION PROCESSORS →	CURRENT SMOOTHED POSITION AND VEL DETERMINATION
WHERE WILLWE BE ?	DEAD RECKONING	EPHEMERIS →	TRAJECTORY PREDICTION PROCESSORS →	BEST ESTIMATE OF FUTURE POSITION AND VELOCITY
WHAT CHANGE OF COURSE IS NEEDED ?	CHANGE OF COURSE DETERMINA-TION	CHANGE OF COURSE DETERMINATION PROGRAMS →	PRE-THRUST PROGRAMS →	TARGET OBJECTIVES FOR THRUST
IS THE PRIME G AND N SYSTEM CONTROLLING THE PROPULSION CORRECTLY ?	MANEUVER CONTROL	CREW INITIATION OR PROCEED CONTROL →	THRUST PROGRAMS STEERING EQUATIONS →	ACHIEVEMENT OF DESIRED CUTOFF CONDITIONS

Fig. 3

NASA-S-66-6555 JUN

AMPLIFICATION OF THE BASIC PROGRAMS

TRAJECTORY DETERMINATION PROCESSORS	BASIC COMPONENT: KALMAN FILTER
TRAJECTORY PREDICTION PROCESSORS	● ENCKE METHOD ● INCREMENTS INTEGRATED ● LARGER STEPS FOR THE SAME ACCURACY
PRE-THRUST PROGRAMS	● SET UP THE TARGETING ● THRUST COMMAND ATTITUDE ● PREFERRED PLATFORM ALIGNMENT
THRUST PROGRAMS	● X PRODUCT STEERING ● E GUIDANCE ● LOCAL VERTICAL COORDINATES

Fig. 4

NASA-S-66-6551 JUN

TARGETING - TYPE 1

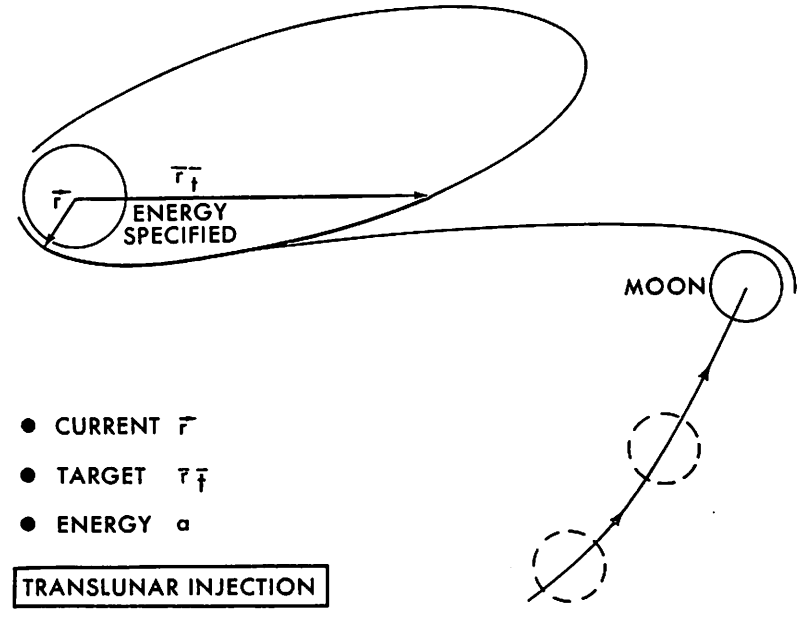


Fig. 5

NASA-S-66-6547 JUN

TARGETING TYPE 1

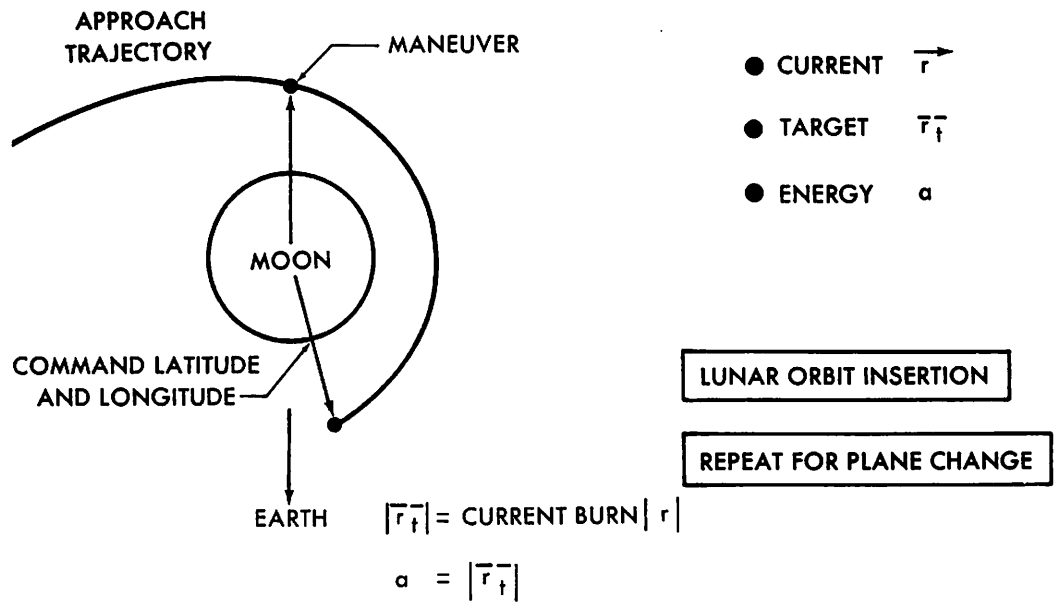


Fig. 6

NASA-S-66-6540 JUN

TARGETING TYPE - 2

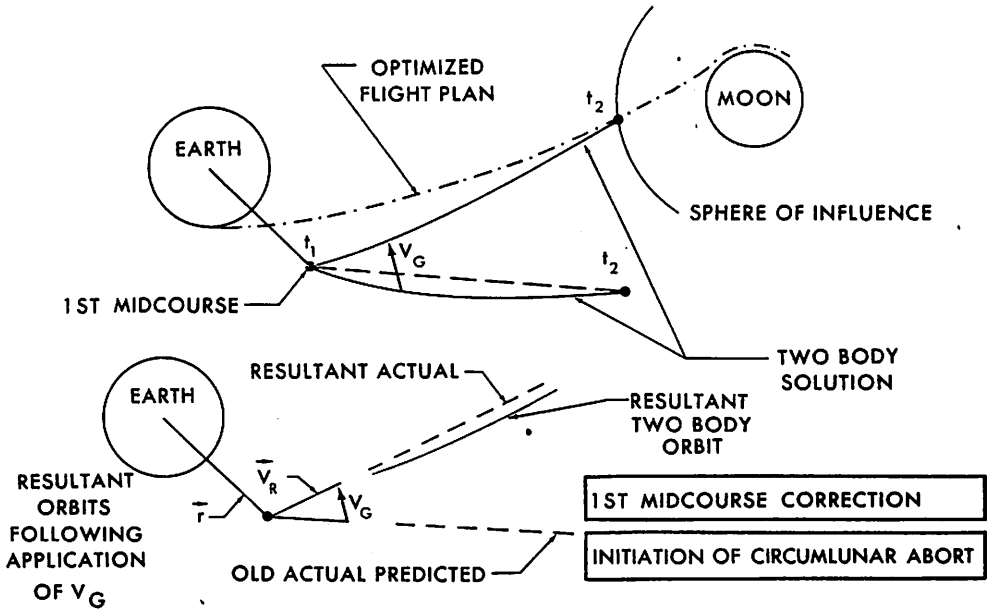


Fig. 7

NASA-S-66-6514 JUN

TARGETING TYPE 2

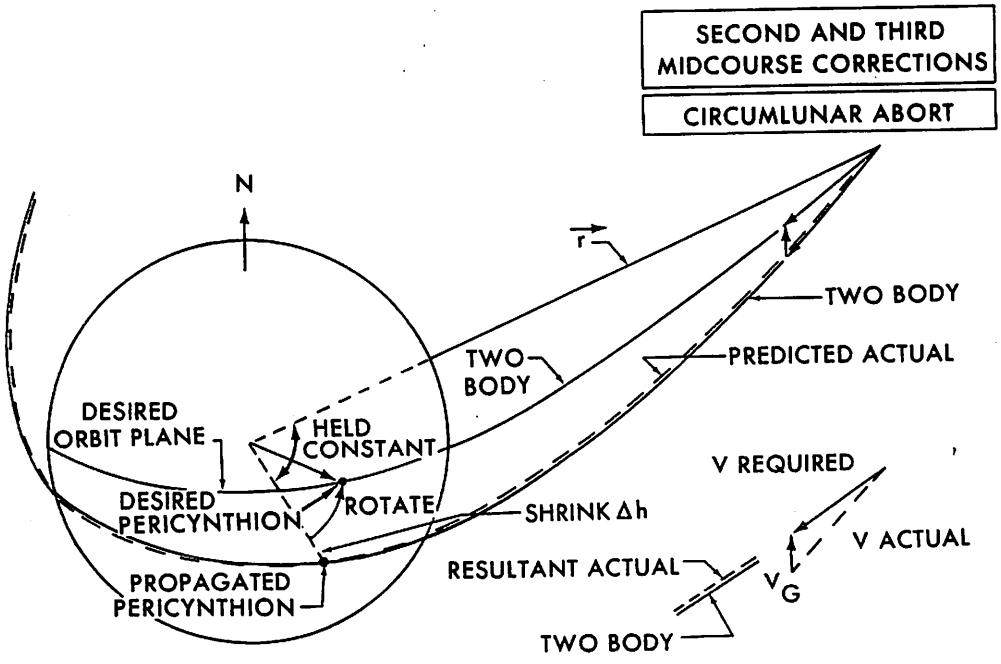


Fig. 8

NASA-S-66-6565 JUN

TARGETING TYPE 3

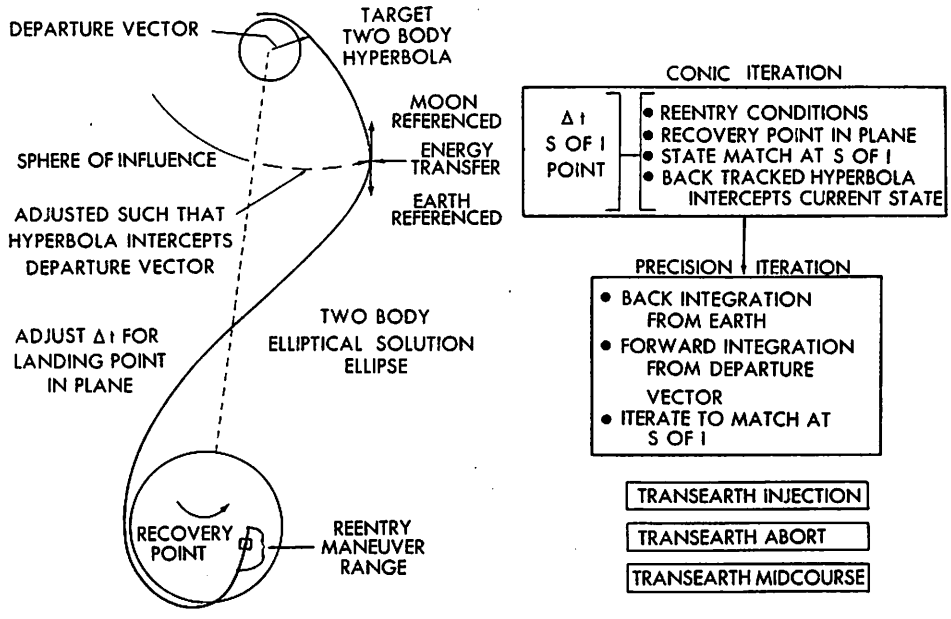


Fig. 9

NASA-S-66-6531 JUN

DESCENT AND ASCENT TARGETING

DESCENT	ASCENT
● PROFILE SHAPING TARGETING	● ORBITAL TARGETING
• INTERMEDIATE POSITION TARGETING	• CUTOFF POSITION AND VELOCITY VECTOR IN CSM PLANE
• WITH AN ASSOCIATED VELOCITY VECTOR	• γ SPECIFIED
• TOGETHER WITH AN ASSOCIATED ACCELERATION VECTOR	• $\lvert V \rvert$
	• CUTOFF $\lvert h \rvert$ SPECIFIED
	• DOWNRANGE FREEDOM IN CUTOFF POSITION WITH SOME RESULTANT FREEDOM IN DIRECTION OF LINE OF APSIDES

PRINCIPLES OF STEERING COMMON TO BOTH: COMMAND THRUST VECTOR DERIVED FROM EXPLICIT EXPRESSIONS INVOLVING CURRENT ACCELERATION VECTOR, AND DIFFERENTIALS OF CURRENT AND TARGET CONDITIONS, i.e., ΔT_{GO}

ΔV

Δr

Fig. 10

NASA-S-66-6856-JUN

CONCENTRIC FLIGHT PLAN

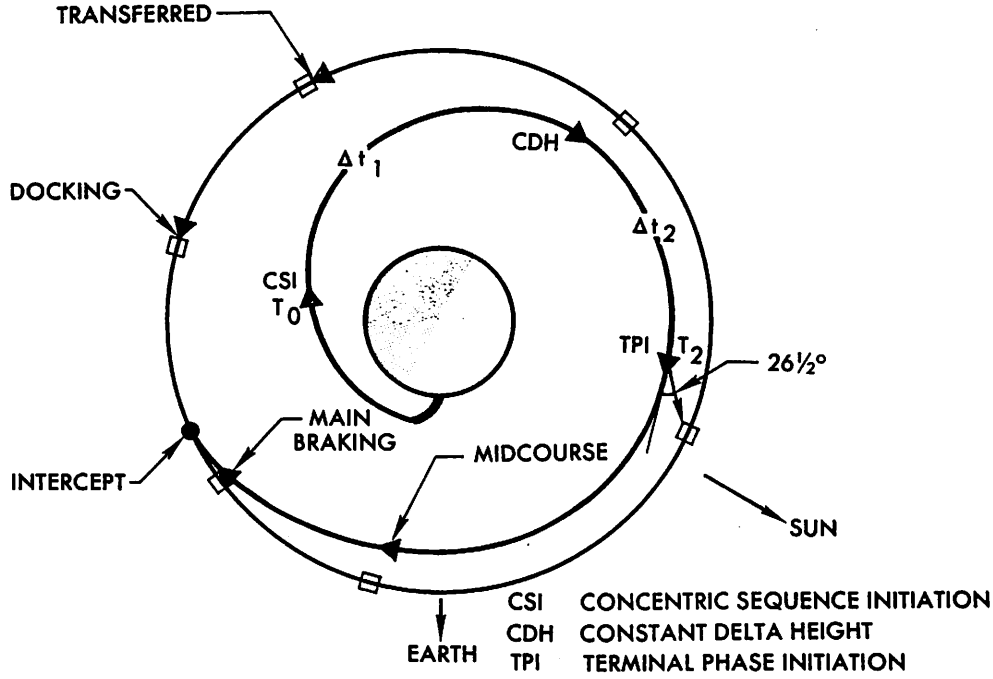


Fig. 11

NASA-S-66-6589 JUN

CREATION OF THE REENTRY REFERENCE

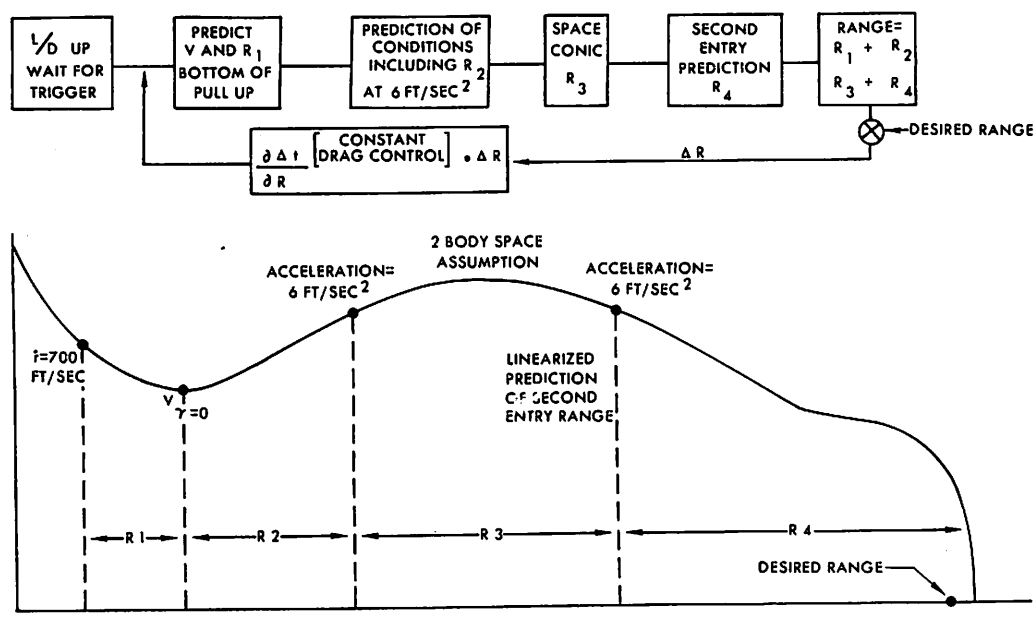


Fig. 12

NASA-S-66-6588 JUN

REAL TIME ITERATIVE CONTROL

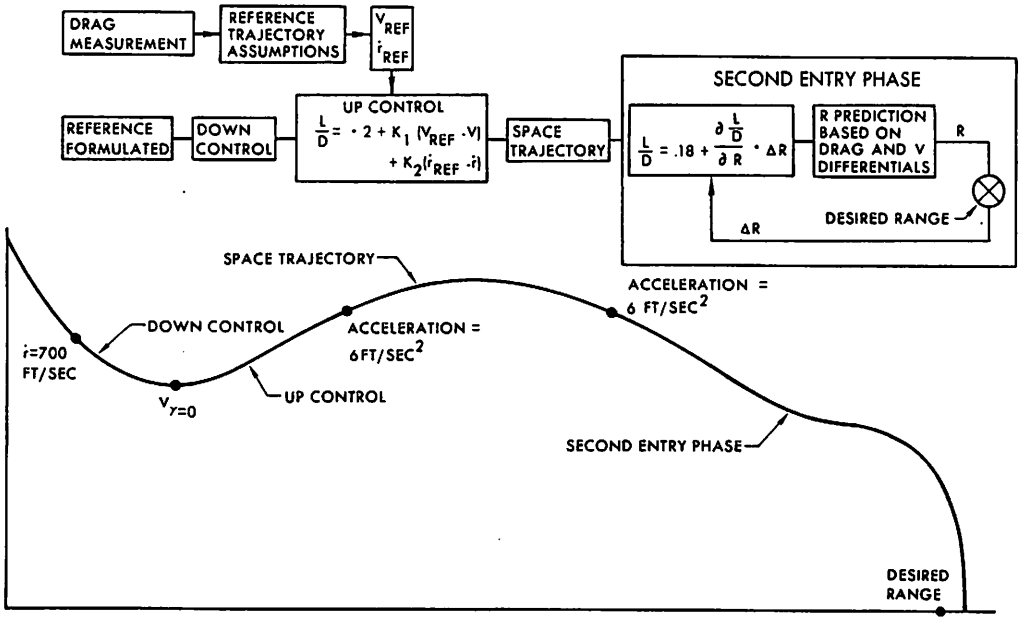


Fig. 13

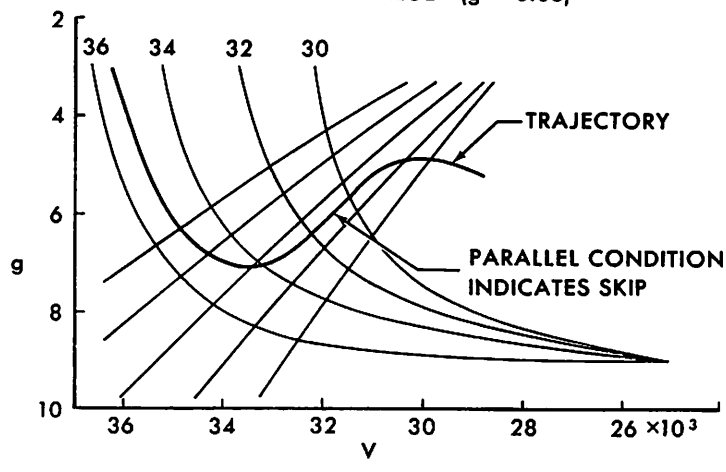
NASA-S-66-6550 JUN

ENTRY MONITORING SYSTEM LOGIC

- CORRIDOR VERIFICATION

0.05 g–10 SECOND DELAY–g
CRITICAL VALUE >(g —0.05)

- (g, V) MONITORING



- FURTHER DEVELOPMENT - RANGE CONTROL LINES

Fig. 14

NASA-S-66-6530 JUN

SOFTWARE CAPABILITY NOT ONBOARD

- PRE-MANEUVER SIMULATION
- REAL TIME TOTAL FLIGHT PLAN EVALUATION
  - PREDICTION OF SEQUENCE EVENTS AND FUEL RESERVES
- CAPABILITY TO RE-OPTIMIZE THE WHOLE FLIGHT PLAN
- COMBINED MANEUVERS FOR ANY TIME LIFT-OFF
- PERIPHERAL INFORMATION
  - LOOK ANGLES/SIGHTING DATA
  - LIGHTING CONDITIONS FOR RENDEZVOUS CONTRAST
    SHADOW CONDITIONS FOR DESCENT
  - TRAJECTORY INFORMATION FOR OPS FACILITIES
  - DIFFERENTIAL RADIATION PREDICTION
- COMBINED O D SOLUTIONS FROM HYBRID OBSERVATIONS
  WITH APPROPRIATE WEIGHTING

Fig. 15

NASA-S-66-6512 JUN

GROUND SUPPLEMENTARY AND BACKUP G AND N FUNCTIONS

- TARGET DETERMINATION AND SUPPLY
- PILOT MONITORING INFORMATION FOR MANUAL MANEUVERS
- MORE EXTENSIVE EXPLORATION OF THE POSSIBILITIES
  ESPECIALLY IN CONTINGENCY SITUATIONS
- UMPIRING FUNCTION: PGNCS AND AGS
  MSFN INFORMATION TO ASSIST IN LM
  - ASCENT SWITCHOVER MONITORING
  - NAVIGATION
  - DEAD RECKONING

Fig. 16

NASA-S-66-6544 JUN

AUSTERITY IN ONBOARD COMPUTER

- CAPACITY
- SCHEDULE
  - PROGRAM FORMULATION
  - CHECKING
  - TESTING
  - VERIFICATION
- MAIN OMISSIONS
  - SELF CHECKING RESTRICTED
  - LOGIC TO AVOID GIMBAL LOCK NOT INCLUDED
  - CMC: NO CONCENTRIC SEQUENCE LOGIC
  - ABSENCE OF FLIGHT PLAN PREDICTION
  - POINT RETURN
  - MSFC ITERATIVE STEERING

Fig. 17

NASA-S-66-6552 JUN

TIME HISTORY OF AN ONBOARD PROGRAM

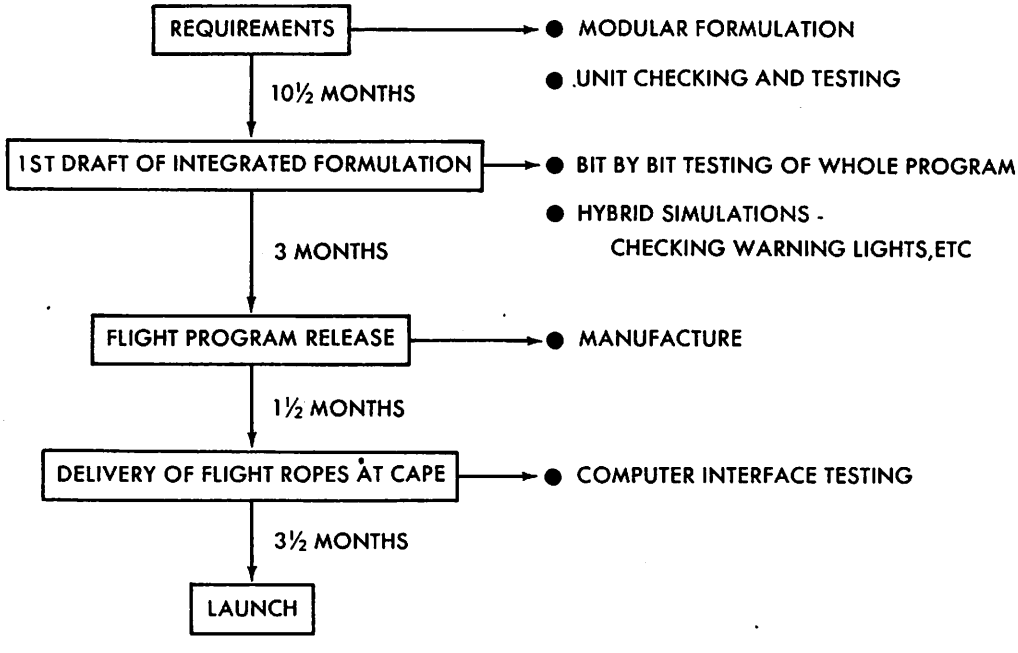


Fig. 18

NASA-S-66-6518 JUN

SUMMARY

- ONBOARD
  - CAPABILITY INSURES CREW SAFETY
  - CAPABILITY IS EFFICIENTLY USED TO ENHANCE MISSION SUCCESS
  - CAN EFFICIENTLY TAKE GROUND INFORMATION
- GROUND
  - CAPABILITY IS ESSENTIAL IN SOME CASES
  - CAPABILITY CONTRIBUTES TO MISSION SUCCESS AND PROVIDES NECESSARY USEFUL PERIPHERAL DATA
- COMBINED ONBOARD AND GROUND
  - CREW SAFETY FURTHER ADVANCED
  - MORE OPTIMUM TRAJECTORIES
  - PROVIDES MAXIMUM PROBABILITY OF MISSION SUCCESS

Fig. 19

COMMUNICATIONS FUNCTIONS

By

Douglas R. Broome, Jr.

COMMUNICATIONS FUNCTIONS

A summary of the spacecraft and launch vehicle functional capabilities is shown in Figures 1 and 2. As can be seen in Figure 1, CSM and LM both provide the S-Band voice telemetry, at both the high and low bit rates, ranging, tracking, and television. One difference in these capabilities is that the LM cannot transmit television except from the lunar surface after the erection of the Lunar Surface Erectable Antenna. In addition to these capabilities, the CSM provides the capabilities of updata, recorded data and voice playback, and scientific data transmission. At VHF, both the CSM and LM have voice communications capabilities and the LM has the capability of transmitting low bit rate telemetry from the LM to the CSM during descent and ascent. The CM has both VHF and HF two-way voice and beacon transmission capability for recovery phases.

The launch vehicle Instrument Unit (IU) provides the capabilities of telemetry, tracking, ranging, and updata at S-Band, as well as telemetry at VHF and tracking at C-Band. The three launch vehicle propulsion stages have VHF telemetry capabilities and range safety destruct.

Figures 3 and 4 show the Manned Space Flight Net (MSFN) support capabilities to support the spacecraft and launch vehicle communications functions. Figure 3 illustrates the ground based station capabilities and Figure 4 illustrates the ship and aircraft capabilities. It should be noted that the ground based stations have identical capabilities except that the Madrid, Canberra, and Goldstone stations have the ability to recieve television from lunar distance. It should also be noted that not all stations can track the IU C-Band transponder. The reentry ships do not have updata capabilities and they utilize 12 foot parabolas rather than 30 foot parabolas. The recovery forces have, in addition to VHF and HF voice and beacon capabilities, S-Band direction finding capability.

Figure 5 represents a ground track for the first two orbits of the nominal lunar mission. The circles shown represent the station coverages for a one hundred nautical mile orbit. Not shown on this map is the fact that for altitudes above 8,000 nautical miles, the Madrid, Goldstone, and Canberra coverages overlap, thus providing continuous, unbroken communications with the spacecraft.

The Radio Frequency Systems have been broken down into several figures in order to make it easier to understand their functions in meeting their respective Module requirements. Figures 6 and 7

represent the data sources for the two Modules. As shown in Figure 6, the Central Timing Equipment, PCM Telemeter, Signal Conditioner, and the Pre-Modulation Processor all contain redundant circuit elements for critical circuits such as power supplies, voice modulators, telemetry modulators, and digital counting blocks. The PCM telemetry, in particular, contains both circuit and block redundant elements. Also, the CSM has three sets of microphones and amplifiers, one set for each astronaut; each of these sets is redundant.

Figure 7 represents the data sources for the LM. Again, circuit redundancy is provided. The LM voice recorder has no playback capabilities. It will be brought back to earth, however.

Figures 8 and 9 are block diagrams of the two respective Modules' S-Band systems. The LM S-Band system is comprised of a coherent transponder, a power amplifier, a diplexer, and an antenna system consisting of two Omni elements, an In-Flight Steerable High-Gain antenna and a Lunar Surface Erectable Antenna. The transponder and power amplifier are completely redundant and there is by-pass switching around the power amplifier to allow for prime power conservation when high bit rate telemetry is not required. The Steerable Antenna is a 2 foot diameter parabola providing a transmit gain of 20.3 db and a receive gain of 16.5 db with a beam width of 13 degrees. The Erectable Antenna, in conjunction with the 750 milliwatt transmitter, allows transmission of voice and low bit rate telemetry to the MSFN without the use of the power amplifier, thus resulting in considerable prime power conservation. This antenna, in conjunction with the power amplifier, also allows transmission of high-quality real-time television pictures to the MSFN from the lunar surface.

Figure 9 presents the CSM S-Band system. This system is essentially the same as that of the LM with the addition of a separate S-Band FM transmitter. In the CSM, as in the LM, the coherent transponder is redundant. The power amplifiers are not shown as being redundant in that, at times, they are used in parallel; that is, power amplifier number 1 being driven by the FM transmitter and power amplifier number 2 by the PM transmitter. However, switching is such that the number 1 power amplifier can be switched to back up the number 2 power amplifier.

The CSM antenna system consists of four Omni antenna elements and a High-Gain antenna. This High-Gain antenna is made up of four 31-inch diameter parabolas and a crossed dipole array. The transmission system has three operating modes--wide, medium, and narrow, with attendant beam widths of 45 degrees, 12.3 degrees, and 4.4 degrees at the half power points. The receiving modes are coarse and fine--the beam widths being 45.5 degrees and 5.5

degrees at the half power points. If the transmit mode switch is in the "Wide" position, the receiving antenna mode will remain in the "Coarse" position upon acquisition of an RF signal. However, if the transmit mode switch is in the "Medium" or "Narrow" position, the receiving mode will automatically switch from "Coarse" to "Fine" upon signal acquisition. Use of the High-Gain antenna begins at an altitude of approximately 2,500 nautical miles after the transposition but prior to docking.

The three beam widths are derived as follows: The wide beam width is provided by using a crossed dipole array; the medium beam width by using one of the 31-inch parabolas; and the narrow beam width by driving the four 31-inch parabolas in parallel.

Figures 10 and 11 represent the VHF systems for the CSM and LM. These two systems are essentially identical in that both utilize a transmitter and receiver at 296.8 MC, and a transmitter and receiver at 259.7 MC. Usage of these equipments will be explained later. The CSM VHF Omni system consists of two scimitar elements located on the Ascent Stage. Note that the LM has an additional VHF antenna which is used for Extra Vehicular Crewmen (EVC) communications. This antenna is a discone mounted on an erectible boom located on the Ascent Stage. EVC signals are picked up by the 259.7 MC receiver, de-modulated, and then used to modulate the S-Band link to earth.

Figure 12 presents the equipments used on board for recovery phase communications. These equipments consist of an HF transceiver/beacon, a VHF beacon, and a VHF survival transceiver/beacon. This latter equipment is provided as part of the astronaut's survival equipment package. In addition, the 296.8 MC transmitter and receiver is used for two-way voice communications during the recovery phase. These equipments are used with three antenna systems which are erected during the recovery phase. The HF recovery antenna is a 16-foot spring steel leafed antenna which is erected following attainment of spacecraft Flotation 1 position. The two VHF recovery antennas are approximately 14 inches long, are made of spring steel such as is used in measuring tapes, and are erected immediately following main parachute deployment in order to allow maximum range communications with recovery forces. One of the recovery antennas has a manual connection provision which will allow use of the astronaut's survival transceiver/beacon with the spacecraft antenna.

Figures 13 and 14 present the pertinent parameters of the two S-Band systems. The CSM utilizes a coherent transponder operating at the frequencies shown on Figure 13 and a separate S-Band transmitter operating at 2272.5 MC. These radio equipments

operate with either S-Band Omni antenna system or the High-Gain antenna. Three RF power transmission modes are available: 250 milliwatts, 5 watts, and 20 watts. Power level is selected as a function of range and data rate.

The LM S-Band system utilizes a coherent transponder, operating at the frequencies shown on Figure 14; no separate transmitter is provided. This system is used in conjunction with one of the three LM antenna systems: the Omni, In-Flight High-Gain, or the Lunar Surface Erectible Antenna. Two RF power levels are provided: 750 milliwatts and 20 watts. Power level selection for the LM is based on data rate and bandwidth requirements. The erectible antenna is used for prime power conservation and also to allow the transmission of high quality real-time television pictures from the lunar surface.

Figures 15 and 16 present the characteristics of the two VHF systems. Power output for all transmitters is 5 watts each. Each in-flight antenna system consists of two radiating elements which allow essentially omnidirectional coverage. The CM also has two VHF recovery antennas and the LM has one VHF antenna for EVC operations.

Figure 17 presents the characteristics of the HF recovery equipment. The transmit and receive frequencies are 10.006 MC for both voice and beacon. Transmission power and peak-envelope power is 20 watts in the single-sideband mode and 5 watts in the double-sideband mode. The antenna used is the 16-foot erectible whip.

The Launch Vehicle RF requirements are presented in Figures 18 and 19. The VHF telemetry requirements are met by fifteen transmitters located by stage as shown. Power output for each transmitter is 18 watts through an Omni antenna system.

The C-Band tracking requirements for the launch vehicle are met using a pulse-type radar transponder and an Azusa transponder. The launch vehicle S-Band system is much like that of the CSM in that it consists of a coherent transponder and a separate FM transmitter operating at the frequencies shown here. Note that the frequencies of the coherent transponder are the same as those of the LM. However, no frequency interference problems are expected due to the physical separation of the launch vehicle and LM when the LM system is activated. Both launch vehicle S-Band transmitters operate at 20 watts through Omni antenna systems.

Figure 20 describes the Lunar Surface Experiment Package (LSEP). The uplink frequency of this package is 2119 MC, and three downlinks are shown. However, a single LSEP will only utilize

one downlink at a time. Since the lifetime of these packages is one year, it is expected that during a year multiple LSEP's will be operating simultaneously. Power output for each package is one watt through a helical antenna. Design life of the packages is one year with 100 per cent utilization. All experiments are multiplexed on one PCM downlink channel at 1060 bits per second to a 30-foot MSFN antenna. The uplink command system provides for system on-off, experiment activation, experiment mode changing, and additional command capabilities not yet allocated.

The communications capabilities provided by the combined space vehicle can be summarized as follows:

1. Earth-Spacecraft two-way voice.

2. Earth-Spacecraft-EVC voice relay and conference.

3. Telemetry from launch vehicle, spacecraft modules, and LSEP to earth.

4. Updata to launch vehicle, CSM, and LSEP from earth.

5. Tracking information enabling utilization of ground-based navigation capabilities.

6. Television from the CSM and the lunar surface to earth.

7. Recorded data and voice playback from CSM to earth.

8. Recovery location aids.

Figures 21 and 22 present the primary and secondary means of providing voice communication between the Spacecraft Modules, EVC's and the MSFN. The primary communication link between the Modules and the MSFN in-flight are the two S-Band coherent systems. In addition, in-flight, the CSM has the ability to dump via the S-Band FM transmitter any recorded data that it has acquired. For recovery, CSM has additional voice capabilities at 243.0 MC and 10.006 MC. The backup voice capabilities between the modules and earth are provided by the redundant S-Band systems in each Module. In addition, the CSM 296.8 MC equipment can be used as a second voice backup during earth orbital phases of the lunar mission.

The CSM-LM primary voice communications are provided with the 296.8 MC equipments in the Modules. The backup to this system is provided by using the remaining VHF equipments as required depending on the nature of the failure experienced. Voice

communications between the modules and an EVC are provided by transmitting to the EVC at 296.8 MC and receiving at 259.7 MC. The backup to this system results in a reversal of these two frequencies. This frequency reversal was found necessary in order to reduce the physical size and complexity of the suit equipment.

Figure 23 illustrates the voice relay and conference capabilities provided by the Spacecraft-EVC-MSFN equipments. The simplest way to describe these capabilities is to say that, given the condition of one or two EVC's on the lunar surface and the CSM in line-of-sight of the LM, any party in the loop can talk to any other party in the loop by relaying through the modules and ground stations as required. Any party can interrupt a conversation which is going on merely by speaking up. It is impossible for any party to lock up the system and prevent another party from entering it, such as has happened in previous manned space flights.

The space vehicle telemetry capabilities are presented in Figure 24. Primary telemetry capability is provided for the launch vehicle during the launch and orbital phases using the VHF/PCM equipment, and during injection and post-injection using S-Band/PCM equipment. During launch, there is no backup to the VHF capability. In orbit, the S-Band FM transmitter provides a backup to the VHF equipment. During the injection and post-injection phases, the coherent S-Band system in the launch vehicle provides a backup to the S-Band transmitter.

Primary telemetry transmission for the CSM and LM are provided using the coherent S-Band equipments. In addition, the CSM can dump recorded data via its separate FM transmitter. The LM transmits low bit rate telemetry from the LM and the CSM during descent and ascent for recording and subsequent to playback to earth. The backup to the two S-Band systems are the redundant equipments; also, the two Module telemeters utilize both block and circuit redundancy. There is no backup to the LM-to-CSM low bit rate capability.

Telemetry is received in the primary mode for the EVC over the 259.7 MC carrier and relayed by the appropriate Module to the earth at S-Band. There is no backup to EVC telemetry capability.

The primary launch vehicle tracking aids during launch and orbit phases are the C-Band and Azusa transponders as shown in Figure 25. During the injection and post-injection phases, the CSM S-Band system is the primary tracking system for the launch vehicle. Backups for the launch vehicle for tracking are as

shown here. Primary tracking for the CSM and the LM is provided by the S-Band coherent equipments. These equipments allow the MSFN to two-way doppler tracking and ranging on the spacecraft Modules. The range ambiguity inherent in two-way doppler tracking is resolved by transmitting a particular type of pseudo-random binary code, called a Ranging Code, to the vehicle-borne transponder where it is turned around and retransmitted to the MSFN. The code length is greater than the round-trip transit time to the moon. Comparison of the in-coming code at the MSFN station with a stored model of the code allows range ambiguity resolution. The backup for these primary equipments for tracking are the redundant equipments on each Module.

Primary equipment used by the launch vehicle for updata is the S-Band coherent system and a digital decoder, and backup capability is provided by its inherent redundancy. The primary system for updata in the CSM is provided by the coherent S-Band receiver and a Digital Command Decoder. Capabilities provided by this system are as shown in Figure 26. The backup to the RF equipment supporting this capability is the redundant S-Band receiver; there is no backup to the decoder. The voice link will provide the necessary redundancy for computer update and real time commands.

There is no updata capability provided for the LM.

The real time ground command functions to the CSM for control of the data system are as shown in Figure 27. Note that the reset switch command resets all switches to the position that they were in prior to the transmission of any real time commands.

The next several figures present a review of the usage of the RF systems by mission phase. Figure 28 illustrates the use of the coherent S-Band system and a separate S-Band transmitter in conjunction with the S-Band Omni antenna system during the ascent and earth orbital phases of the mission. Figure 29 illustrates the usage of these same equipments for the translunar and transearth coast phases, but note that during these phases these equipments are normally operated in conjunction with the High-Gain antenna on the CSM, rather than the Omni. Failure in the High-Gain antenna would require use of the Omni antenna resulting in the necessity of changing to the low telemetry bit rate. Figure 30 illustrates the usage of equipments when the CSM and LM are separated but with the LM not yet on the lunar surface. The significant difference here is that the LM communicates directly with the CSM by two-way voice and VHF telemetry transmission during descent and ascent phases as well as with the MSFN at S-Band.

Also shown on this figure are the frequencies used by the CSM and an EVC. The next two figures, 31 and 32, illustrate communications between the earth and the two spacecraft modules with the LM on the lunar surface and one EVC out and with two EVC's out. The frequency utilization shown here is the same as that previously described. However, should LM receiver capture or interference problems occur with two EVC's out, the EVC not transmitting telemetry can manually switch to his reverse frequency equipments, which will allow him to carry on duplex voice communications with the other EVC.

The next group of figures present the percentage availability of functional capabilities by mission phases. Referring to Figures 33 and 34, during launch, injection and translunar insertion, the primary functions of voice, high bit rate telemetry, ranging, tracking, and updata are available 100 per cent of the time. No playback will be utilized during these phases. The 26 per cent figures shown for parking orbit are based on the coverage capability for the first two orbits of the nominal mission. The ground-tracking capability is also shown as being 26 per cent. The 74 per cent figure shown for low bit rate telemetry was derived by assuming continuous low bit rate recording when the CSM is in line-of-sight of a ground station. The 20 per cent figure for playback is slightly less than the 26 per cent figure for the other functions because playback to a MSFN station will not begin until lock up of the primary system has been achieved. In Figure 34 note that voice, tracking and updata capabilities are available 100 per cent of the time for the three mission phases shown. The percentages shown for telemetry transmission are based on normal transmission of low bit rate telemetry with short periodic transmissions of high bit rate telemetry in order to conserve prime power during the coast phases. The one per cent figure shown for ranging on both of these charts is representative of the fact that very little actual ranging is required in order to maintain a valid range tally. The two per cent figure shown for TV is a guesstimate of actual TV usage which has not yet been determined, and is not caused by any limitation of equipment or power availability.

The next four figures, Figures 35 through 38, present the lunar phase communications usage. These figures are self-explanatory once it is understood that the 60 per cent figures are based on the fact that for 40 per cent of a lunar orbit the modules are behind the moon. The numbers shown for television usage, both in lunar orbit and on the lunar surface, are again guesstimates and again do not represent either equipment or power limitations. The normal operation for LM telemetry transmission during descent

and ascent is for the LM to transmit low bit rate telemetry to the CSM at any time that the LM is not in line-of-sight of the MSFN. Whenever LM-MSFN signal lock-up is achieved, the LM will immediately switch to high bit rate for transmission directly to the MSFN via S-Band.

Figure 39 presents capability utilization fo.' transearth coast and entry phases. The same comments as applied to Figure 34 apply here. Figures 40 and 41 illustrate the utilization of equipment during the recovery phase. The HF equipment is used for long range direction finding; the VHF equipments are used for short range direction finding and communications with recovery aircraft and ships. Finally, a swimmer hardline is available to allow direct voice communications between the astronauts in the command module and frogmen in the water. In Figure 41, the 40 per cent utilization of the VHF beacon is achieved by a power programmer which automatically cuts the power on for two seconds and off for three seconds continuously for 48 hours. Manual programming is employed to enable utilization of the HF equipment in the following manner: six minutes of carrier transmission, two minutes of single sideband voice transmission, and 52 minutes listen only, repeated hourly for 48 hours. This power programming is required in order to provide 48 hour post-landing recovery communications capability.

The last figure presents the status of the Unified S-Band Compatibility Test Program being conducted at the Manned Spacecraft Center with the cooperation of Goddard Space Flight Center, Marshall Space Flight Center, Jet Propulsion Laboratory, and contractor personnel. The Block I Engineering Model, Block II Engineering Model, Launch Vehicle Command and Communication System, and Block I Production Model compatibility tests (using an acutal MSFN receiving system) have been completed, and post-test reports have all been issued except for the latter test report which is due to be published in early July. The Block II CSM-MSFN compatibility test began upon receipt of the Block II S-Band communications system. The gap in the Block II test program results from lending Goddard Space Flight Center the Block II equipment for fly-by tests.

Questions and Answers

COMMUNICATIONS

Speaker: Douglas R. Broome

1. Can the transmission from the lunar surface experiments package be received during the ascent phase and during conclusion of mission?

   ANSWER - Yes.

2. Will second lunar landing affect the operation of the first lunar surface experiments package?

   ANSWER - No. A difference in transmitting frequency has been made.

3. Why is TV used only 5% of the time?

   ANSWER - Usage is based on power limitations.

NASA-S-66-6039 MAY

SPACECRAFT FUNCTIONS

	CSM	LM
S-BAND		
• VOICE	×	×
• TELEMETRY		
• HIGH BIT RATE (51.2 KBS)	×	×
• LOW BIT RATE (1.6 KBS)	×	×
• RANGING	×	×
• TRACKING	×	×
• UP DATA	×	
• VOICE/DATA PLAYBACK	×	
• TELEVISION	×	×
• SCIENTIFIC	×	
VHF		
• VOICE	×	×
• TELEMETRY (LM TO CSM)		×
• RECOVERY-VOICE/BEACON	×	
HF		
• RECOVERY-VOICE/BEACON	×	

Figure 1

NASA-S-66-5029 JUNE 1

LAUNCH VEHICLE FUNCTIONS

	S-IC	SII	SIVB	IU
S-BAND				
TELEMETRY				X
TRACKING				X
RANGING				X
UPDATE				X
VHF				
TELEMETRY	X	X	X	X
RANGE SAFETY	X	X	X	
C-BAND				
TRACKING				X

Figure 2

NASA-S-66-5999 JUL 5

MSFN CAPABILITIES

CALL SIGN	STATION	S-BAND							VHF		HF	C BAND
		VOICE	SC. SIVB & ALSEP TLM	TRACKING & RANGING	UP DATA	TV	ANT. DISH (FT) 30	ANT. DISH (FT) 85	VOICE	SIVB TLM	VOICE/ BEACON	SIVB TRACKING
MLA/CNV	CAPE KENNEDY	X	X	X	X		X		X	X		X
BDA	BERMUDA	X	X	X	X		X		X	X		X
GBM	GRAND BAHAMA	X	X	X	X		X		X	X		X
GTI	GRAND TURK								X	X		X
ANG	ANTIGUA	X	X	X	X		X		X	X		X
ACN	ASCENSION	X	X	X	X		X		X	X		X
CYI	GRAND CANARY	X	X	X	X		X		X	X		X
MAD	MADRID	X	X	X	X	X		X				
CRO	CARNARVON	X	X	X	X		X		X	X		
CNB	CANBERRA	X	X	X	X	X		X				
GWM	GUAM	X	X	X	X		X		X	X		
HAW	HAWAII	X	X	X	X		X		X	X		X
CAL	PT. ARGUELLO								X			X
GDS	GOLDSTONE	X	X	X	X	X		X				
GYM	GUAYMAS	X	X	X	X		X		X	X		
TEX	CORPUS CHRISTI	X	X	X	X		X		X	X		

Figure 3

NASA-S-66-6631 JUL 6

MSFN CAPABILITIES(CONT)

CALL SIGN	STATION	S-BAND							VHF		HF	C BAND
		VOICE	S/C. SIVB & ALSEP TLM	TRACKING & RANGING	UP DATA	TV	ANT DISH (FT) 30	ANT DISH (FT) 85	VOICE	SIVB TLM	VOICE/ BEACON	SIVB TRACKING
S-1	INSERTION-SHIP NO. 2	X	X	X	X		X		X		X	X
S-2	INJECTION-SHIP NO. 2	X	X	X			12'		X		X	X
S-3	INJECTION-SHIP NO. 3	X	X	X	X		X		X		X	X
S-4	REENTRY-SHIP NO. 4	X	X	X			12'		X		X	X
S-5	REENTRY-SHIP NO. 5	X	X	X			12'		X		X	X
A/C	INSTRUMENTATION A/C (8)	X	X	X					X	X		
RECOVERY FORCES				X					X		X	

Figure 4

NASA-S-66-5569 JUL 5

MANNED SPACE FLIGHT NETWORK COVERAGE

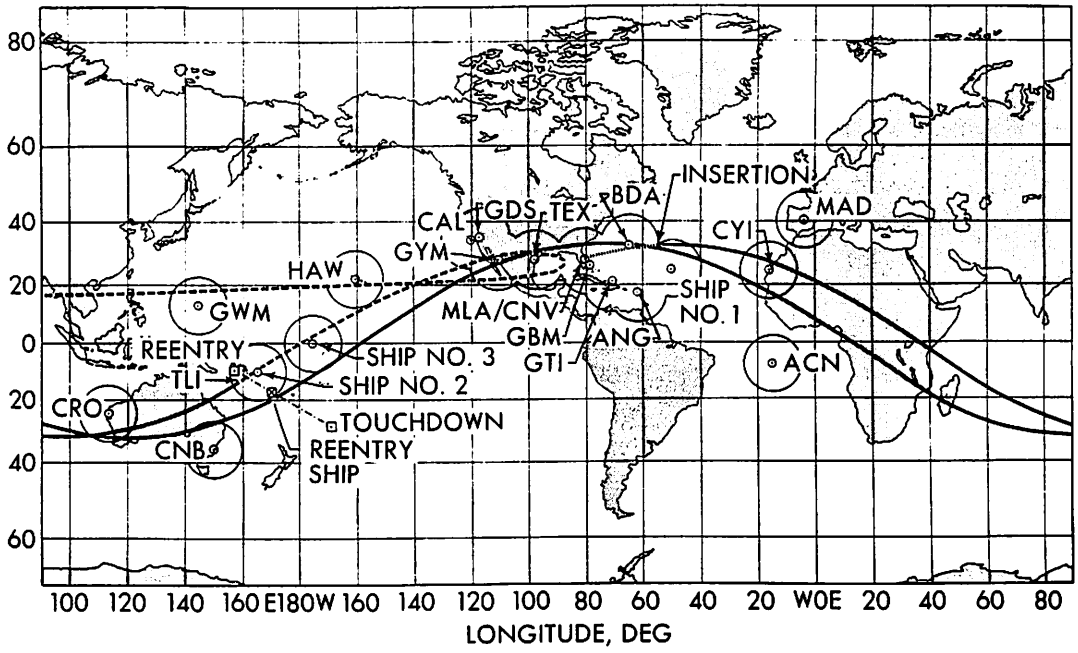


Figure 5

NASA-S-66-6839 JUN

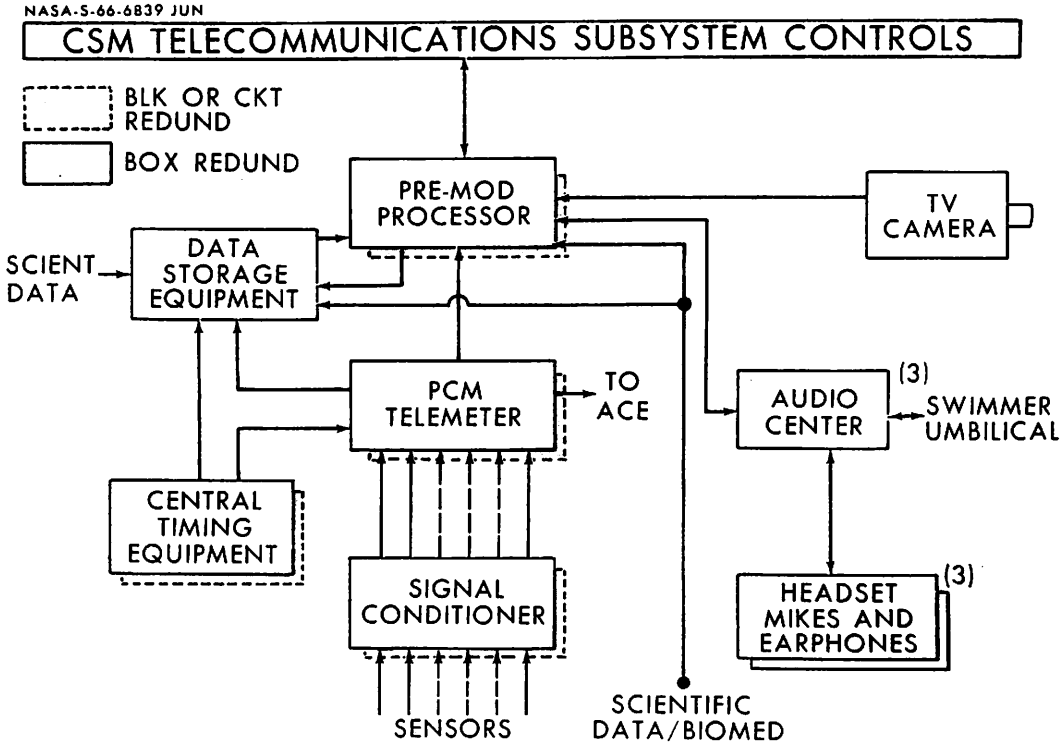


Figure 6

NASA-S-66-6844 JUN

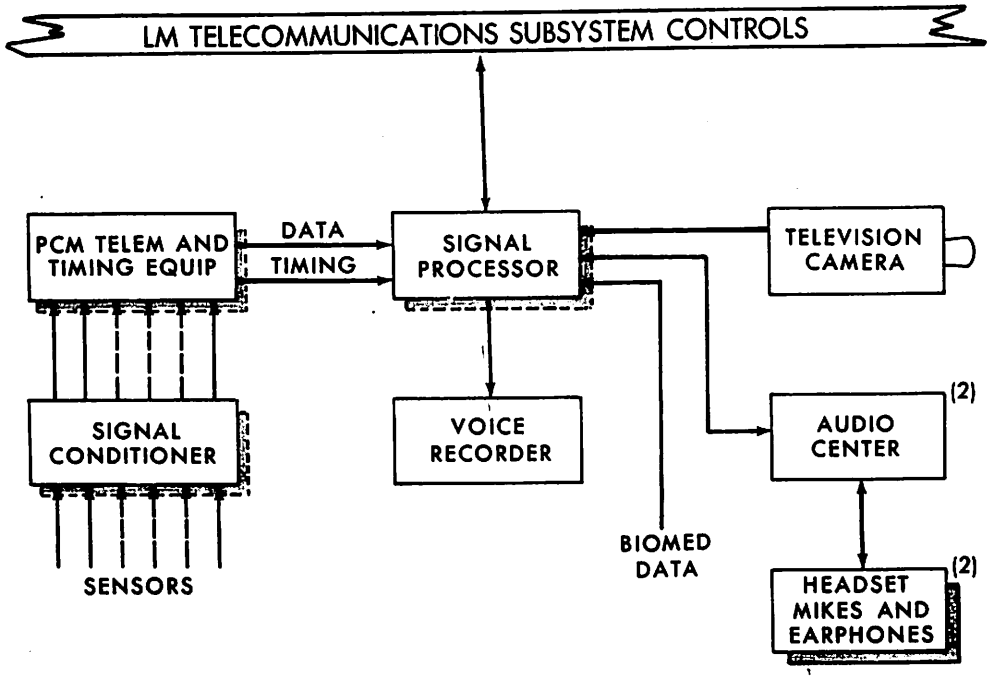


Figure 7

NASA-S-66-6624 JUL 6

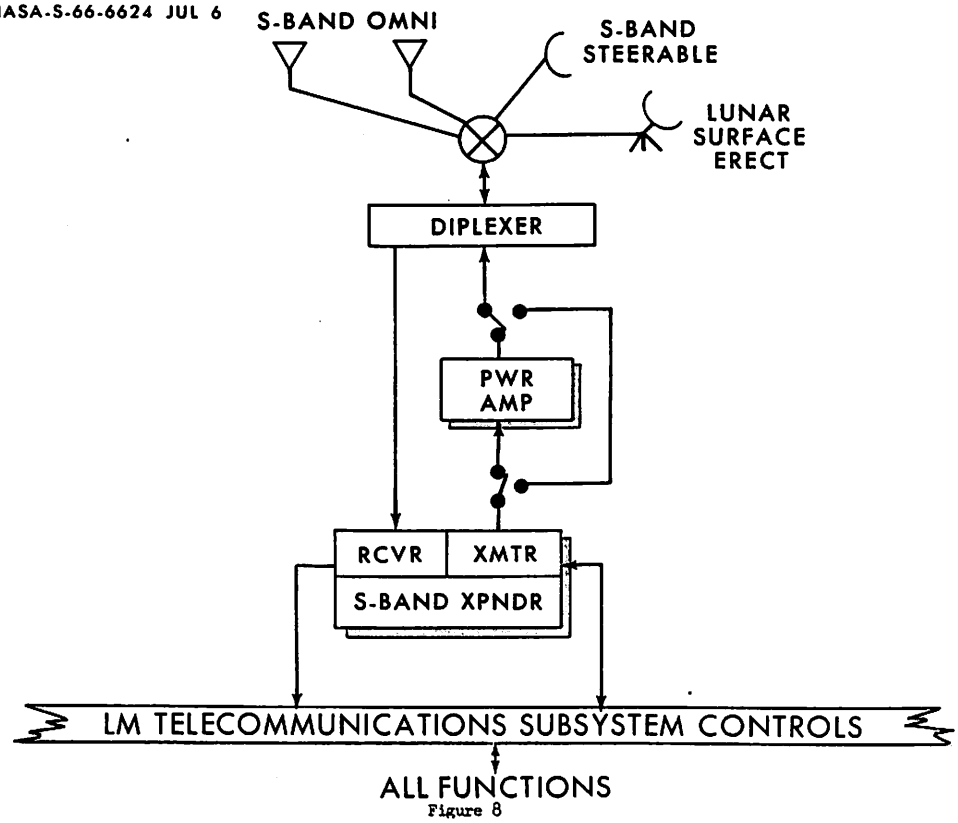


Figure 8

NASA-S-66-6632 JUL 6

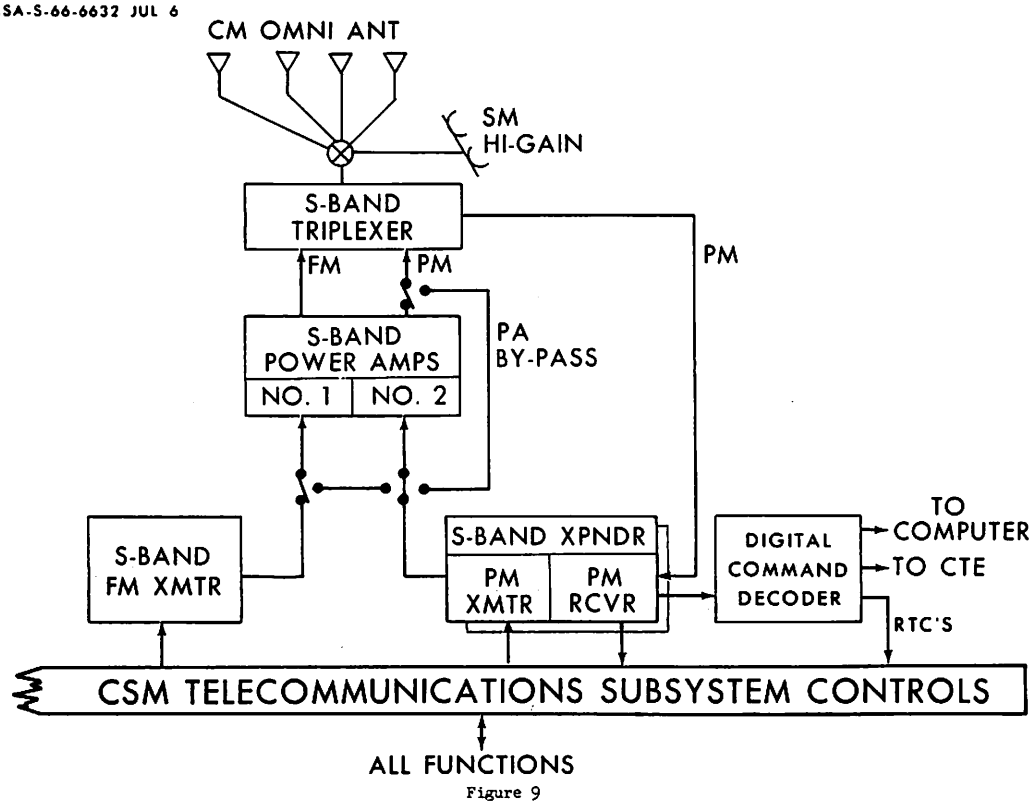


Figure 9

NASA-S-66-6840 JUN

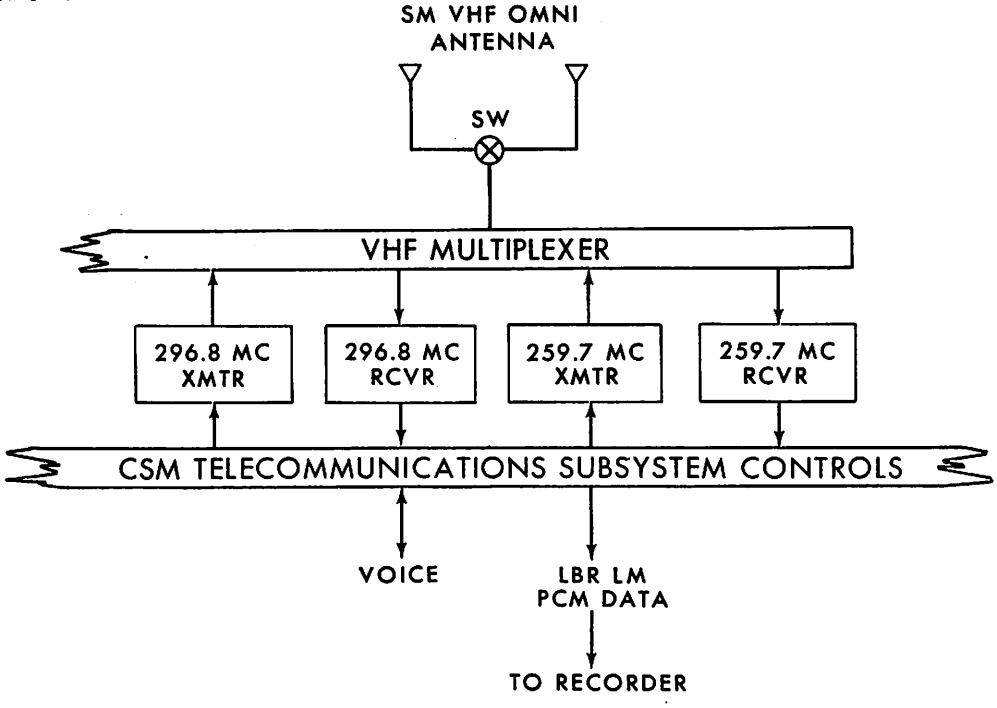


Figure 10

NASA-S-66-6843 JUN

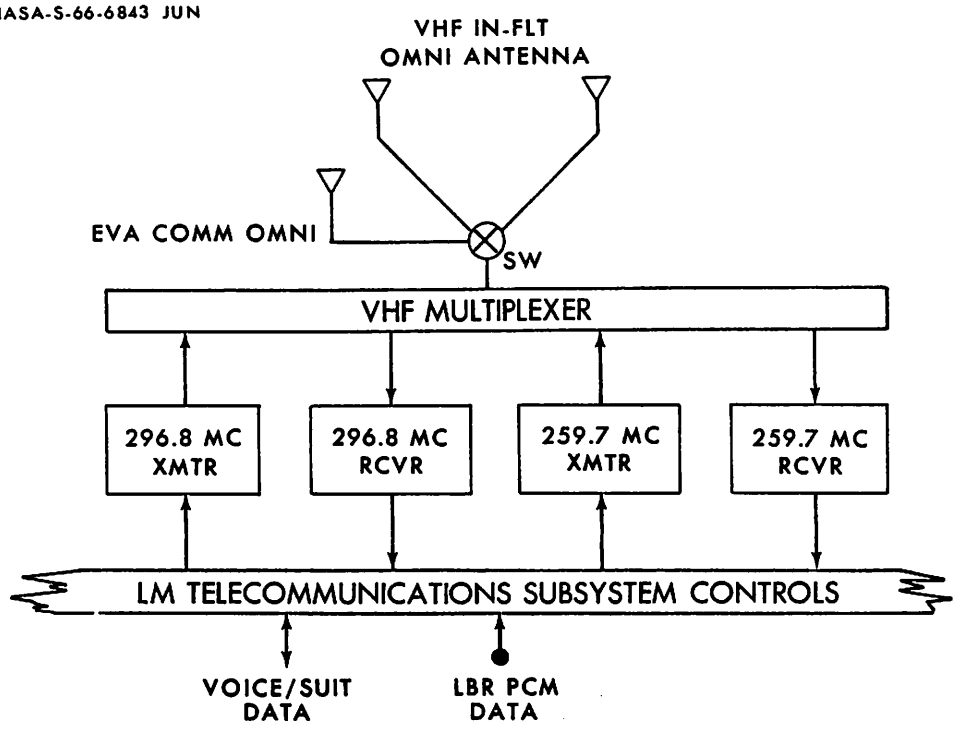


Figure 11

NASA-S-66-6841 JUN

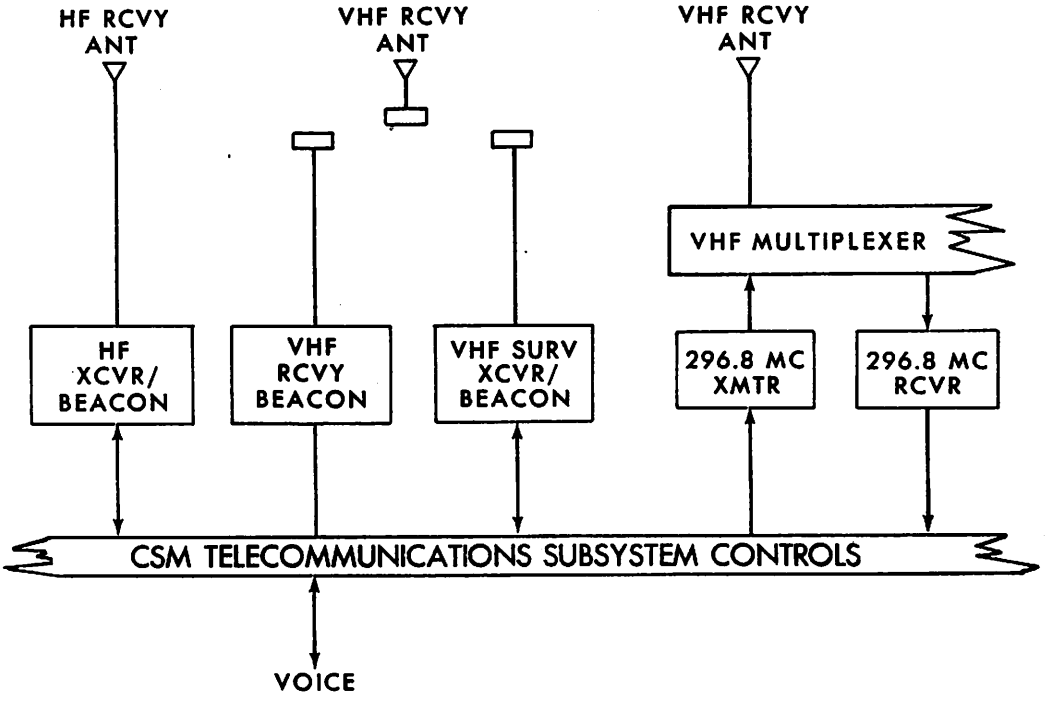


Figure 12

NASA-S-66-6036 MAY

S-BAND-CSM

- FREQUENCIES:
  - UP LINK–2106.4 MC
  - DOWNLINK–2287.5 MC
  - DOWNLINK–2272.5 MC

  COHERENT PAIR
- ANTENNAS:
  - 4 OMNI
  - 1 HIGH GAIN (WITH RF TRACKING)
- POWER:
  - LOW ¼ WATT
  - MED 5 WATTS
  - HIGH 20 WATTS

POWER LEVEL SELECTED AS A FUNCTION OF RANGE AND DATA RATE

Figure 13

NASA-S-66-6027 MAY

S-BAND-LM

- FREQUENCIES:
  - UP LINK 2101.8 MC
  - DOWNLINK 2282.5 MC

  COHERENT PAIR
- ANTENNAS:
  - 2 OMNI
  - 1 HIGH GAIN (WITH RF TRACKING)
  - 1 HIGH GAIN ERECTABLE (SURFACE)
- POWER:
  - LOW ¾ WATT
  - HIGH 20 WATTS

POWER LEVEL SELECTED AS A FUNCTION OF DATA RATE. ERECTABLE ANTENNA USED FOR POWER CONSERVATION.

Figure 14

NASA-S-66-6038 MAY

VHF-CSM

- FREQUENCY:
  - 296.8 MC
  - 259.7 MC

  DUPLEX
  - 243.0 MC - RECOVERY (VOICE/BEACON)
- POWER:
  - 5 WATTS
- ANTENNAS:
  - 2 FLIGHT (ON SM)
  - 2 RECOVERY (ON CM)

Figure 15

NASA-S-66-6037 MAY

VHF-LM

- FREQUENCY:
  - 296.8 MC
  - 259.7 MC

  DUPLEX
- POWER:
  - 5 WATTS
- ANTENNAS:
  - 2 OMNI-FLIGHT
  - 1 OMNI-EVA

Figure 16

NASA-S-66-5060 JUNE 1

HF - CSM

- FREQUENCY — 10.006 MC RECOVERY (VOICE/BEACON)
- POWER — 20 WATTS IN SSB
  5 WATTS IN DSB
- ANTENNA — 1 OMNI

Figure 17

NASA S 66 5028 JUNE 1

VHF - LAUNCH VEHICLE

- FREQUENCY
  - 231.9 MC, 252.4 MC, 240.2 MC, 235.0 MC, 256.2 MC, 244.3 MC — S-IC
  - 241.5 MC, 234.0 MC, 299.9 MC, 248.6 MC, 236.2 MC — S-II
  - 253.8 MC, 258.5 MC — S-IVB
  - 250.7 MC, 245.3 MC — IU
- POWER
  18 WATTS
- ANTENNA
  OMNI

Figure 18

NASA-S-66-5027 JUNE 1

LAUNCH VEHICLE (CONT)

- C-BAND TRACKING

FREQUENCY	5765 MC	RADAR
	5000 MC	AZUSA

- S-BAND (UP DATA, RANGING & TRACKING)

FREQUENCY	2282.5 MC	TRANSMIT
	2101.8 MC	RECEIVE
POWER	20 WATTS	
ANTENNA	OMNI	

- S-BAND TELEMETRY

FREQUENCY	2277.5 MC
POWER	20 WATTS
ANTENNAS	OMNI

Figure 19

NASA-S-66-5062 JUNE 1

LUNAR SURFACE EXPERIMENTS PACKAGE

- FREQUENCY — UPLINK 2119 MC
  DOWNLINK 2275.5, 2276.5 & 2278.5
- POWER — 1 WATT
- ANTENNA — HELICAL
- DESIGN LIFE — 1 YEAR
- DESIGN USAGE — 100%
- ONE PCM DOWNLINK CHANNEL WITH ALL EXPERIMENTS MULTIPLEXED
- TRANSMITS AT NORMAL BIT RATE OF 1060 BPS TO A 30 FOOT ANTENNA
- UPLINK COMMAND CAPABILITY
  - SYSTEM ON-OFF
  - ACTIVATE EXPERIMENTS
  - CHANGE EXPERIMENT MODES
  - (COMMAND LIST TO BE DETERMINED BY EXPERIMENT REQUIREMENTS)

Figure 20

NASA-S-66-6628 JUL 6

VOICE - PRIMARY

CSM	LM
CSM-EARTH S-BAND DUPLEX 243.0 MC RCVY 10.006 MC RCVY 296.8 MC RCVY S-BAND TAPE DUMP	LM-EARTH S-BAND DUPLEX
CSM-LM 296.8 MC SIMPLEX	LM-CSM 296.8 MC SIMPLEX
CSM-EVA 296.8 MC TRANSMIT 259.7 MC RECEIVE	LM-EVA 296.8 MC TRANSMIT 259.7 MC RECEIVE

Figure 21

NASA-S-66-5069 JUN

VOICE - BACKUP

CSM	LM
CSM-EARTH REDUNDANT S-BAND XPNDRS AND PWR AMPS 296.8 MC NEAR- EARTH ONLY	LM-EARTH REDUNDANT S-BAND XPNDRS AND PWR AMPS
CSM-LM ALTERNATE VHF EQUIPMENTS	LM-CSM ALTERNATE VHF EQUIPMENT
CSM-EVA 259.7 MC TRANSMIT 296.8 MC RECEIVE	LM-EVA 259.7 MC TRANSMIT 296.8 MC RECEIVE

Figure 22

NASA-S-66-5068 JUNE 1

VOICE RELAY AND CONFERENCE

- RELAY
  - EVA → CSM → EARTH (ONE EVA OUT)
  - EVA → CSM → EARTH (TWO EVA'S OUT)
  - EVA → LM → EARTH (ONE EVA OUT)
  - EVA → LM → EARTH (TWO EVA'S OUT)
  - EVA → LM → EVA
  - LM → EARTH → CSM
  - CSM → EARTH → LM
  - EVA → LM → EARTH → CSM
- CONFERENCE
  - LM ↔ EARTH ↔ CSM
  - EVA ↔ LM ↔ EARTH ↔ CSM
  - EVA ↔ CSM ↔ EARTH

Figure 23

NASA-S-66-5026 JUNE 1

TELEMETRY

SYS	L/V	CSM	LM	EVA
PRIMARY	LAUNCH VHF/PCM ORBITAL VHF/PCM POST INJECTION S-BAND/PCM	S-BAND CARRIER (REAL-TIME XMSN) S-BAND CARRIER (TAPE DUMP)	S-BAND CARRIER (REAL-TIME ONLY) VHF XMSN OF LBR TLM DATA TO CSM FOR RECORD AND PLAY B TO EARTH	259.7 MC CARRIER, RECEIVED IN LM OR CSM AND RELAYED TO EARTH VIA S-BAND OVER THE REAL-TIME VOICE SUBCARRIER
SECONDARY	LAUNCH NO BACK-UP ORBITAL S-BAND/PCM POST INJECTION S-BAND/COMMAND & COMMUNICATION SYSTEM	REDUNDANT S-BAND EQUIPMENTS REDUNDANT PCM BLOCK CIRCUITRY	AS FOR BLOCK II CSM (NO BACKUP FOR LM-TO-CSM TLM XMSN)	NONE

Figure 24

NASA-S-66-5025 JUNE 1

TRACKING

SYS	L/V	CSM	LM
PRIMARY	LAUNCH C-BAND AZUSA SYSTEM ORBITAL C-BAND RADAR POST INJECTION SC S-BAND	LUNAR MISSIONS S-BAND XPNDR (2-WAY COHERENT DOPPLER) PLUS RANGING)	SAME AS FOR CSM
SECONDARY	LAUNCH C-BAND RADAR ORBITAL SC S-BAND POST INJECTION C-BAND RADAR	REDUNDANT S-BAND EQUIPMENTS	SAME AS FOR CSM

Figure 25

NASA-S-66-5024 JUNE

UP-DATA

SYS	L/V	CSM	LM
PRIMARY	S-BAND COM- MAND AND COMMUNICA- TION SYSTEM ALL PHASES	S-BAND DIGITAL COMMAND SYSTEM • RTC'S • CTE UPDATE • COMPUTER UPDATE	NO UPDATA CAPABILITY
SECONDARY	BUILT IN REDUNDANCY	REDUNDANT S-BAND RCVR NO REDUNDANT DECODER (VOICE LINK IS REDUNDANT FOR COMPU- TER UPDATE & RTC'S)	N/A

Figure 26

NASA-S-66-5023 JUNE 1

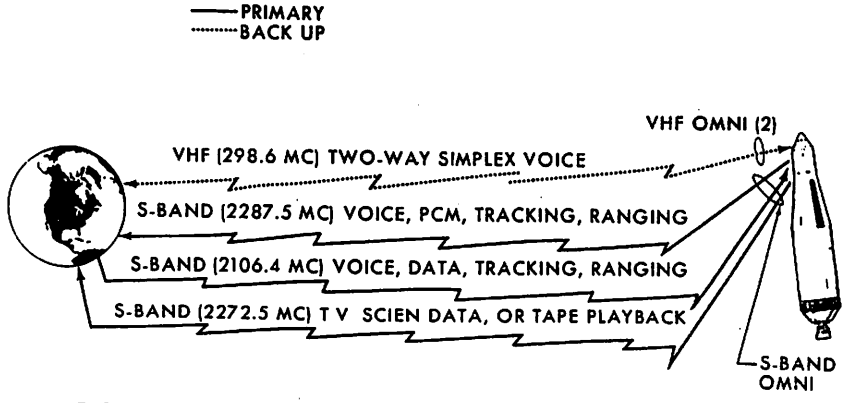
REAL TIME GROUND COMMAND FUNCTIONS TO CSM

- FLIGHT CREW ALARM SIGNAL
- PCM DATA RATE
- S-BAND POWER
- S-BAND RANGING
- S-BAND TAPE
- S-BAND PCM ON-OFF
- TAPE RECORDER
- RESET SWITCH
- ABORT REQUEST LIGHT
- SWITCH ANTENNA (HIGH GAIN TO OPPOSITE OMNI)

Figure 27

NASA-S-66-6001 MAY

ASCENT & EARTH ORBIT COMMUNICATIONS



TAPE PLAYBACK: RECORDED CSM VOICE, PCM, & SCIEN DATA

Figure 28

NASA-S-66-6629 JUL 6

TRANSLUNAR AND TRANSEARTH COAST COMMUNICATIONS

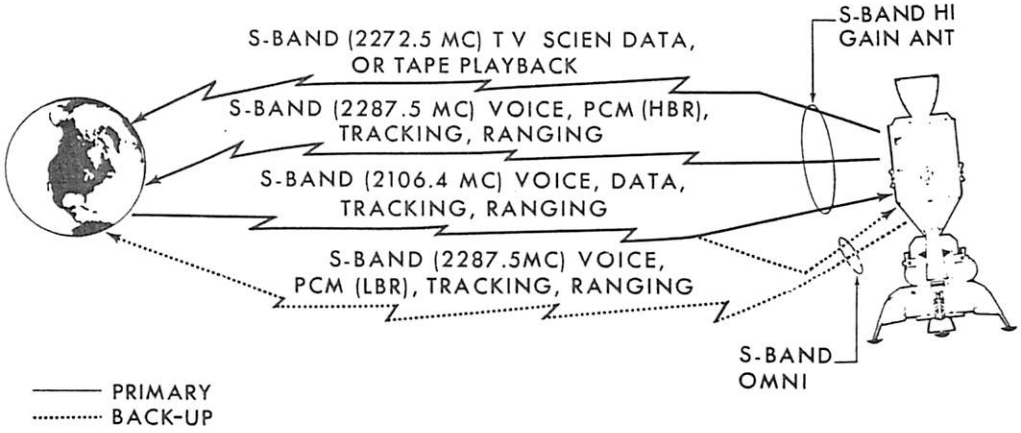


Figure 29

NASA-S-66-6018 MAY

CSM-LM EARTH COMMUNICATIONS LINKS

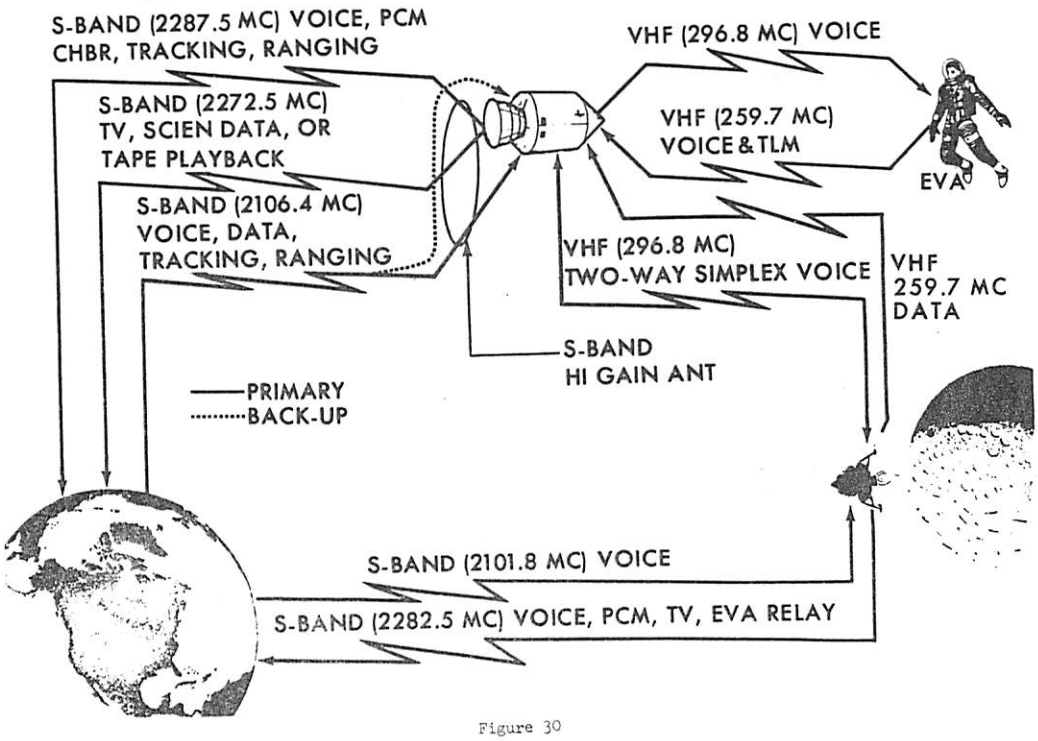


Figure 30

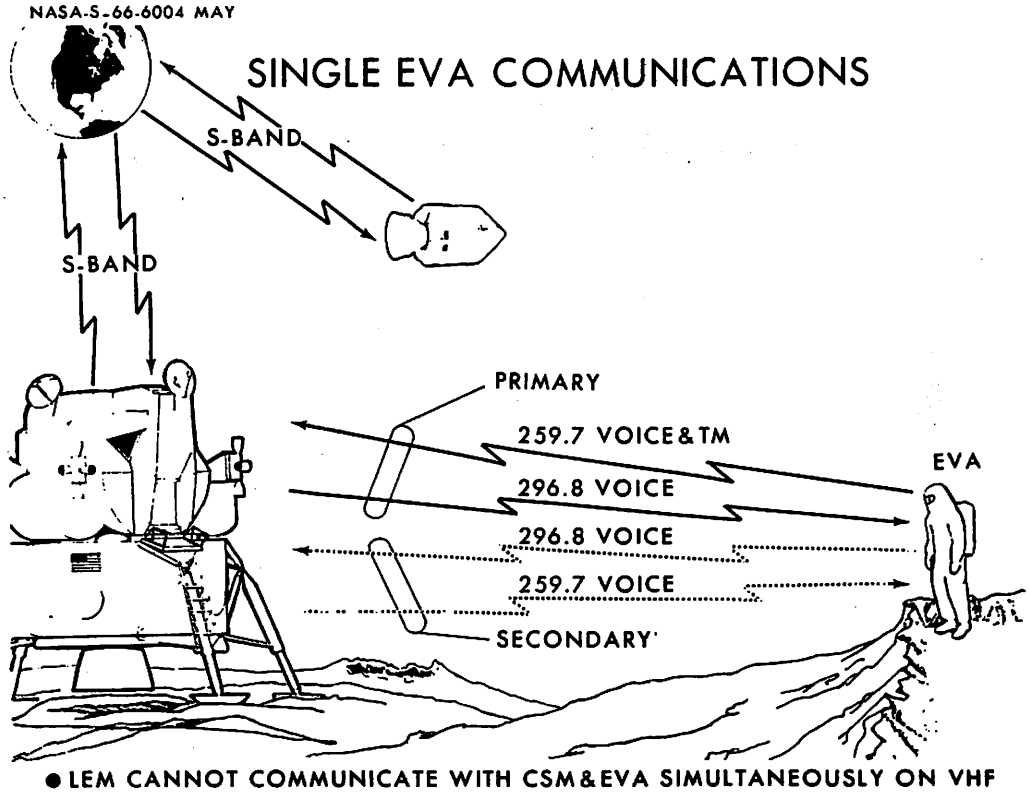


Figure 31

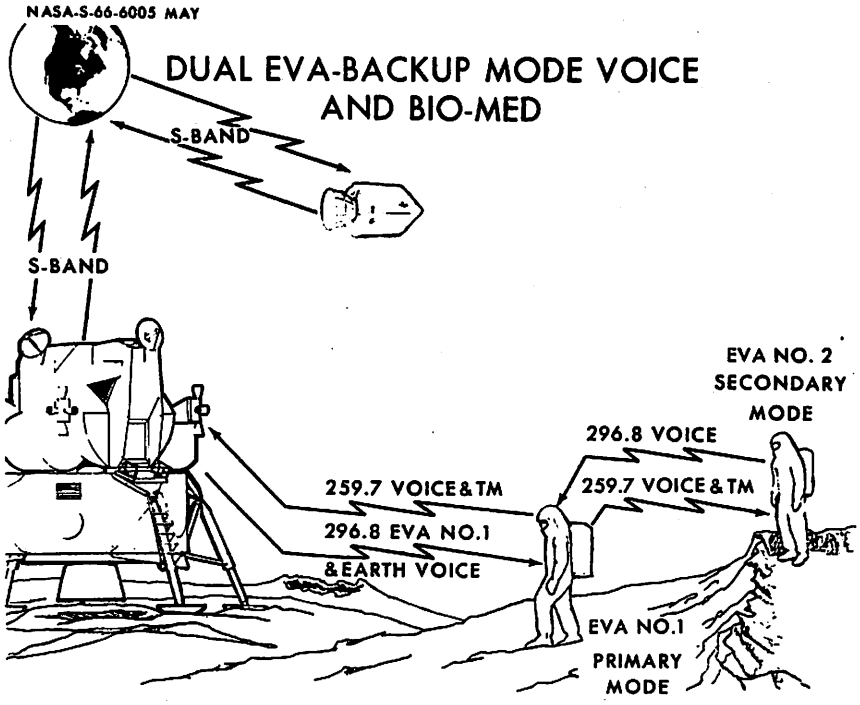


Figure 32

NASA-S-66-5067 JUNE 1

ASCENT AND EARTH ORBIT COMMUNICATIONS

	LAUNCH TO INJEC-TION	EARTH PARKING ORBIT	TRANS-LUNAR INJEC-TION
S-BAND - VOICE	100%	26%	100%
S-BAND - HIGH BIT RATE TELEMETRY	100%	26%	100%
S-BAND - LOW BIT RATE TELEMETRY	N/A	74%	N/A
S-BAND - RANGING	100%	1%	100%
S-BAND - TRACKING	100%	26%	100%
S-BAND - UP DATA	100%	26%	100%
S-BAND - VOICE/DATA PLAYBACK	0%	20%	0%
VHF - VOICE	100%	26%	100%

Figure 33

NASA-S-66-6621 JUL 6

TRANSLUNAR COAST COMMUNICATIONS

	TRANS-LUNAR COAST	MID COURSE CORREC-TION	LUNAR ORBIT INSERTION*
S-BAND - VOICE	100%	100%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	100%	100%
S-BAND - LOW BIT RATE TELEMETRY	90%	N/A	N/A
S-BAND - RANGING	1%	100%	100%
S-BAND - TRACKING	100%	100%	100%
S-BAND - UP DATA	100%	100%	100%
S-BAND - VOICE/DATA PLAYBACK	AS REQ'D	AS REQ'D	AS REQ'D
S-BAND - TELEVISION	2%	N/A	N/A

*WHILE IN LINE OF SIGHT OF MSFN

Figure 34

NASA-S-66-6627 JUL 6

LUNAR PHASE COMMUNICATIONS

	DUAL LUNAR ORBIT COAST	SOLO COAST		DESCENT		LUNAR STAY (LM)
		CSM	LM	CSM	LM	
S-BAND - VOICE	60%	60%	60%	60%	60%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	10%	10%	60%	60%	5%
S-BAND - LOW BIT RATE TELEMETRY	50%	50%	50%	0%	0%	95%
S-BAND - TRACKING	60%	60%	5%	60%	60%	100%
S-BAND - RANGING	1%	1%	1%	1%	1%	1%
S-BAND - UP DATA	100%	100%	N/A	100%	N/A	N/A
S-BAND - VOICE/DATA PLAYBACK	10%	10%	N/A	10%	N/A	N/A
S-BAND - TELEVISION	2%	N/A	N/A	N/A	N/A	5%

Figure 35

NASA-S-66-6619 JUL 6

LUNAR PHASE COMMUNICATIONS

	DUAL LUNAR ORBIT COAST	SOLO COAST		DESCENT		LUNAR STAY (LM)
		CSM	LM	CSM	LM	
VHF - VOICE	-	10%	10%	100%	100%	10%
VHF - TELEMETRY (LM TO CSM)	-	-		N/A	40-60%	N/A
LUNAL SURFACE EXPERIMENT PACKAGE						*90%

*LSEP DESIGNED FOR CONTINUOUS OPERATION ON LUNAR SURFACE FOR ONE YEAR

Figure 36

NASA-S-66-5081 JUN 3

LUNAR PHASE COMMUNICATIONS

	ASCENT		RENDEZVOUS & DOCKING		TRANS-EARTH INJEC-TION
	CSM	LM	CSM	LM	
S-BAND - VOICE	60%	60%	0%	0%	100%
S-BAND - HIGH BIT RATE TELEMETRY	60%	60%	0%	0%	100%
S-BAND - LOW BIT RATE TELEMETRY	0%	0%	100%	100%	N/A
S-BAND - TRACKING	60%	60%	0%	0%	100%
S-BAND - RANGING	1%	1%	0%	0%	100%
S-BAND - UP DATA	0%	N/A	0%	N/A	100%
S-BAND - VOICE/DATA PLAYBACK	10%	N/A	10%	N/A	N/A
S-BAND - TELEVISION	N/A	N/A	N/A	N/A	N/A

Figure 37

NASA-S-66-6623 JUL 6

LUNAR PHASE COMMUNICATIONS

	ASCENT		RENDEZVOUS & DOCKING		TRANS-EARTH INJEC-TION
	CSM	LM	CSM	LM	
VHF - VOICE	100%	100%	100%	100%	N/A
VHF - TELEMETRY (LM TO CSM)	N/A	40-60%	N/A	100%	N/A
LUNAR SURFACE EXPERIMENT PACKAGE	100%	100%	100%	100%	100%

Figure 38

NASA-S-66-6622 JUL 6

TRANSEARTH COAST AND ENTRY COMMUNICATIONS

	TRANS-EARTH COAST	MID COURSE CORREC-TION	ENTRY TO BLACK OUT
S-BAND - VOICE	100%	100%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	100%	100%
S-BAND - LOW BIT RATE TELEMETRY	90%	N/A	N/A
S-BAND - RANGING	1%	100%	100%
S-BAND - TRACKING	100%	100%	100%
S-BAND - UP DATA	100%	100%	100%
S-BAND - VOICE/DATA PLAYBACK	AS REQ'D	AS REQ'D	N/A
S-BAND - TELEVISION	2%	N/A	N/A

Figure 39

NASA-S-66-6009 MAY

RECOVERY COMMUNICATIONS

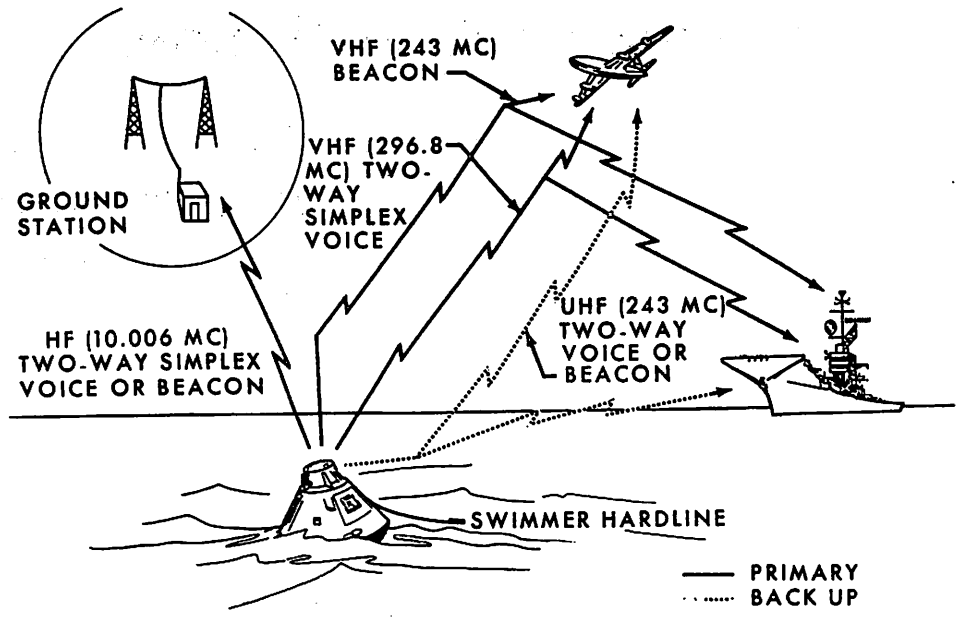


Figure 40

NASA-S-66-6620 JUL 6

POST LANDING COMMUNICATIONS

	POST LANDING
VHF - RECOVERY BEACON	40%
VHF - VOICE	3%
HF - VOICE	10%
HF - BEACON	3%

NOTE:

POSTLANDING SYSTEMS REQUIREMENTS ARE BASED ON A MAXIMUM OF 48 HOURS

Figure 41

NASA-S-66-6625 JUL 6

ELECTRONIC SYSTEMS TEST PROGRAM

SPACECRAFT-USB-MSFN TEST SCHEDULE

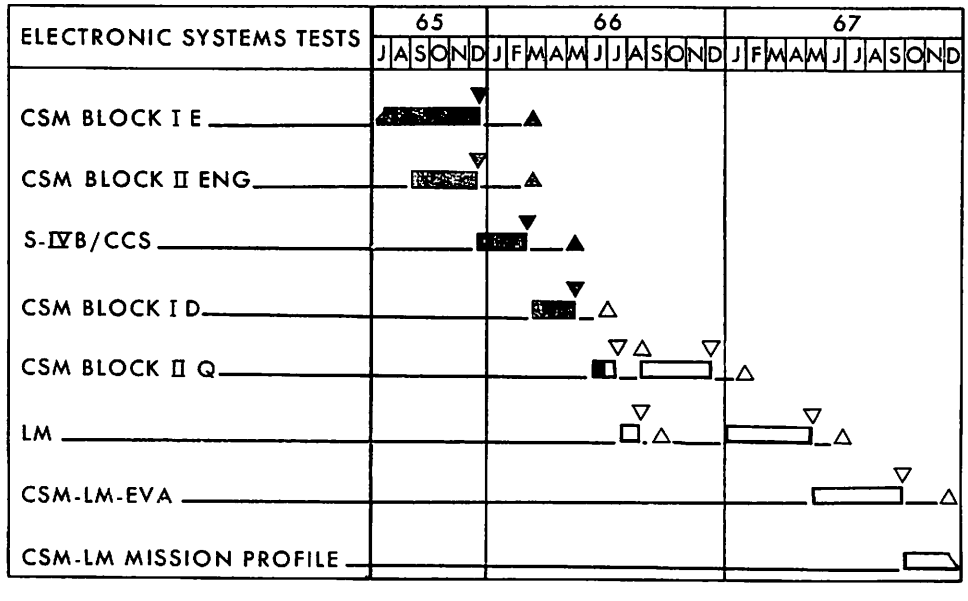


Figure 42

CREW TASKS AND TRAINING

by

J. P. Loftus
R. L. Bond

CREW TASKS AND TRAINING

The object of this paper shall be to provide an appreciation of the crew activities during the lunar landing mission. The emphasis is pictorial and qualitative rather than parametric and quantitative as in the papers presented to this point. We shall discuss examples of typical crew activity without emphasis upon detail because such detail does not lend itself to a presentation of this type and is often misleading when taken out of context. The training facilities will be discussed briefly not only in their customary role as training devices but as development tools for operating procedures and strategies. Crew activities shall be reviewed by mission phase.

NOTE: A motion picture film was used to furnish the nine sequences noted below and was an integral part of this paper. The film is available through the NASA-MSC film library and carries the NASA identification number S-66-411.

The sequences are: (1) Abort Studies, a brief sequence showing pilot reaction to system anomalies in a launch simulation; (2) Landmark Sighting, an animated view of a typical landmark sighting through the scanning telescope optics; (3) CM Active Docking, a pilot's-eye view of the lunar module during a command module active docking sequence; (4) Star Landmark, an animated view of what the navigator sees through the guidance optics during a star-landmark sighting; (5) Zero "G" Sequence, a short sequence filmed in the zero "G" aircraft showing transfer tunnel activity; (6) Lunar Landing Research Vehicle, a simulated lunar module descent trajectory flown by the LLRV at Edwards AFB; (7) Lunar Module Activity, a series of sequences showing typical lunar module cabin activity by the two crewmen subsequent to lunar module touchdown on the lunar surface; (8) Lunar Module Active Docking, a profile view showing the lunar module as the active vehicle in the docking sequence; and (9) Earth Approach, a film produced by inputting a computer with a moon-to-earth trajectory and letting the output drive an earth image on a cathode ray tube. One film frame represents four minutes of mission elapsed time. The resulting sketches indicate the apparent growth of the earth as seen by the homeward bound crew.

CREW REQUIREMENTS AND ORGANIZATION

The requirements placed upon the crew emphasize the role of the crew as a sensor, as a control element, and as a logic source. (Figure 1)

The crew participate as the abort initiation system in many parts of the launch. There are some failures requiring automatic abort in the early phase of the launch where failures can be abrupt and not suitable for crew action. A great number of the failures requiring abort can be detected more positively by the crew onboard the vehicle because of their capability, as a direct sensor, to confirm instrument indications by assessment of motion, vibration, and noise. The ground can observe some guidance failures which the crew might not perceive so readily and can request abort action.

In flight control functions the crew always acts in the sense of a programmer, selecting the suitable mode of the system for a given operation. The crew acts as a sensor in certain modes and as a control element in the manual and direct control modes.

The subsystem management role allows wide ranges of variation in the rate and manner of use of the consumables and in the programming of the systems. Crew action also provides a complex logic for use of the alternate modes and many forms of redundancy provided in each of the systems.

In navigation activities as in flight control functions the crew serves as a control element and as a sensing element.

The onboard mission management functions are primarily those where time is critical, where the crew by virtue of their presence in the vehicle have better data, and during those time periods in the mission during which they are out of communication with the ground complex.

The crew organization is an authority structure with duties being primarily assigned as a function of operating station rather than directly associated with an individual. (Figure 2) The crew is cross-trained and is trained as a team in the interest of enhancing reliability and to maximize the capacity of the crew to accept high work loads in certain portions of the mission.

TRAINING

The types of training are those with which we are familiar from Gemini and Mercury. (Figure 3) Systems training consists of briefings, textual materials, and actual participation in a number of spacecraft tests and engineering reviews. There are a number of specific task training elements related to particular manual skills, such as docking, extravehicular activities, and landing. Mission segment training is that type of training which is defined by the mission phase. Characteristic of this type of training are rendezvous, transposition docking, and entry. Specific mission training relates to those training activities determined by the trajectories and objectives of a particular mission. Illustrative of this class might be the lunar landmark briefings which would precede a particular mission to a particular site. Finally, there is experimental activity training which is both mission specific and general.

Training for normal modes of crew activity occupies perhaps the smallest portion of the total training time. (Figure 4) Training in malfunction recognition and response and emergency mode activity occupies the largest portion of crew time in training. The requirement for both prompt action and for high reliability establishes the requirement; the complexity of the system makes it time consuming. We have again emphasized the training of the crew as a team not only amongst themselves but in a number of exercises in conjunction with the flight controllers.

A summary listing of the major subsystems with which the crew training deals is presented in Figure 5. There is cross-summarization between the command module (CM), the launch vehicle, and the lunar module (LM) in this listing. The extravehicular mobility unit (EMU) is often overlooked, but there is significant training involved in the effective use of this complex.

Training facilities represent rather a large array of equipments. (Figure 6) This is due to the dimensions of the problem, to the limits of particular devices, and to the requirement to be able to concurrently train several crews.

The systems trainers are animated schematics wherein the spacecraft system is laid out on free-standing vertical panels. (Figure 7) The spacecraft controls and displays

related to these subsystems are shown on the upper portion of the left panel. Located on the lower portion is a panel where malfunctions and anomalies can be created in the system so that the character of the malfunction and the symptoms associated can be studied.

There exists a family of devices known as part task trainers. We have mockups of both of the vehicles involved - the CM and the LM. (Figures 8 and 9) These devices are used in training in the many stowage and housekeeping functions, extravehicular activity, and other geometry defined activities. The dynamic crew procedures trainer, currently in the Gemini configuration, can accept a CM configuration. (Figure 10) This device is used to provide the dynamic environment associated with various launch conditions and entry conditions and it is primarily devoted to training in recognition of failures requiring abort. It provides sound, motion, and visual cues suitable for this environment since these are a significant part of the development of crew responses for such emergency conditions. This device can also accept a LM configuration for training in the dynamics of the LM motion in landing and in ascent. (Figure 11) The transposition and docking trainer is shown in the Gemini configuration in Figure 12, but it can be configured with the LM or the CM as the active element in the docking configuration. The lunar landing research vehicle (LLRV) shown in Figure 13 is illustrative of the lunar landing training vehicle which can provide a flight environment simulation for the terminal portions of the LM landing. We will discuss this device more at a later time. The egress trainer is used for training the crew in the post landing and recovery phases of the mission and integration of their activities with those of the recovery forces. (Figure 14)

The Apollo Mission Simulator (AMS) and the Lunar Module Mission Simulator (LMS) represent the major complexes in the training equipment array. (Figures 15 and 16) The AMS is controlled by the computer assembly shown in the background. There is a model house for the generation of the docking images, and an instructor's console where the device can be programmed and malfunctions can be inserted. The CM is nested among a large array of infinity image optical systems which are reflective optical transfer systems for providing images to the windows of the vehicle. There is also an assembly for providing the images to the guidance optics. The magnitude of this device can perhaps be appreciated when one considers that it has a 184 thousand word memory system with four central processors

in parallel. The processors have $2\frac{1}{2}$ microsecond add times, effectively giving a nanosecond computer capability. There is a fully buffered channel capacity of one million words per second. The LMS is only slightly smaller. The computer complex contains three computers with a 98,000 word memory capacity. It has an image generation system for the CM docking portion of the mission and for the lunar touchdown portion of the mission. Film provides the approach image generation, again through the infinity image window systems. The window on the right side is shown in Figure 16 and the left window is shadowed in order that you can see the orientation of the crew within the vehicle. There is an image system for presentation to the overhead window which is used for observation of the horizon during powered descent and through which the docking sightings are made. These two devices can be operated in an integrated mode and can be integrated with the mission control center (MCC) to provide complete operational rehearsals of the mission. As a standard of comparison one may note that the BMEWS operational program required approximately 100 thousand instructions.

In addition to these devices, which are primarily oriented to the training and operational rehearsals, we have certain engineering devices which are used for special functions and tests as well as for certain specific types of training. The centrifuge shown in Figure 17 accepts a fixture which has three couches and all of the controls and displays suitable for launch, entry, and other high acceleration portions of the mission. There is an Air Force KC-135 which is used to simulate zero g and 1/6 g periods, again primarily for development testing but also for certain selected types of training primarily associated with extravehicular activities and with certain types of docking activity. (Figure 18)

Representative of another large class of engineering simulation devices is the lunar landing research facility (Figure 19) which is primarily a development device but which can be used for training in certain selected portions of the mission.

In addition to the devices shown we have a large number of engineering simulation facilities at the various contractors - at North American Aviation, Inc. (NAA), Grumman Aircraft Engineering Corporation (GAEC), and Massachusetts Institute of Technology (MIT).

CREW ACTIVITIES PROFILE

The mission for which these devices are used to prepare the crew involves a number of significant phases. In Figure 20 a typical profile of the mission shows two of the major constraints which govern the flight plan. The control events, shown as black diamonds are the launch, translunar injection, transposition docking, and midcourse corrections. The dark bars show the sleep periods which are allocated to the crew. The initial sleep period is obviously dictated by the time of the crew's waking and preparation for the launch. The schedule of activities must be arranged so that at lunar orbit insertion and the lunar descent portion of the mission the crew is suitably refreshed to enter a period of rather high density activity. The profile shown here reflects a constraint that sleep periods shall be at least six hours and that such periods shall occur at intervals not greater than 18 hours. It also presumes that three crewmen sleep at the same time. This takes advantage of the capacity of the ground to view the vehicle at all times and to wake the crew in the event that some anomaly is observed.

In the lunar portion of the mission an exploration period is scheduled immediately after the landing. The logic which dictates this schedule is that the portable life support systems (PLSS) are at this time fully charged and the vehicle is pressurized. If we perform an exploration at this time we can have an exterior check of the vehicle shortly after landing. The crew can donn the PLSS and the thermal meteoroid garments (TMG) in a pressurized cabin, depressurize the cabin, perform the exterior activities, return, pressurize the cabin, and then doff this gear and re-charge the PLSS in a pressurized configuration. If we had chosen to sleep first, an alternative which we can choose in real time, back-to-back exploration periods would require re-charging of the PLSS in an evacuated ascent stage. There are procedures for doing so, but in the early missions it seemed desirable to have the additional benefit of the pressurized cabin. This also allows a suitable period of rest prior to the activities of ascent into lunar orbit and transearth injection.

Figures 21 through 24 summarize the proportions of mission time - a total of 216 hours - and the proportions of the total crew man hours - 648 - which are devoted to various types of specific activity. The control tasks, as one would expect, occupy a relatively small proportion of the total mission time

though they are obviously the most critical events. The relative proportions devoted to various activities are obviously a function of the particular details of the mission, though one may expect certain proportional relationships to remain constant for all missions. Notice that in such activities as monitoring, which is done primarily by one man on a watch configuration, there is some nominal accumulation of potential crew time as opposed to mission time. This can be more apparent than real. A number of tasks require activity by all three crewmen while others are performed primarily by one man at a given time. Figure 23 illustrates mechanical manipulations - the changing of stations within the vehicle, the donning and doffing of the suit, the manipulation of the docking equipment. The exploration period, effectively the payload, represents approximately 2% of the total mission time and of the total crew man hours. A third of the mission time in crew man hours is devoted to sleep. In this particular summary 31% of the mission hours or 46% of the crew man hours are not scheduled.

The validity of such numbers is only as good as the analysis of the times required to perform particular functions, and is highly sensitive to the details of the particular flight plan. Experience to date indicates that most activities take substantially longer in flight than during simulations, a factor of two or more being quite common.

There is a further artifact pertinent to the form of this summary. It does not distinguish usable free time from blocks of time of no value. Time, in the sense of crew man hours, may appear to be available but constraints such as vehicle arrangement, subsystem configuration, or location of crewman may vitiate its utility. It is also important to schedule some "free" time in the same fashion as planned "hold periods" in a countdown.

AVAILABLE WORK AREA

Figure 25 illustrates the available volumes in each of the vehicles. The geometry of the CM is dictated by the entry requirement, the offset center of mass providing the effective L/D, and by the sweep volume required by the couch in land landings. The crew compartment volume is defined as the lightly shaded area and the effective free volume is that area not occupied by such objects as the couch, the PLSS, and the various other stowed equipments. The LM geometry is

defined by the requirement to counter balance the loads around the ascent engine. Here also there is a reduction in the total pressurized volume by the various equipments. Comparable figures for the Gemini vehicle are 80 cubic feet of pressurized volume and 50 cubic feet of effective free volume. We expect that the larger volume and its favorable arrangement will enhance the effectiveness of a number of types of crew activity.

CONTROLS AND DISPLAYS

The CM main display console shown in Figure 26 is color coded to indicate the various proportions of functional activities allocated display space. The exact proportions are not shown since large numbers of the displays are time shared. It is significant, however, to note the relatively large areas devoted to the sustaining systems. Since this vehicle must accomplish the long time operations of the mission a large number of levels of redundancy and cross-switching are available within the environmental and the electrical power generation and distribution systems. The sequential events and staging associated with launch and entry are also a major function of the CM.

The LM panel (Figure 51) is predominantly given to propulsion and flight control. This reflects the character of its portion of the mission and the simpler configuration of its sustaining systems. The use of batteries for electrical power, and bottled gas rather than cryogenic stores for the life support system make the sustaining systems simpler though less flexible.

MISSION OUTLINE

At this point it would be well to review the configuration of the vehicle at each of the phases of the mission and to examine the crew functions which characterize that phase of the mission.

Launch

During launch the crew is primarily concerned with monitoring the characteristics of the launch vehicle and its flight performance, maintaining communication, and monitoring the condition of the spacecraft. (Figure 27) The vehicle at this time is arranged with a number of soft goods stowed on the floor,

the lithium hydroxide (LiOH) cannisters used in the environmental control system (ECS) are installed along the lower edge and a large number of equipments are stowed on the upper wall and in a number of compartments on both sides of the vehicle. (Figure 28) The crew is not shown in the launch configuration because they would obstruct the view but they are beneath the array of harnesses, oxygen umbilicals, and communications lines. It is difficult to describe, in a sense that can be appreciated, the monitoring activity in launch. However, Figure 29 outlines a typical sequence of events for one selected launch condition in which an abort would be necessary subsequent to 61 seconds of powered flight.

Once an abort decision is taken, either by the crew or on ground request, the crew has the capability to monitor a large number of the events which take place automatically. The timer would reset so that the time sequence characteristic of the abort could be monitored. There is a capability to note the acceleration of the CM by launch escape motor both by accelerometer and directly. If this does not occur the event can be commanded by the crew. Those items shown with question marks in the figure are items checked to confirm the functioning of the automatic sequencer; verifying that the CM reaction control system (RCS) has pressurized confirms that the relays have actuated which cut the tension ties, deadface the CM and service module (SM), and arm the CM batteries to provide electrical power in that configuration and pressurize the CM RCS for attitude control. Eleven seconds after the initiation of the abort the canards would deploy. If they do not deploy the crew can command that event through an independent path. The crew can confirm that the earth landing system logic has been armed and provide an alternate path for the jettison of the launch escape tower. Drogue chute deployment can be confirmed at 24,000 feet or below by reference to the barometric altimeter, and in the event that function does not occur they can again provide an alternate command. Once on the main parachutes, no longer requiring the attitude control of the CM RCS, they would close the cabin pressure relief valves, command the necessary dumping and purging of the CM RCS, re-open the valves which have been closed to prevent injestion of the gases and, after landing, release the main chute.

The identifying characteristic of the abort after 61 seconds is that there is not an automatic dumping of the RCS propellants, since above the altitude attained by that point in launch one may require use of the attitude control system for orientation.

Earth Orbit

In the earth orbit phase of the mission the crew has control of the total space vehicle, including the S-IVB, through the interface between the spacecraft guidance computer and launch vehicle guidance computer. After earth orbit insertion the crew would realign the inertial measurement unit (IMU) using the guidance optics. Provided there were time, as would be the case with translunar injection on the second orbit, they can perform certain navigation activities in earth orbit. (Figure 30) These activities would be done to confirm the condition of the onboard system since the ground stations would provide the primary navigation. The technique used for orbital navigation is, however, also used in lunar orbit, with known orbital conditions, to assess the altitude of landing sites.

Prior to translunar injection the crew would fine align the IMU (Figure 31) and begin the countdown for the ignition of the second burn of the S-IVB. The crew can inhibit this command should their onboard data indicate that the vehicle is not ready for the mission. There is, however, only a single opportunity due to the requirements of the S-IVB propellants.

Translunar Injection

During the translunar injection burn, as during all other propulsive maneuvers, the crew will have prepared the vehicle for peak power loads by bringing both batteries and fuel cells on the line. They will monitor the condition of the various systems and in particular monitor the guidance performance. (Figure 32) Perhaps of some interest is that during this maneuver we see the first of a number of very low acceleration environments, approximately 1 1/3 g during this burn.

Transposition and Docking

Transposition and docking is the first of a number of activities which the crew directly and completely controls. Immediately after the conclusion of the translunar injection burn the crew initiates a roll and pitch maneuver with the total space vehicle to place the high gain antenna for the S-IVB in proper orientation to illuminate the Manned Space Flight Network stations on the earth. This maneuver also assures optimum lighting for the docking maneuver to follow. This activity takes approximately 5 minutes due to the low rates of maneuver used with this high

mass configuration. It takes approximately 10 minutes for the ground stations to confirm the adequacy of the orbit to which the spacecraft has been injected and subsequent to this point in time the crew is free to proceed with the maneuvers separating the CM to a distance of approximately 100 feet. (Figure 33) Again they enter a roll and pitch maneuver to provide antenna orientation for the command and service module to the ground stations and begin the sequence of activities leading up to the docking. (Figure 34)

Figure 35 shows the fashion in which the couches are moved about pivot points and along a rack in order to provide an optimum view for the crew through the rendezvous and docking windows, which can be used by the crewmen on the left and on the right. The view available to the pilot performing the maneuver is illustrated in Figure 36. Figures 37 and 38 show the docking target that the pilot sights on to perform this precision maneuver.

The docking target is observed by the CM pilot through an optical device much like a gun sight. The white cross stands 14 inches above the red target. In its final configuration it will be T-shaped with a diamond in the center of the intersection of the bars. As long as that diamond remains within the white circle the docking is being performed within the capture range of the probe and drogue. The verticle stand-off gives the pilot some cue as to his errors out of the line of approach. The lines of the T are an index of the proper rotational indexing of the CM and the LM.

After having made the initial contact with the probe and drogue the probe mechanism provides for drawing the LM and the CM together and setting four latches at which time a soft docking is achieved. A sequence of activities is begun to pressurize the tunnel area against a leak rate. A pressure hatch and a thermal hatch are removed by the crew. Eight additional latches are hand-set by the crew to achieve a hard dock and structural integrity. (Figure 39) Redundant umbilicals are connected to the LM to provide for the electrical current required in the LM during the translunar leg of the mission and to provide a path for the pyrotechnic device actuations which will release the LM from the adapter. After this operation has been performed the docking mechanisms are restored in the tunnel and the crew activates the pyrotechnic device and withdraws the LM from the adapter by using the SM RCS in a minus-X translation mode. This is done using a series of short intermittent burns to reduce the amount of impingement

of the SM reaction control engines upon the thermal coatings of the LM. The crew establish a separation rate of approximately three feet per second and then orient the vehicle for passive thermal control during the remainder of the mission.

Translunar Coast

Translunar coast is characterized primarily by the mid-course correction requirements determined by the precision of the initial guidance and by the necessity to monitor and maintain periodic checks on the systems. (Figure 40) Periodic maintenance of systems such as the fuel cells and the ECS and a number of the general housekeeping activities are necessary for living during the period while we go to the moon.

This seems a suitable point to discuss the housekeeping problem. Figure 41 shows the various stowage compartments in the three walls of the vehicle - the lower equipment bay, the right hand equipment bay, the left hand equipment bay, and the rear side of the main display console. There are a large number of items stowed in each of these compartments - the numbers being indicated in Figure 42. These items and operations in a small volume in which three men must live for a protracted period make a situation in which everything must have its place and be in that place. The numbers are reasonably impressive all by themselves, while obviously subject to a good deal of discussion in terms of what should be defined as an operation or what one should define as a unit. The numbers indicate that there is a significant problem in simply keeping track of the location and usage of each of the devices. This number of things and operations contributes to the requirement for a good deal of formality and care in various procedures. It also emphasizes how a small error in estimate of time required can adversely affect flight plans.

Figure 43 illustrates the configuration of the vehicle during those periods when the crew would sleep in the CM. Two of the crewmen sleep under the couches in sleeping bag arrangements which provide some measure of thermal control and which further allow the crew to be restrained in the zero g environment. The third man sleeps in his couch where he has direct access to the environmental and electrical control systems which sustain the vehicle. Should any anomaly develop the crew can be awakened in the CM by a direct updata link command. At this point the suits and helmets would be stowed in bags.

The arrangement for the preparation of food is illustrated in Figure 44. Food is stowed in the compartment facing the crewman in man/meal type containers. A water delivery probe located to the crewman's left provides the source for reconstitution of freeze-dried food. The system has the capability for providing three hot meals at the same time to the crew. There is a velcro covered work shelf provided so that equipment can be manipulated in the environment.

Figure 45 illustrates the arrangements of the mechanical components of the ECS which allow the crew to have access to the LiOH cannisters installed in the system. There are two cannisters in parallel. Each has a 24-hour life and they are cycled so that there is a cylinder change each 12 hours. In addition to providing the removal of carbon dioxide these cylinders also contain an amount of carbon to minimize the accumulation of unpleasant odors. That problem, however, is relieved primarily by venting such odors directly overboard.

Typical of the types of status checks which may be conducted during this portion of the mission is an ECS periodic review where the parameters characterizing the nominal performance of the vehicle can be examined. (Figure 46) A number of these are displayed continuously, such as the glycol steam pressure and the glycol discharge pressure. In the event that the crew are in suits it is possible to confirm the oxygen flow and pressure. Some of the displays are time shared. The radiator outlet temperature is an index of the adequacy of the passive thermal control maneuver. Glycol temperature indicates the condition of the electronics cooling. The partial pressure of carbon dioxide is an indication of the safety of the atmosphere. Any event which is critical to the safety of the crew is in the logic of the caution and warning system where an array of annunciations can direct the crew's attention should something occur that is not immediately perceived in general monitoring of the systems.

Navigation activities can also be performed on the translunar and transearth portions of the mission. (Figure 47) The geometry of this activity has been described in an earlier paper.

In the early missions we contemplate checking out the LM prior to lunar orbit entry in order to have the benefit of its systems for certain abort contingencies and to confirm the condition of the vehicle prior to the commitment to

lunar orbit insertion. (Figure 48) This has some additional advantage in that it allows the complete sequence of activity to be observed by the ground. In order to perform this activity the crew must enter the LM and since it has probably depressurized since launch the procedure is to pressurize the tunnel area and the LM from the CM and confirm that the pressure is equivalent across the tunnel and in the vehicles. The crew then removes the pressure hatch, the ablative hatch, the probe, and the drogue, opens the LM hatch, and enters the LM. (Figures 49 and 50) Figure 49 shows the configuration of the CM during the stowage of the pressure and ablative hatches, the drogue and the probe. One crewman's foot can be seen as he is beginning the transfer. It would be expected that two of the crewmen would enter the LM to conduct the checkout.

The orientation of the crew within the vehicle can be appreciated by noting Figure 52. One of the devices noteworthy in this configuration is the sequence camera in the right LM window mounted in a fixed bracket, parallel to the crew's line of sight so that it can observe the lunar landing and record it. The TV camera is stowed at the lower right front. There is stowage of much of the needed equipment in the bags below the right and left side-panels.

Characteristic of the checkout sequence to be conducted is the procedure shown in Figure 53. The regulator status would be verified by confirming the position of the talk-backs associated with each switch and examination of the pressurization of the system by checking the descent pressurization indicator, helium pressure, and the other system status points shown. This would be the series of activities for the descent propulsion system up to the point where the crew would begin that sequence associated with the throttle manipulation.

Lunar Orbit Insertion

Lunar orbit insertion is again a guidance maneuver which has been discussed in significant detail in a preceeding paper. The activities are very similar on the part of the crew for this maneuver as for the translunar injection and for earth orbit insertion. (Figure 54) There is an aspect of the geometry which has not been commented upon. Figure 55 shows the growth in the apparent size of the moon as a function of mission time. The schematic in Figure 56 shows the effect of the lighting conditions. At the 75:36:36 point in time the moon occupies 23 degrees, an hour later it occupies 63

degrees, and 15 minutes later it has grown to approximately 130 degrees with a 134 degree apparent size in lunar orbit. The significant thing is that we will be approaching the dark of the moon and the earth shine is from a half earth. The crew can, however, perceive that this is not a collision course by observing the apparent regression of the limb of the moon against the line of sight.

After entry into lunar orbit, the two crewmen would again transfer into the LM and transfer a significant amount of equipment. (Figures 57 and 58) The PLSS has been carried in the CM because it provides a capability for emergency extravehicular activity. The emergency oxygen supply (EOS) provides a five minute capability in a high pressure bottle, and is used to provide an emergency backup to the PLSS. These have been stowed in the CM in order to have them in a more favorable thermal environment. The extravehicular gloves and the TMG have been kept in the CM to provide for an extravehicular capability if required. There is but a single radiation survey meter and it would now go to the LM where the inherent shielding by the vehicle is less effective. There is only one TV camera. The EVCT is the extravehicular crew transfer device. The umbilicals would remain in the CM.

Lunar Descent

The series of maneuvers associated with the LM descent to the lunar surface has been discussed in considerable detail in preceeding papers. Figure 59 is a very brief review of the activities assigned to the crew in conjunction with the primary guidance system and the abort guidance system. In addition to these there are a number of pilotage activities which can be performed by the crew as opportunity allows. Through the overhead window, which is above the left crewman, it is possible to have a view of the lunar horizon during the descent phase of the mission, which may provide some useful indications of attitude and altitude. The primary function of the crew during the braking phase is to monitor the automatic systems and to provide suitable initiation of various equipments. They would confirm the status of the RCS and the ascent propulsion system prior to final approach in order to confirm the existence of their abort capability. It is interesting to note that during this portion of the mission the crew will experience approximately one third of a g with the vector along their body axis as in standing. (Figure 60)

Figure 61 shows the LLRV which has now flown approximately 140 flights. To simulate a LM descent trajectory it takes off in a VTOL mode using the jet engine as its thrust source with attitude controlled by a hydrogen peroxide RCS. The pilot climbs to approximately 600 feet and establishes the initial conditions for the entry to the simulated portion of the LM descent trajectory. He establishes these conditions at approximately 400 feet altitude, having 45 feet per second forward velocity and approximately 9 feet per second descent velocity at that point in time. There is a departure from the fidelity of the simulation in that the line of sight of the seated crewman in this vehicle cannot be depressed so far as in the LM and he is seated rather than standing. The seating requirement derives from the use of an ejection seat for safety. The pilot has cross-range and down-range indicators, the flight director, and thrust-to-weight indicators. There is a three axis hand controller for attitude control. The geometry of viewing to the instruments and to the available window is correct. When the pilot has established his initial conditions, he transfers to a lunar simulation mode in which the engine is gimballed and provides 5/6 of a g thrust. The simulated descent engine, also a hydrogen peroxide engine, is ignited and from this point on the attitudes are characteristic of the LM descent and its piloting characteristics can be evaluated. The control authority is that of the LM. The attitude of the vehicle is approximately 12 to 15 degrees pitch up to reduce the horizontal velocities. He enters a hover at approximately 75 feet and begins a controlled rate of descent to the touchdown point, reducing the rate to approximately $3\frac{1}{2}$ feet until he is very near the surface where a major departure in the simulation occurs. This vehicle is flown in a thrust to touchdown mode to minimize landing loads. The weight limitations do not allow adequate attenuation for the routine practice of thrust-off landings.

Lunar Stay

The graph in Figure 62 is in the format used earlier to show the activities characteristic of the lunar stay time. Immediately after landing the vehicle is checked to assure that conditions are suitable for remaining on the surface. The descent tanks are vented in order to preclude problems due to thermal buildups and the crew immediately aligns the IMU and places it in a standby mode to have an any time departure capability. Postlanding checkout is

scheduled for 33 minutes and this time estimate is based upon rehearsals conducted in a LM mockup.

This particular plan shows the immediate donning of the extravehicular equipments and an initial exploration activity. Representative tasks are inspection of the vehicle for any apparent exterior damage or leaking, confirmation of the conditions of the landing such as depressions in the lunar surface, slide marks, and measurement of gear stroke distance. (Figure 63) Other early extravehicular activity would be the deployment of the necessary equipments, such as the antennas, and an initial survey from the LM platform to do TV and film scans of the area for later analysis.

The next series of Figures shows the configuration of the vehicle for various activities. In Figure 64 the crewman is donning the PLSS. It shows one crewman in the TMG with the PLSS and the EOS attached. The second PLSS is shown in the recharge station. The EVCT is shown above the wall mounted PLSS. Food and other equipments are stowed immediately below the EVCT.

As noted, one of the first extravehicular activities would be a camera survey of the area and a postlanding inspection. Figure 65 shows the net and cable device used to transfer equipment to the surface from the ascent stage.

Figure 66 illustrates the sleeping arrangements within the vehicle. One of the crewmen suspends a hammock arrangement from the front edge of the ascent engine dome back to the rear wall and the other crewman sleeps across the floor of the vehicle.

Since the vehicle is pressurized during descent and touchdown and because it enhances the speed and efficiency of operations the crew could be expected to remove their helmets and gloves while donning the extravehicular gear. These activities can be performed by a single individual but they go much more rapidly and with considerably more confidence when two men can perform an operation and check each other. Again, the complexity of the operation is one which makes it a formal checklist operation. The PLSS transfered from the CM and stowed temporarily on the floor of the vehicle is mounted in the harness called the donning station where it is suitably mounted adjacent to the checkout controls and where it can be held in place while the crewman donns the rest of the equipment which he requires. Figure 67 shows the TMG trousers.

The carrying straps for the PLSS are threaded through the TMG jacket to minimize the difficulties of thermal shorts in the garment. The jacket is difficult to donn but this is a tradeoff between ease of donning and the later cumbersomeness of excessive material.

Two men with the PLSS's and various other elements of the suit occupy a pretty substantial volume and when pressurized they move in a somewhat awkward fashion. It takes a great deal of time to perform a number of these operations because they are performed very very carefully.

The extravehicular visor, which provides various degrees of transmissibility, is donned prior to egress to protect the crewman from the extreme range of lighting values to be encountered.

Stationed on the initial portion of the ladder is an A-frame type step ladder. This is provided to enhance the ease of access to various portions of the descent stage, to contend with various orientations of the vehicle, and to make easier the crew's access to the fixed ladder on the front leg. Figure 68 shows that the first man out would free the device and guide its descent to the surface while the man remaining in the cockpit lowered it via the equipment lift line. Figures 69 and 70 show the completion of the descent cycle for the first man. The height of the fixed ladder above the surface for a "soft" landing, as illustrated, shows why the ladder is required.

It is apparent from the nature of these activities that substantial amounts of time are required to perform tasks that one expects to be done rather expeditiously. This accounts to some degree for the fashion in which we schedule crew time and for the fact that it is not very difficult during the course of the mission to encounter conditions wherein things take substantially more time than we have contemplated. The mere act of fastening a snap which can be very straightforward in one g and street clothes can become a very demanding operation in a pressurized suit under weightless conditions. The problem is less severe but not removed at 1/6 g.

All TMG's, PLSS's, and associated equipment must be returned to stowage locations so that the arrangements for sleep can be made. The harness which was used to hold the PLSS in a donning configuration is used to provide the hammock configuration for sleeping. The crew are to sleep suited in

the LM. (Figure 71) The oxygen supply system has some capability in the event of a puncture but the thin skin and location on the lunar surface have a higher, though very low, probability of pressure vessel failure than we find in the CM. If there is a puncture the system can hold pressure in the vehicle for a substantial time period.

When the crew is ready to depart there is a substantial amount of equipment left behind (Figure 72) - the used LiOH cannisters from the LM itself and from the PLSS's, the batteries, bags containing the condensate collected in the PLSS, the urine, and other fecal materials (treated with germicidal agents) are all stowed in a compartment in the descent stage. These equipments are left behind in the interest of saving weight and volume in the ascent stage enhancing the amount of material which can be returned for scientific purposes.

Lunar Ascent

Lunar ascent has been described in considerable detail in a previous paper. Figure 73 shows the configuration of the vehicle at this time. The PLSS is stowed on the floor to ease access to the optical telescope which would be used to align the IMU just prior to liftoff. The other PLSS previously stowed in the recharge station has been discarded. The equipment is stowed in such a fashion as to maintain the symetry of loads as well as possible.

The crew activities during this maneuver are essentially the same as those which we have reviewed earlier for other powered flight maneuvers. (Figure 74) During the ascent portion of the mission the crew will again experience a 1/3 g acceleration environment which is quite acceptable on the standing configuration. The details of the rendezvous and docking sequence as executed by the guidance system have been discussed in detail earlier. It is perhaps well at this point to note that in addition to those activities it is possible to use the pilotage routines based upon observation of the FDAI and of the other instruments as they have been used in Gemini.

Rendezvous and Docking

In this phase of the mission the LM is the active element of the docking sequence. (Figure 75) The pilot of the LM can observe the command and service module through an overhead

window (Figure 76) and he has a docking aid similar to the one shown earlier installed in the window of the CM. (Figure 77)

When the two vehicles have been docked there is a repetition of the sequence in which the docking mechanisms must be removed from the tunnel and then the crew can transfer the equipment and themselves back to the CM. (Figure 78) At this point in time a number of devices are left in the LM to relieve conjestion in the CM and to enhance the arrangements for stowage of significant items returned from the lunar surface. It is perhaps noteworthy that one of the PLSS's is retained in the CM, its oxygen supply being used as a backup to the entry oxygen supply during that portion of the mission. The TV camera is returned to the CM and the LM flight plan which is now the log and record of that flight is returned for record purposes. The sample return containers and the films from the sequence camera and other cameras are returned. The data storage electronic assembly is the voice recorder provided for crew usage in the LM. The devices transferred to the LM are those no longer required in future phases of the mission.

It is possible that during the docking sequence there can be a failure of the mechanism which would not allow transfer through the tunnel. There is a device called the EVCT (Figure 79) which is a metalic tape boom reeled out to some 25 feet in length which engages a device called the bailer bar at the command and service module interface. (Figure 80) Either the crewman in the CM can open the hatch or the hatch can be opened after the vehicle is depressurized by the crewman on the outside of the CM. The procedures would call for transfer of one of the crewmen from the LM using the PLSS and this device. He would then use this device to return the PLSS to the other crewman or, having established the configuration of the CM, the second crewman could be brought across using this device as a tether and using the EOS to provide breathing gases.

Transearth Injection

Transearth injection for the crew operations is very similar to the other flight maneuvers previously described, and the trajectory and guidance considerations are reviewed in other papers. It has a somewhat higher acceleration than the maneuvers discussed earlier, the light weight of the system now providing an effective 1/2 of a g acceleration.

Transearth Coast

Transearth coast is characterized by those activities which were discussed for the translunar leg of the mission. (Figure 81) The phenomena of apparent growth in size of the earth will occur on this leg.

Entry

The entry sequence has been described in some detail in a previous paper but it is perhaps important to note that a large number of sequential events are controlled directly by the crew and those not directly under their control in the nominal mode are subject to crew backup. (Figures 82 and 83) Some of the particular events are those noted in Figure 29 in items 5 and subsequent, excepting, of course, those items associated with LES tower operation.

SUMMARY

The critical role of the crew in providing flexibility and reliability during the mission has been emphasized. The relation of training equipments and procedures to such requirements has been noted. The lunar landing mission illustrates the wide range of capabilities of a manned spacecraft.

Questions and Answers

CREW TASKS AND TRAINING

Speaker: Joseph P. Loftus

1. Mr. Nix - Has consideration been given to opening the top hatch and looking around?

   ANSWER - Yes, and this approach has been rejected because it requires removing the drogue from the LM top hatch which is awkward and potentially hazardous. Also, the increased length of oxygen umbilical imposes pressure drop penalties upon the ECS and interference with general crew mobility. The view from the front of the LM is greater than 200 degrees and is considered adequate.

2. Mr. Davidson - How much of the time can the crew see the earth during translunar and transearth phases of the missiion mission?

   ANSWER - A substantial portion of the time. The field of view of the windows and optics is large and will make some view available most of the time.

3. Mr. Davidson - What is the effect of crew movement on spacecraft attitude?

   ANSWER - Effects are expected to be minor.

4. Mr. Beattie - Is the copilot task during LM landing essentially a monitoring one?

   ANSWER - No, essentially a team operation is planned.

5. Dr. Reiffel - Is there time allocated for inflight experiments?

   ANSWER - There will possibly be some time available since not all of the crew time has been allocated to specific spacecraft or mission operations.

6. Dr. Reiffel - Do you plan to sterilize the food containers and fecal cannisters?

   ANSWER - Yes. Germicidal provisions are included in each of the containers.