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MSC INTERNAL NOTE NO. 67-EG-23

APOLLO PROGRAM

APOLLO RENDEZVOUS WITH COMMAND MODULE ACTIVE

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N70-35751
(ACCESSION NUMBER)
58
(PAGES)
TMX 64487
(NASA CR OR TMX OR AD NUMBER)
(THRU)
1
(CODE)
31
(CATEGORY)
FACILITY FORM 602

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

Houston, Texas

September 28, 1967

RECEIVED
NASA STI FACILITY
INPUT BRANCH
AUG 1970

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Springfield, Va 22151

SUMMARY

The primary mode of rendezvous in the Apollo mission is one in which the Lunar Module is the active vehicle as it maneuvers to intercept the Command and Service Module. In the event a rescue of the Lunar Module is required, the Command Module must become the active vehicle. Since the CSM does not have a rendezvous radar as does the LM, but utilizes its 28-power sextant as a means of updating its onboard computed state vectors, and also because the CSM translation acceleration is low, it was not known what would be an optimum concentric orbit for the CSM rescue of the LM. Thus, a piloted simulation study was conducted to determine the correlation between differential altitude (ΔH) of the concentric orbits prior to Transfer Phase Initiation (TPI) and (1) the CSM/RCS fuel requirement, (2) the relative state vector information uncertainties, and (3) the ease of system monitoring and control during the rescue. The study was constrained to evaluate problems relating to a transfer phase central angle of 140°. Consideration is given to crew task loading where one crew member will be aboard and controlling the Command Module to rescue the Lunar Module or where a full three man crew will rendezvous with an unmanned S-IVB target.

The study concludes the following: (1) The pre-TPI ΔH should be constrained to approximately 10-15 n mi. The upper bound is limited by the translational acceleration capability of the CSM to cope with high intercept velocities and also by the SM-RCS fuel available for rendezvous. The lower bound is constrained by the uncertainty in the knowledge of the vehicle state vectors when applied to midcourse maneuvers and the resulting ΔV penalty. (2) The rendezvous phasing should be constrained to allow at least the last 1 n mi of closure to occur in daylight. This constraint can be removed by the addition of an independent ranging device on the CSM.

It is concluded that if the two above mentioned constraints are met, and if the PNGCS and MSFN are operating as primary-mode-operation is defined herein, the CSM possess a satisfactory rendezvous capability using the Concentric Flight Plan.

A backup rendezvous capability also exists which requires appreciably more RCS fuel than the primary mode and can result in non-standard final approach conditions.

INTRODUCTION

During the Gemini flight program a technique of rendezvous was developed termed the "concentric flight plan" which afforded good usage of the spacecraft control system, guidance and navigation system, and pilot. This technique has been brought forth from its evolution in Gemini and introduced into the Apollo lunar mission. Normally, during the lunar mission the Command-Service Module (CSM) will remain in a near-circular orbit around the moon while the Lunar Module (LM) descends to the surface. The LM then launches, and returns to the CSM. However, if at some point after the LM ascent and insertion the LM should become immobilized, the CSM must then become the active rendezvous vehicle and effect a LM rescue. Since the CSM does not have a rendezvous radar as does the LM, but utilizes its 28-power sextant as a means of updating its onboard computed state vectors, and also because the CSM translation acceleration is low, it was not known what would be an optimum concentric orbit for the CSM rescue of the LM. Thus, a piloted simulation study was conducted to determine the correlation between differential altitude (ΔH) of the concentric orbits prior to Transfer Phase Initiation (TPI) and (1) the CSM/RCS fuel requirement, (2) the relative state vector information uncertainties, (3) the ease of system monitoring and control during the rescue. The study was constrained to evaluate problems relating to a transfer phase central angle of 140°. It was also desirable that backup rendezvous navigation and guidance procedures specifically tailored for the CSM evolve from the study. This paper presents and discusses the results of that study.

SCOPE

Each simulation run began at a point $9\frac{1}{2}$ minutes prior to the transfer phase initiation maneuver (TPI) and terminated at a relative range of 1000 feet with the range-rate below 1 fps and the LOS rate below 0.1 mr/sec (intercept course). Provisions were made for automatic TPI and midcourse maneuvers with manual backup control available. The terminal phase was completely manual. The average elapsed time required for one run was on the order of 45 minutes.

SYMBOLS

(A, E)	- target LOS angles with respect to the CSM body axes, deg
$(\dot{A}, \dot{E})$	- target LOS angular rate, mr/sec
B	- Inertial to CSM coordinate transformation matrix
B^{-1}	- Inverse transformation matrix
(F_x, F_y, F_z)	- Inertial acceleration force components due to thrust, lbs
g	- earth gravity, ft/sec^2
(I_{xx}, I_{xz}, I_{yz})	- moments of inertia about the CSM body axes, slug-ft^2
(I_{xy}, I_{xz}, I_{yz})	- products of inertia about the CSM body axes, slug-ft^2
Isp (RCS)	- specific impulse of SM/RCS jets, seconds
Isp (SCS)	- specific impulse of SM/SPS, seconds
(J_{1-16})	- SM/RCS jet nomenclature
Mo	- initial CSM mass, slugs
Mt	- CSM mass at time t, slugs
(p, q, r)	- CSM angular rates about its body axes, deg/sec
r_f	- inertial position vector of CSM, feet
(r_{fx}, r_{fy}, r_{fz})	- components of CSM inertial position vector, feet
r_s	- inertial position vector of LM, feet
R	- relative range of CSM from LM, feet
$\dot{R}$	- rate of change in range, ft/sec
(R1, R2, R3)	- DSKY display registers

S	- LaPlacian operator
t	- simulation run elapsed time, seconds
Tsps	- SM/SPS thrust, lbs
TG	- "time to go" to end of TPI burn, seconds
TTI_o	- initial "time to ignition" for TPI burn, seconds
TTI	- current "time to ignition", seconds
(T_x, T_y, T_z)	- body acceleration force components due to thrust, lbs
VG_o	- initial "velocity to be gained" for TPI burn, fps
$(VG_{ox_I}, VG_{oy_I}, VG_{oz_I})$	- inertial components of initial "velocity to be gained", fps
VG_t	- current "velocity to be gained" for TPI burn, fps
$(\Delta V_{TX_b}, \Delta V_{TY_b}, \Delta V_{TZ_b})$	- inertial TPI trim ΔV's transformed to CSM body axes, fps
$(\Delta V_{\delta x}, \Delta V_{\delta y}, \Delta V_{\delta z})$	- accumulated change in velocity due to SM/RCS translation commands along CSM body axes, fps
$\Delta V_{\delta tot}$	- summation of accumulated velocity changes due to SM/RCS translation commands, fps
V_I	- relative inertial velocity vector of CSM to LM, fps
(Wp, Wq, Wr)	- accumulated propellant usage due to rotation commands about the CSM body axes, lbs
$W_{a_{tot}}$	- summation of accumulated propellant usages due to rotation commands, lbs.
(X_B, Y_B, Z_B)	- components of CSM relative inertial position vector transformed to CSM body axes, feet
$(\dot{X}_B, \dot{Y}_B, \dot{Z}_B)$	- components of CSM relative inertial velocity vector transformed to CSM body axes, fps
(X_I, Y_I, Z_I)	- components of CSM relative inertial position vector, feet
$(\dot{X}_I, \dot{Y}_I, \dot{Z}_I)$	- components of CSM relative inertial velocity vector, fps
$(\ddot{X}_I, \ddot{Y}_I, \ddot{Z}_I)$	- components of CSM relative inertial acceleration vector, ft/sec^2
$(\bar{X}cg, \bar{Y}cg, \bar{Z}cg)$	- center of gravity location relative to CSM body axes, inches
$(\delta x, \delta y, \delta z)$	- translation commands along CSM body axes
(θ, ψ, ϕ)	- Euler angles, degrees
λ	- central angle rotation of the LM about the earth, degrees
μ_e	- earth gravitational parameter, ft^3/sec^2
$(\sigma y, \sigma z)$	- SPS thrust misalignment, radians
τ_1	- time constant of SM/RCS jets, sec
ω	- angular rate of LM radius vector, rad/sec

DESCRIPTION OF SIMULATION
GENERAL

The motion of the CSM was simulated in six degrees-of-freedom and that of the LM in three degrees-of-freedom using general purpose computers. The long period dynamical equations (orbital mechanics) were solved on a digital differential analyzer (DDA) and the short period dynamics (rotational equations of motion) were mechanized on analog computers. The CM-SCS was mechanized in the Block II configuration on an analog computer. A simulator cockpit was coupled with the general purpose computers for pilot monitoring and control of the rendezvous trajectory. A virtual image visual display system, driven by the DDA, displayed a model of the LM/SIVB to the pilot in simulated three dimensional space. Figure 1 gives a general block diagram of the simulation mechanization.

EQUATIONS OF MOTION

The motion of the CSM relative to the LM was expressed in three translational and three rotational degrees of freedom. The equations of motion describing translation of the CSM relative to the LM are referenced to a displaced inertial coordinate system shown in figure 2. The coordinate system is termed displaced since the origin is always centered in the orbiting LM; however, the axes remain inertially fixed in direction.

A math model of the translation equations of motion is given in figure 3. Two identical sets of equations are programmed to provide for a simulation of the actual trajectory being flown as well as the onboard computed trajectory. Since all guidance parameter displays come from the onboard trajectory computations, it was important to simulate the onboard state vector errors which can conceivably exist during each phase of the mission.

The rotation equations of motion, as mechanized in the attitude control system for the pitch, yaw and roll axes, are given in figures 4a, 4b, and 4c respectively. The Euler angle sequence used to reference the CSM body axes to the inertial frame was (θ, ψ, ϕ) and is defined by the transformation matrix given in Appendix A.

SIMULATED CSM

General

Since the purpose of the simulation was to study a CSM rescue of the LM from a point just prior to TPI, it was necessary to simulate the CSM systems which will be used for this phase of the mission. At present the Command Module Computer (CMC) is programmed to compute the TPI and midcourse maneuvers and then automatically fire the translation propulsion and perform attitude steering for these maneuvers. Terminal phase control, which includes range rate braking, is presently a manual task performed by the pilot. In the simulation, however, steering and translation thrusting for the midcourse maneuvers were performed by the pilot after obtaining the required maneuver from the simulated CMC. Although primary attitude control in the CSM uses the digital auto pilot (DAP) it was decided to simulate only the stabilization control system (SCS), which is an analog control system, since more information was available on it at the time.

Command Module Computer

The essentials of several CMC programs were simulated. These programs were (1) TPI PRETHRUST PROGRAM, (2) SPS THRUST PROGRAM, (3) TRANSFER PHASE MIDCOURSE (TPM) PROGRAM, and (4) TRANSFER PHASE FINAL (TPF) PROGRAM. The simulation of each program was as follows:

(1) TPI PRETHRUST - The nominal TPI ΔV and attitude were precomputed offline. The correct TPI attitude was preset into the simulated CMC and the SCS automatically controlled the CSM to this attitude when the pilot placed the SPACECRAFT control switch in CMC. The nominal TPI ΔV was set into the ΔV meter by the pilot.

(2) SPS THRUST - This program computed the time to ignition (TTI) of the SPS and displayed it on the first register (R1) of the display and keyboard (DSKY). The velocity to be gained (VG) was displayed on R2 and the velocity measured (VM) was displayed on R3. During an automatic SPS thrust maneuver the ΔV meter and VG counted toward zero while VM counted from zero up to the final ΔV applied. SPS thrust was terminated when the ΔV meter counted to zero. A 20.4 second RCS 2-jet ullage maneuver was automatically made just prior to any SPS burn (ullage) started at TTI = -19.2 sec). After the SPS burn, RCS trim ΔV's were computed and displayed in body axes on R1 (ΔVx), R2 (ΔVy), and R3 (ΔVz). The trim ΔV's were applied manually.

(3) TRANSFER PHASE MIDCOURSE - This program utilized Keppler's and Lambert's routines to compute midcourse corrections required for maintaining an intercept trajectory. The program could be called by the pilot at any time between TPI and intercept. The midcourse maneuver was computed and displayed in body axes on R1 (ΔVx), R2 (ΔVy), and R3 (ΔVz). The maneuver was executed by manually thrusting along the three body axes to zero R1, R2, and R3.

(4) TRANSFER PHASE FINAL - The purpose of this program is to display pertinent guidance parameters to the pilot during the terminal phase; thus, range to the LM (R), rate of change in range ($\dot{R}$), and the angle (θ) between the CSM X-body axis and the CSM local horizontal were computed and displayed on R1, R2, and R3, respectively.

Stabilization and Control System

A simplified stabilization and control system representative of the Block II configured electronics was simulated which provided translation and attitude control using Service Module reaction control system jets. However, the attitude fuel associated with X-axis translation was not exact due to a simplification in jet logic. A correction in pitch and yaw of 7% and 4% respectively of X-axis RCS translation fuel should be used to modify the attitude fuels shown in Table III. For PNGCS mode control the correction should be subtracted; for SCS it should be added. The attitude control modes simulated were acceleration command, rate command (with attitude hold), and minimum impulse. Math models of the simulated SCS for the pitch, yaw, and roll channels are given in figures 4a through c. The system was rate limited to 0.65°/sec in all three axes with the RATE switch in LOW position. In HIGH position the rate limits were 7.0°/sec in pitch and yaw and 20°/sec in roll. The attitude deadbands were 0.5 degrees and 5.0 degrees with the ATTITUDE DEADBAND switch in MIN or MAX positions. Translation control was by a simple acceleration command system. Pertinent data relative to CSM attitude and translation control are given in Appendix B.

Service Propulsion System

The SPS was simulated for use where large thrust maneuvers were required and was programmed to fire automatically if desired. The primary application was that of making the transfer phase initiation (TPI) maneuver. Pertinent characteristics of this engine are also given in Appendix B.

Displays and Controls

The displays and controls simulated were those necessary for the control of rendezvous from TPI to intercept. These are indicated by heavy lines in figure 5a. A photograph of the displays and controls mockup is shown in figure 5b. The panel configuration is Block II. Since a Block II computer was not available to drive an actual DSKY, the DSKY was simulated using digital voltmeters and switches to provide the displays which are normally available during rendezvous. One deviation was made from the Block II configuration. The FDAI source switch was used to change the 8-ball display from inertial to local vertical attitude. The attitude and translation controllers are shown in figure 5b in the lower right and left hand corners respectively.

CSM Geometry

The Service Module Reaction Control System (SM/RCS) jet geometry are given in figures 6a and b. Figure 6c gives the location of the center of gravity (c.g.) for the CSM in this mission phase. The importance of the CSM jet geometry and c.g. location is that translation thrusting also creates rotational torques about that spacecraft body axes. These disturbance torques, in turn, must be counteracted by firing additional jets using the attitude control system. Thus, as the number or duration of translation commands is increased, the attitude fuel requirement is also increased.

SIMULATED TARGET

The target for rendezvous was a simulated Lunar Module docked to an SIVB booster. A photograph of the simulated LM/SIVB is given on figure 7. A light which flashed at a rate of once per second was mounted on the LM to simulate the Agena acquisition light which will actually be used in the mission. The simulated target was attitude stabilized about all three axes. To the pilot, the target appeared as a point source of light until a range of 6000 feet was reached. At this time the target was moved closer to the TV camera at a rate proportional to the range rate. The pilot viewed the target through a virtual image projector which gave an effect of three dimensional space.

SIMULATED RENDEZVOUS TRAJECTORIES

Nominal Trajectories

In order to evaluate the effect of varying the target/chase vehicle differential altitude, trajectories of 5, 10, 15, and 20 nautical mile ΔH were simulated.

The nominal condition for all ΔH cases was an onboard computer assumption that the two vehicles were coelleptic; i.e., at a constant differential altitude which had been maintained since the previous ground-directed CDH (Constant Delta Height) maneuver. The actual trajectory encountered was a function of the simulated initial condition errors at 9 minutes 30 seconds prior to the computed (nominal) transfer maneuver (Terminal Phase Initiation). These errors are discussed in a later section.

The terminal phase orbital travel (λ) was constrained to 140° for this study. This angle is the one presently planned for use during Apollo rendezvous mission because of its good intercept approach angle (approximately 45° with respect to the local horizontal), and satisfactory excess orbital energy which results in a nominal chase vehicle apogee of .5 nautical mile above the target (fig 8). This excess energy aids in overcoming small-error miss cases. Furthermore, the 140° transfer allows adequate time (35 minutes earth orbit and 46 minutes lunar orbit) to perform onboard state vector improvement for a possible midcourse maneuver prior to the braking phase. Figure 9a displays the various nominal ΔH cases for the 140° transfer in a target local horizontal reference frame. Figure 9b magnifies the terminal portion of these trajectories in which midcourse and braking maneuvers would be accomplished. Note the linear relationship of range and ΔH for constant time from TPI. Figure 9b also depicts the initial lighting coverage provided by an Agena-type acquisition light installed on the LM/SIVB target vehicle. This orientation does not provide full visual coverage for nominal trajectories; however, further discussion will be devoted to this problem.

Nominal range rate versus range time histories for the four ΔH's investigated are given in figure 9c. It can be seen that, as ΔH increases, the range and range rate at any elapsed time from TPI is proportionally higher. Figures 9d and 9e display the line-of-sight information which is characteristic of all 140° transfers. The fact that the LOS time history is the same for any ΔH, as long as the transfer is maintained at 140° affords a satisfactory method of monitoring the terminal phase of rendezvous. These monitoring methods will be discussed in depth in a later section.

Trajectory Errors

As indicated above, various state vector errors were simulated corresponding to the navigation systems which will exist in such a mission. The navigation schedules to be used also have a bearing on the magnitude and direction of the errors which exist at the point where the simulator runs began. It should be pointed out that the primary navigation sensor of the CSM is a 28 power sextant (CSM-SXT) and that the simulation did not possess this primary navigation capability. Thus, the initial condition errors for this simulation were obtained from a separate digital simulation of the Manned Space Flight Network (MSFN) and the CSM-SXT navigation. The various state vector errors used, and their corresponding navigation schedules, are given in Table I. Figure 10 gives a characteristic plot of the local vertical trajectory for each error condition given in Table I where no control of the trajectory was performed except for the nominal TPI maneuver. The miss distance of a trajectory gives a good indication of the relative effort required to restore the vehicle to an intercept trajectory.

RESULTS AND DISCUSSION

Rendezvous Control Procedures

Background - For Apollo missions, three primary and one auxiliary crew station are provided within the CSM. Figure 11a depicts these stations. For all high-accéleration flight phases (boost, entry, and service propulsion system maneuvers) the flight crew occupies the three primary stations. Primary vehicle control during these phases is accomplished at the command pilot station (Sta. 1) with the CSM pilot (Sta. 2) assisting. However, during coasting flight the auxiliary crew station in the Lower Equipment Bay can be used for optical navigation and inertial platform alinement. When this station is being utilized the CSM pilot couch can be folded down to clear the area. Thus, normal operation can be accomplished conveniently for a three-man crew.

During the Lunar Module operations the command pilot and LM pilot (Sta. 3) will depart the CSM. The CSM pilot will then continue to monitor the mission phase and navigate from the Lower Equipment Bay Station. However, if the LM becomes disabled to the degree that it cannot perform the rendezvous maneuver, the CSM pilot must perform both the navigation and control functions to effect a successful rescue.

As the mission plans were developed, it became necessary to define the procedural control requirements for the CM rendezvous operations. These procedures were to satisfy earth orbital demonstration as well as lunar orbital rescue. The two cases differ somewhat since all crew members will be aboard the CSM in the initial Apollo earth orbital rendezvous demonstration configuration. All will be available for performing the individual tasks required for rendezvous, that is, for sextant tracking to improve state vector knowledge and also for control of the vehicle. However, in a lunar orbit rescue of the LM, only one CM crew member will be available to perform all the tasks necessary to rendezvous; therefore, it is obvious that the CSM rendezvous control procedures must be developed for one man operation.

The generalized procedure for the crew during the Gemini missions and that which is planned for the Lunar Module requires that their main attention during the rendezvous phase be devoted primarily to the visual tracking of the target in which the longitudinal body-axis of the vehicle is boresighted upon the target vehicle. This target can be seen in darkness by attached flashing lights or in the daytime by reflected sunlight. Radar acquisition can be either through an automatic or manual mode. Following acquisition, the guidance computer is updated automatically by radar range, and line-of-sight data. Prior to terminal phase initiation (TPI), the crew will receive maneuver information available both from the ground (MSFN), from the flight computers of the primary guidance system, or from backup flight charts. This permits the crew to guide the vehicle into a trajectory which is co-elliptic (nominally concentric) with the target vehicle for a period of time prior to the Terminal Phase Initiation.

Following the Terminal Phase Initiation maneuver, the ground has degraded capability to aid the crew in determining the magnitude of maneuvers which should be used for mid-course correction prior to intercepting the target vehicle. It is therefore necessary during this period, that the crew be afforded a maximum autonomous capability to determine the maneuvers necessary to assure a rendezvous intercept. During the Gemini Program, guidance monitoring flight charts evolved which gave the crew the ability to monitor the progress of the rendezvous phase with some facility. The technique employed made use of a relative position plot (figure 11b), which was based on a target centered local vertical coordinate system. By boresighting the target vehicle using a collimated reticle alined to the spacecraft longitudinal body axis, the crew could then observe the pitch angle to the target with reference to the local horizontal and simultaneously record the range to the target from radar data. By recording this data on the graphical display provided them, a fairly accurate knowledge of their position relative to the target could be maintained. This position was compared to a nominal trajectory also placed on the chart which gave a qualitative indication of the maneuvers which might be required as the mission progressed. Also provided the crew were charts based upon nominal linearized maneuver solutions (fig. 11c). These charts allowed the crew the capability of utilizing range,

range rate, and line-of-sight data to determine what off-nominal increment would be required at the next maneuver point. In this manner satisfactory guidance monitoring of the primary solutions afforded by the ground and the onboard computers was obtained. The charts previously mentioned are ones which account for the coplanar components of velocity change required for intercept. Other charts were also developed to evaluate the out-of-plane requirements and to determine the time at which to make the appropriate corrections of these out-of-plane errors. Generally, since the out-of-plane components are sinusoidal in nature, the chart itself is somewhat more simplified.

These charts are used until about two-thirds of the terminal phase trajectory has been completed. The remaining portion of the trajectory is monitored by observing the inertial line-of-sight rates of the target, using either a starfield background or an attitude-stabilized vehicle, range to the target, and range rate. At this time the pilot attempts to control the vehicle's line-of-sight rates to as near zero as possible. As predetermined range gates are reached, the range rate is reduced to a specified value in order that a well-controlled, safe approach may be made.

Although this rendezvous technique nominally affords an approach that is always from below and in front of the target, state vector errors existing at TPI, which are not compensated for by the TPI burn or midcourse corrections, can cause the approach path to the target to be badly off-nominal. These conditions can adversely affect the lighting conditions and line-of-sight/range rate control. Thus, great care is afforded to keep the chase vehicle on a standard approach to conserve fuel in the terminal phase.

<u>CSM Rendezvous Navigation</u> - The CM guidance system (PNGCS) is essentially the same as the LM guidance system with respect to inertial components. However, for rendezvous navigation, the CM uses a 28 power sextant, whereas the LM uses a radar/transponder system. Recursive navigation techniques have evolved which give an accurate estimation of relative state vectors of the LM (LM relative to CM) through optical sightings using the sextant; however, since this sighting task is a manual procedure, it requires one of the CM crew members to be stationed at the Lower Equipment Bay during the updating period. This requirement involves crew task loading and spacecraft geometry considerations, inasmuch as the pilot in the left seat cannot boresight the vehicle simultaneously while the pilot at the navigation base is taking optical sightings through the sextant.

A possible procedural modification to the navigation system during the state vector updating process would be to use the collimated docking reticle in lieu of the sextant and make sighting marks using the computer enter-button when the CSM X-axis is alined (boresighted) to the target satisfactorily. The navigation program knowing the relationship between the reticle line-of-sight and the CSM body-axis would then instantaneously measure the IMU gimbal angles and process line-of-sight data in order to achieve the desired state vector updates. Operationally, this would alleviate the command pilot requirement of changing the CSM body attitude in order that another pilot or crew member could use the sextant during the terminal rendezvous phase. It would also mean that a single pilot could control the vehicle from the left seat and make navigation sightings simultaneously. From the analytical viewpoint, the degraded resolution and accuracy of the collimated reticle compared to that of the sextant would infer that the accuracy of the knowledge of the relative state vector; i.e., range and range-rate, would be less precisely known. However, the possibility of taking more sightings as the rendezvous progressed would insure that any velocity errors could be maintained at a minimum during the terminal phase.

Simulation Primary Mode Procedures - The primary guidance and navigation procedures used in this simulation were slightly different than that which would be used in an actual mission. The TPI maneuver consisted of automatic ullage and SPS burns for 10 and 15 nautical mile ΔH cases using the nominal TPI ΔV which was pre-computed. On a 5 nautical mile ΔH, the TPI maneuver was manually thrusted with the RCS jets. In the midcourse phase, however, sextant sightings as such were not made because the navigation bay, i.e., CSM sextant and telescope was not built into the simulation cockpit. Therefore, on runs where sextant sightings would have been made, the CSM relative state vectors were updated "off-line" of the simulation at 12 minutes after TPI using pre-computed sextant tracking data. This required the simulation to be stopped at that point, and then resumed after the state vector update had been made. After the update had been made and the simulation run resumed, the pilot called for the midcourse correction program to compute the required midcourse correction based on the updated state vectors. The pilot then made the midcourse correction at 17 minutes after TPI. Maneuvers were accomplished with the CSM X-axis boresighted to the target LOS using the manual RCS along the three spacecraft body axes. The terminal phase LOS and range-rate control procedures were completely manual and were essentiall the same for both primary and backup modes. These procedures are described in a later section.

Simulation Backup Mode Procedures - Considering the system contingency cases a CSM is unlike the Lunar Module, or Gemini. In Gemini, it was assumed that if the onboard guidance computer failed, the terminal rendezvous portion of the mission could be completed using data from the inertial platform, radar displays of range and range rate and the flight charts designed for rendezvous. In the case of a CM rendezvous, however, range and rage rate estimations are derived through optical sightings which are processed in the onboard computer. Thus, the loss of the computer infers loss of range and range rate data.

At TPI, the maneuver was made in the same manner as the primary mode, i.e., the nominal TPI ΔV was set into the ΔV meter and then thrust was applied until the ΔV meter read zero. However, for the midcourse phase, a simplified technique for determining midcourse corrections was developed which uses only LOS information. In the development of this technique, it was determined that a successful rendezvous could consistently be made with trajectory dispersions and state vector uncertainties as high as 10,000 ft and 10 fps at TPI.

This technique is one which requires only the measurement of inertial line-of-sight change vs time (LOS rates). Introduction of this data into a nomogram type chart (figure 12), which is referenced to time after TPI, provides the magnitude of the required orthogonal coplanar ΔV maneuvers to be made along and normal to the target line-of-sight. The procedure requires four line-of-sight measurements and a possibility of four midcourse corrections following TPI. The basic measurement is the time required for the target inertial LOS to traverse 40 milliradians as determined by the CSM pilot using the Crew Optical Alinement Sight (COAS).

The four measurements are initiated at 1:13, 4:50, 8:50, and 13:25 after TPI respectively. The schedule of these sightings was determined by the probable time required for the sighting and the associated maneuver. It was determined, however, that the measurement at 1:13 after TPI could be omitted if an up-down correction were made at TPI using a Gemini TPI chart.

Because of the desire to guarantee rendezvous without a control-and-monitor device such as radar, it was necessary to design the midcourse correction chart so as to give an adequate closing velocity in the face of lower than normal energy transfer orbits. Generally, this design is characterized by a 5-10 fps higher range rate at intercept than is experienced in the no-error case (21 fps for 10 mile ΔH). The lowest closing rate at intercept experienced using the backup technique was 15 fps.

In all rendezvous cases where no direct ranging data are available, one of the major objectives of the midcourse correction is to eliminate the possibility of a "low apogee" miss; i.e., a case where the chase vehicle orbital energy at apogee is inadequate to reach the target vehicle altitude (fig 10, trajectory 2-10).

Although the primary guidance attempts to establish an exact intercept based on its best estimate of the relative state vectors, the backup guidance chart previously described was designed to establish a trajectory which will either intercept the target, or if a miss occurs, it will always be behind the target and of no greater a distance than about 1500 feet at intercept altitude. This type of trajectory will always have enough orbital energy to get up to the altitude of the target. The reason for this is that the midcourse corrections maintain the range rate at a slightly higher-than-nominal value throughout the transfer. The procedural reason for the backup chart being designed this way is that it is much easier to detect and correct a miss which passes behind the target than one which misses in front. Where no direct ranging information is available, the pilot simply has to rely on LOS angular motion to determine his relative trajectory errors. A miss on the back side of the target is easily determined and corrected by monitoring the LOS rate, whereas, LOS rate is insensitive to some trajectory errors which cause a miss on the front side. More specifically, on a miss which is caused by insufficient orbital energy, the chase vehicle apogee is reached before the target altitude is reached. The chase vehicle then begins to descend in altitude. However, during this trajectory, the LOS rate will indicate an intercept because it will appear near nominal.

Terminal Phase LOS Rate Control - Following the midcourse correction phase (primary or backup), a technique of LOS rate control was used to eliminate the residual velocity errors, and yet minimize any possible deteriorating effect on the intercept velocity. Figure 13 shows three types of miss trajectories which can occur and the terminal LOS rate control philosophy which was used on each. Two of the trajectories are termed "high energy miss cases" because the chase vehicle reaches the target vehicle altitude even though a miss occurs. The third trajectory is termed a "low energy miss case" because it did not reach the target vehicle altitude. As can be seen from the backup chart of figure 12, the nominal LOS rate becomes zero at 21 minutes after TPI; thus, the essentials of terminal phase LOS rate control are to see that the LOS rate reaches near-zero at 21 minutes after TPI. It is then maintained near zero until intercept. If the backup midcourse corrections are executed properly, the LOS rate will be slightly high at 21 minutes after TPI, that is to say, the target inertial LOS will still be moving slightly downward (negative LOS rate relative to the pilot). This would result in a high energy miss behind the target (case 1). To control the LOS rate to a nominal zero at TPI + 21 minutes on this trajectory requires a translation maneuver that is perpendicular to the LOS at point 1. Another maneuver of the same type can be made at point 2 if necessary. These LOS maneuvers can only increase the orbital velocity of the chase vehicle, and thus, the intercept velocity will not be adversely affected. The amount of ΔV required is estimated using the best estimate of range and the timed LOS rate in the equation $\Delta V = R$ (ft) x LOS rate (mr/sec) x 10^{-3}.

If the midcourse corrections are not properly executed, either a high or low energy miss in front of the target can occur (case 2). It is very difficult to distinguish one from the other before the "low energy case" has progressed so far that it is almost impossible to control. These statements would not be true if direct ranging information were available. Both of these trajectories have a characteristic LOS rate which reaches zero sooner than TPI + 21 minutes. The basic difference between the two trajectories, however, is that the high energy case has a constantly increasing LOS rate in the positive direction (target moves up in pilot's window); whereas, the low energy trajectory has a LOS rate which is characteristic of the nominal trajectory (after LOS rate has changed from minus to plus). This means that, for the low energy case, the LOS rate will reach zero slightly early, then will increase in the positive direction up t approximately 0.1 mr/sec, and then decrease to zero and remain there.

In making a terminal phase correction on either of these trajectories, it is best to thrust toward the target, at point 1, with a ΔV of about 10 fps. If a translation maneuver were made, at this point, perpendicular to the LOS to control the LOS rate back to zero, the orbital velocity of the chase vehicle would be reduced and thus the situation would be aggravated. At point 2 of the high energy miss in front of the target, the pilot will be able to distinguish that it is a high energy case because the local vertical pitch angle (θ) and the LOS rate will be steadily increasing. A correction can then be made perpendicular to the LOS to control the LOS rate to zero. The downward component of the LOS correction does not visibly affect the intercept since the ΔH is small by this time and differential gravity effects are negligible. At point 2 of the low energy case, the thrust must be directed slightly below the LOS (relative to the pilot) to effect an intercept. This is a very costly intercept and should be avoided if possible.

Terminal Phase Range Rate Control - When the onboard computer (CMC) is operating, the computed range and range rate are displayed on the DSKY and are used as long as they appear valid. However, since the CMC is updated at intervals with the sextant rather than continuously, the range and range rate information displayed on the DSKY can become degraded to various degrees depending on the sextant sighting schedule. Figures 14a through d give time histories of the actual and onboard computed range rate versus range for a 15 n mi ΔH transfer with various initialization errors prior to TPI. It can be seen that, even with errors of 659 feet and 1 fps just prior to TPI, the onboard computed range and range rate information could no longer be used after an actual range of 20,000 feet has been reached. As the initialization errors get larger, the divergence between the actual and computed data worsens as expected. From this, one can see the importance of updating the CMC after TPI. With a realistic sextant sighting schedule being used after TPI, studies have indicated that the range and range rate errors in the CMC propagate to about 3500 ft and 4 to 5 fps at an actual range of a mile. With this magnitude of error, the preplanned braking schedule given in Table II can be followed down through the 1 n mi range gate. From that point, the pilot must rely on visual cues and normal range/range rate parameter behavior for the remaining braking maneuvers.

In the backup mode without direct range measurements (and also for the last mile of operation in the primary mode), the CSM pilot is totally dependent upon what he can perceive visually for controlling the CSM closing rate to the target. Previous studies have described the pilot's braking capability using the apparent change in size of the target as adequate for the maneuver when this phase occurs during daylight. However, if the rendezvous is targetted for intercept in darkness or, because of pre-TPI target/chase vehicle phasing errors, the intercept inadvertently occurs in darkness, the CSM pilot cannot determine the magnitude of braking maneuver to be made.

If the terminal arrival lighting conditions can be controlled, the CSM pilot can, by 4-5 fps braking maneuvers, adequately complete the rendezvous. He must, of necessity, approach the target faster than the normal range/range rate braking gate permits. Overcontrol of range rate could result in a slow approach and a poor approach angle. However, he also must not approach at too high a closing rate because (1) the CSM translation acceleration is low and (2) the LOS rate control problem can become very difficult within the last mile with a high closing rate. It was determined, however, that closing rates as high as 35 fps at a range of one mile could be handled with little difficulty.

Generally, targets of the size of the Agena or the Lunar Module will afford the pilot adequate range rate information from a distance of about 1 nautical mile. Larger targets such as a SIVB stage would provide information at proprotionally greater ranges. Simulation results have shown the pilot generally brakes the CSM in 5-10 fps increments at about one mile and 3-4 thousand feet, and then approaches at a rate of 10-15 fps until a range of 500-1000 feet is reached.

Control Modes - During the braking and LOS control phase, the pitch angle relationship to the target changes rapidly. Because of this change a considerable amount of RCS fuel can be used in attitude control. Therefore, because of the high pitch moment of inertia it is advantageous to operate the vehicle pitch channel in the minimum impulse mode while tracking the target. During LOS measurements the accuracy of inertial tracking in minimum impulse vs the attitude hold mode is a function of individual pilot training and control technique. However, LOS measurement in daylight in the CSM PNGCS-failed mode must be made in the attitude hold mode.

Test Program

Test Matrix - Prior to running a preplanned test matrix, several weeks were spent making simulator runs for the purpose of developing the CSM backup procedures and charts previously discussed. Subsequent to this, a test matrix of runs was made to evaluate the backup procedures and to compare the fuel performance of these procedures with that of the primary guidance. One pilot engineer and one non-pilot engineer were used as test subjects, both having had extensive training in flying rendezvous simulations prior to this study.

Only a few runs were made on the 20 n mi ΔH rescue before it was determined that the LOS and braking maneuvers required were too large for the CSM acceleration. Thus, this ΔH was discarded. Error cases which were run for the 5, 10, and 15 n mi ΔH transfers are as follows (see Table I for the description of error cases):

5 n mi ΔH: error cases (1), (2a), (3), (4a), and (5)

10 n mi ΔH: all error cases were run

15 n mi ΔH: error cases (1), (2a), (4a), and (5)

Three to five runs were made on each error case flown for the three different transfer altitudes; thus, a total of approximately two hundred 45 minute runs were made during the study.

Data Acquisition - Data were obtained using digital printouts, 8-channel strip recorders, and X-Y plotters. The digital printout was used to record the end conditions of each run, such as attitude fuel, translation ΔV and final position and velocity of the CSM relative to the LM. The 8-channel recorders were used to obtain time dependent information such as attitude and translation thruster duty cycles. Plots of the CSM relative motion and range rate/range profiles were made on the X-Y plotters.

Simulation Data

Attitude and Translation Fuel Data - The attitude and translation fuel data obtained in the simulation are summarized in Table III for the various ΔH's investigated. It was determined quite early in the simulation that a ΔH as large as 20 n mi should not be used in a CSM active rendezvous because the intercept velocity (47 fps) was too large to handle with the CSM translation acceleration without the primary system operating or ranging data available. Thus, that ΔH was discarded and data were recorded only on 5, 10, and 15 n mi ΔH transfers. In Table III, the total fuel used has been broken down into body axis. Moreover, the translation ΔV has been listed with respect to the RCS and SPS ΔV used at TPI, RCS ΔV used for range rate and LOS rate corrections at each midcourse correction, and finally, RCS ΔV used for braking and LOS rate corrections at TPF. It should be noted that the RCS jets were used for the complete TPI burn on all 5 n mi ΔH transfers because the TPI maneuver (10.42 fps) was marginally small for the SPS. However, for 10 and 15 n mi ΔH transfers, the TPI maneuvers are 20.97 and 31.49 fps respectively; therefore, the SPS was fired at TPI after a nominal 3.8 fps (20 seconds) RCS ullage maneuver had been performed.

Another point which should be noted is that a rendezvous was not possible on case 5-5 with the backup chart as designed because the dispersions which existed were too large in relation to the ΔH being flown. The backup chart could be designed to account for that case, but probably at the expense of using more ΔV on the other trajectories flown.

Effect of Error in Knowledge Velocity Vector on ΔV Penalty - The data obtained indicated a definite correlation between ΔV penalty and the initial knowledge of the velocity vector prior to TPI. Figure 15 gives plots of ΔV penalty for the range of initial velocity errors which were investigated in the simulation. The data also indicated that the same relation of ΔV penalty with initial velocity error held true for velocity errors as large as 4 fps without respect to the ΔH being flown. At that point, however, the ΔV penalty for a 5 n mi ΔH diverged from that of 10 and 15 n mi ΔH transfers. Since no primary mode runs were made on a 5 n mi ΔH with errors larger than 2.1 fps, it is not known if a divergence would occur for the primary mode as it did for the backup mode. However, in general, the data indicated that a 5 n mi ΔH should not be flown when large dispersions are possible. There is a net difference of 22 fps ΔV penalty between the primary and backup modes. This is due to the fact that CMC state vector updates are made after TPI in the primary mode, and thus, more accurate midcourse corrections are made.

Effect of Velocity Error on Attitude Fuel - The attitude fuel used on any ΔH trajectory varied with respect to the individual pilot and the initial velocity error. Figure 16 gives upper and lower bounds of attitude fuel usage (shaded area) dependent upon the initial velocity error. For a zero velocity error (nominal trajectory), the average amount of fuel used in attitude control was 55 pounds. The average asymptotically approached 100 pounds as the velocity error was increased. The dispersion of attitude fuel data points above and below the average was high, however, and did not reflect any consistent difference between primary or backup mode control. In all flights, the pilot used minimum impulse attitude control during coasting periods, but an effort was not necessarily made to minimize the attitude fuel usage. Therefore, with good training it is expected that the operational crew man could exceed the performance indicated.

Fuel Requirements for Future Rendezvous Missions - From the results obtained in this simulation, it is possible to predict how much fuel will be required on any future rendezvous as long as the control mode and TPI velocity errors are known. To determine the total fuel required, one must add together the impulsive TPI and TPF maneuvers, the ΔV penalty associated with the TPI velocity error, the station keeping and docking ΔV, and the fuel required for attitude control for the whole rendezvous sequence. Figures 17-19 give the predicted fuel requirements for the planned CSM rendezvous on AS-258 and for CSM rescues of a passive LM on AS-258, AS-503, and AS-504.

CONCLUSIONS

1. The CSM possesses satisfactory rendezvous capability using the Concentric Flight Plan if:

   a. The pre-TPI ΔH is constrained to approximately 10-15 n mi. The upper bound is limited by the translational acceleration capability of the CSM to cope with high intercept velocities and the SM RCS fuel available for rendezvous. The lower bound is constrained by the uncertainty in the knowledge of the vehicle state vectors when applied to midcourse maneuvers and the resulting ΔV penalty.

b. The primary system (PNGCS & MSFN performance as defined herein) is operating.

c. The rendezvous phasing is constrained to allow at least the last 1 n mi of closure to occur in daylight. This constraint can be removed by the addition of an independent ranging device on the CSM.

2. Without independent ranging the CSM still has a backup rendezvous capability, however it uses an average of 88 lb more SM-RCS fuel than the primary mode. Also, this mode can result in non-standard final approach conditions in terms of line-of-sight angle and range rate.

3. For the ΔH range considered, there is a correlation between velocity vector uncertainty and ΔV penalty which allows an estimate (for budgeting purposes) of the fuel required for rendezvous.

Table III - Average Fuel Data for 5, 10, and 15 NM. ΔH Transfers

Case	ΔH (NM)	Procedure Used	Initialization Errors ϵ_R (FT)	Initialization Errors ϵ_V (FPS)	Update Errors ϵ_R (FT)	Update Errors ϵ_V (FPS)	Attitude Fuel W_P (LBS)	Attitude Fuel W_Y (LBS)	Attitude Fuel W_R (LBS)	Attitude Fuel W_T (LBS)
1-5	5	B	0	0	-	-	8.75	17 88	7 04	33 67
2a-5	5	B	650	1	-	-	13 29	49 93	15 80	79 01
3-5	5	A	1200	2 1	1150	2 2	8 90	57 70	24 12	90 72
4a-5	5	B	2000	3	-	-	13 46	47 99	15 79	77 23
5-5	5	B	4000	6	-	-	13 15	44 34	20 59	78 08
1-10	10	B	0	0	-	-	11 46	27 49	13 36	52 31
2a-10	10	B	650	1	-	-	12 68	65 99	29 95	108 62
2b-10	10	A	650	1	430	0 8	6 71	42 84	11 94	61 49
3-10	10	A	1200	2 1	1150	2 2	10 12	41 01	28 37	79 50
4a-10	10	B	2000	3	-	-	13 86	46 99	25 13	85 98
4b 10	10	A	2000	3	460	1 2	9 13	59 75	26 54	95 42
5-10	10	B	4000	6	-	-	15 47	73 75	39 40	128 61
6-10	10	B	4000	6	-	-	23 79	57 04	1 91	82 74
7-10	10	B	9500	.9	-	-	22 41	31 88	0 70	74 98
1-15	15	B	0	0	-	-	14 29	31 46	14 92	60 67
2a-15	15	B	650	1	-	-	12 03	33 52	16 78	62 33
4a 15	15	B	2000	3	-	-	17 32	70 89	26 15	114 56
5 15	15	B	4000	6	-	-	20 61	74 20	38 04	132 85

Case	RCS Translation ΔV ΔV_x (FPS)	ΔV_y (FPS)	ΔV_z (FPS)	ΔV_T (FPS)	TPI ΔV RCS/SPS	TPM #1 ΔV R/LOS	TPM #2 ΔV R/LOS	TPM #3 ΔV R/LOS	TPM #4 ΔV R/LOS	TPF ΔV BRK/LOS	Total Fuel (LBS) RCS	Total Fuel (LBS) SPS
1-5	22 40	0 16	2 36	24 92	104/0	-	-	-	-	12/2.52	13335	0
2a-5	37 30	5 32	27 49	70 61	107/0	325/650	-	-	-	23 5/26 21	36145	0
3-5	45 99	4 95	16 39	67 32	104/0	127/10	-	-	-	22 85/20 3	36000	0
4a-5	38.20	13 55	36 47	88 22	102/0	45/900	20/40	05/10	-	21.0/28 08	43011	0
5-5	42 54	4 68	44 44	91 66	170/0	70/140	10/20	100/200	-	10 000 FT MISS	-	0
1-10	30 00	3 23	2 49	35 72	38/1720	-	-	-	-	26 06/5.72	195.19	57 45
2a-10	37 55	4 88	37 90	80 33	38/1720	35/70	10/20	088/175	-	28 38/32 11	429 94	57 45
2b-10	34 52	4 77	15 37	54 86	38/1720	63/68	-	-	-	24 4/13 53	280 93	57.45
3-10	33 90	2 82	10 60	47 32	38/1790	76/08	-	-	-	22 5/12 83	268 78	59 79
4a-10	41 05	10 15	43 63	94 84	580/1920	55/110	3.0/60	3.25/650	-	23.5/30 28	465 34	64 13
4b 10	36 30	7 14	40.95	84 39	380/1726	109/240	-	-	-	22.1/24 09	432 98	57 65
5-10	53 00	12 37	77 21	142 62	380/2240	128/256	50/100	-	-	31 4/53 84	699 09	74 82
6-10	42 30	5 62	48 42	96 34	580/1720	20/80	0/15	0/20	4/55	32 5/37 04	468 10	57 45
7-10	39 55	7 66	48 51	95 72	48/1720	0/205	0/70	175/350	675/130	32 3/25 17	457 86	57 45
1-15	42 30	4 03	14 60	60 93	38/2645	-	-	-	-	38 5/18 63	304 39	88 34
2a-15	50 30	5 40	37 84	93 54	38/280	25/50	-	-	-	44.0/38 24	436 49	93 32
4a 15	58 90	15 30	63 18	137 38	38/280	100/200	10.0/20.0	-	-	35 0/38 48	664 08	93 52
5 15	53 80	27 66	79 57	161 03	38/280	75/150	50/100	50/100	25/50	30 0/67 23	776 97	93 52

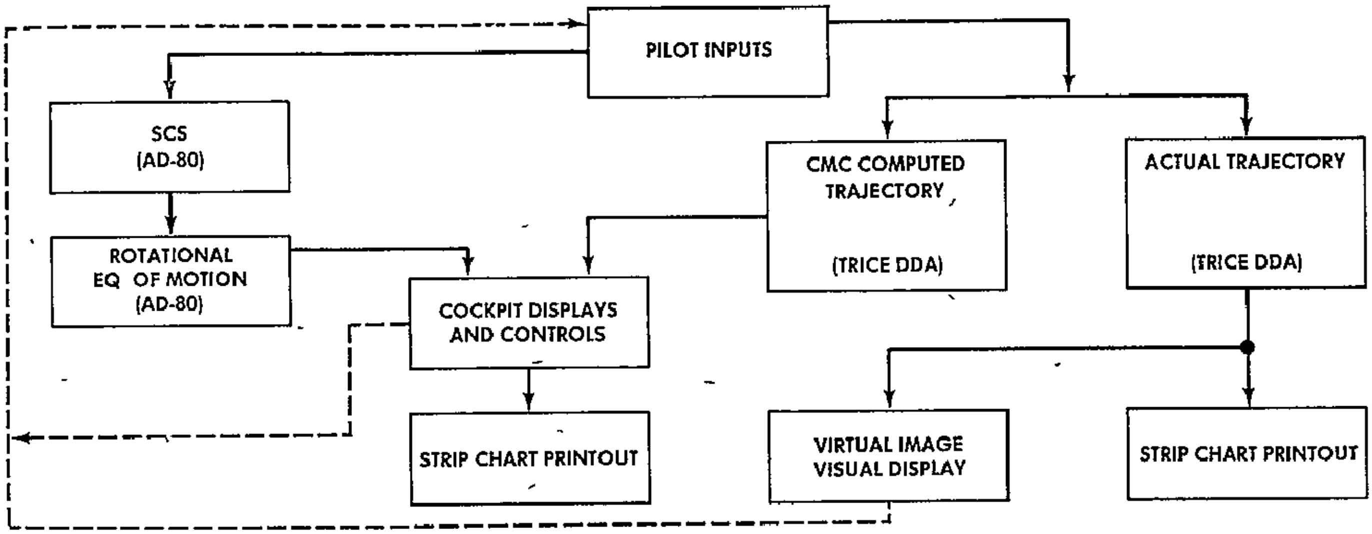


Figure 1. - Simulation block diagram.

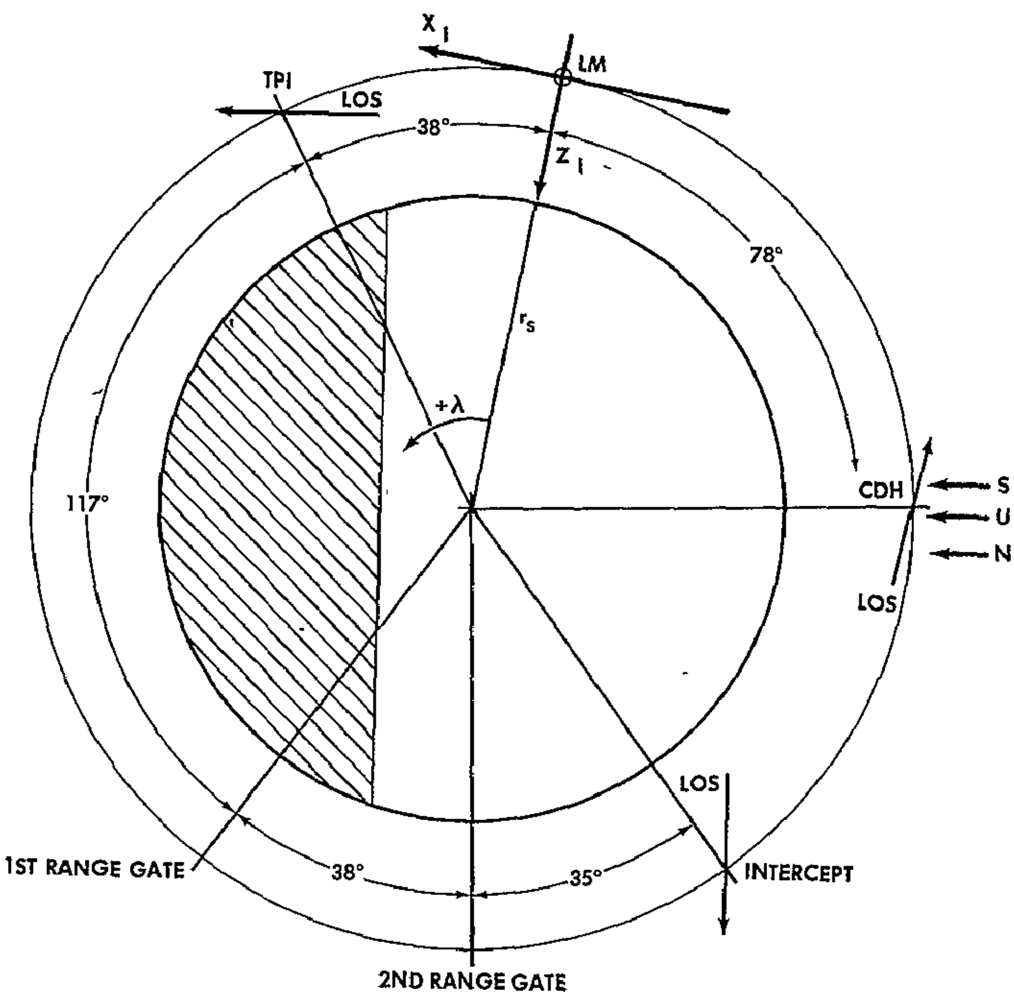


Figure 2. - Preliminary reference trajectory for Apollo Mission AS-278.

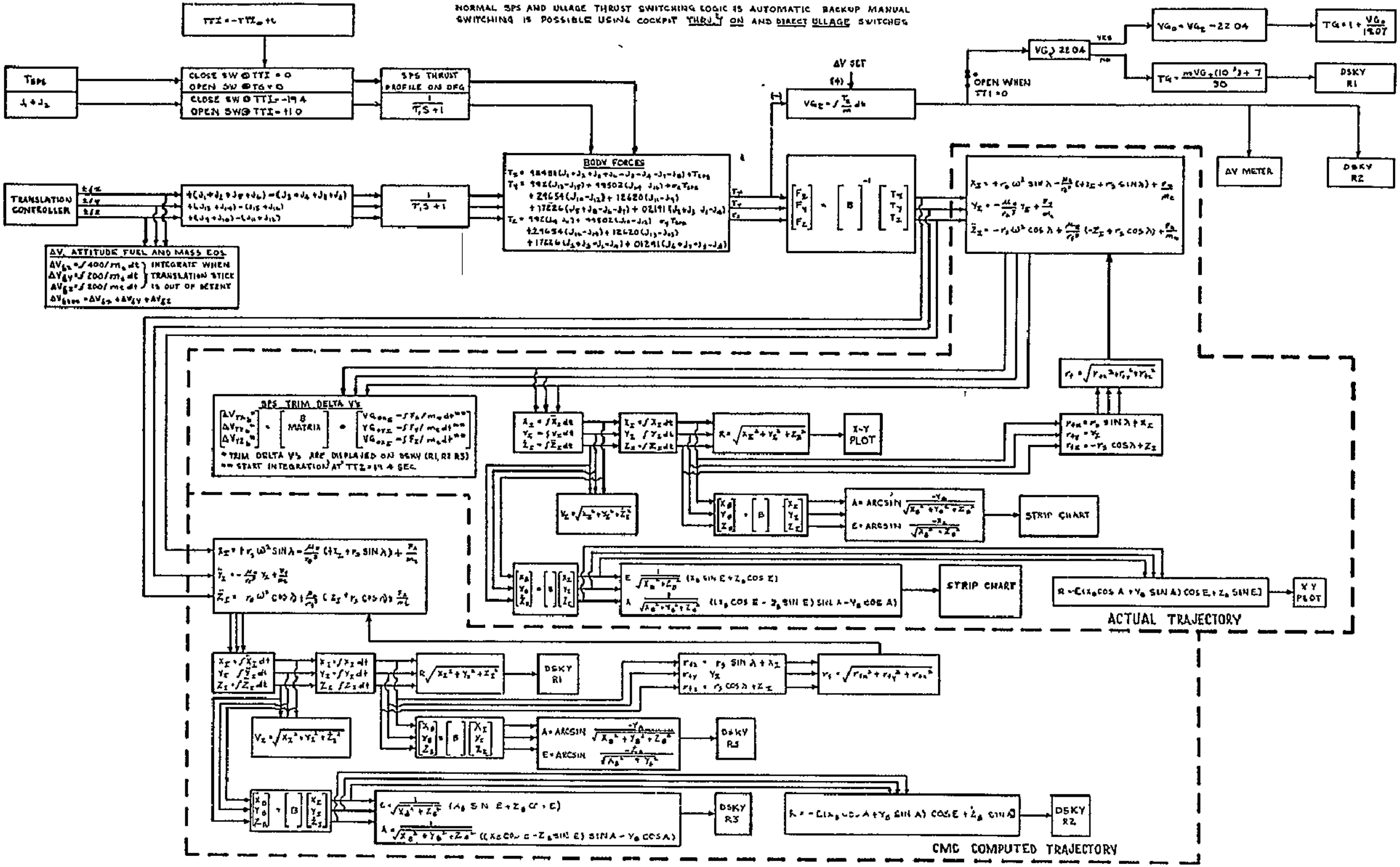


Figure 3 - Simulated Actual and Onboard Trajectory Computations

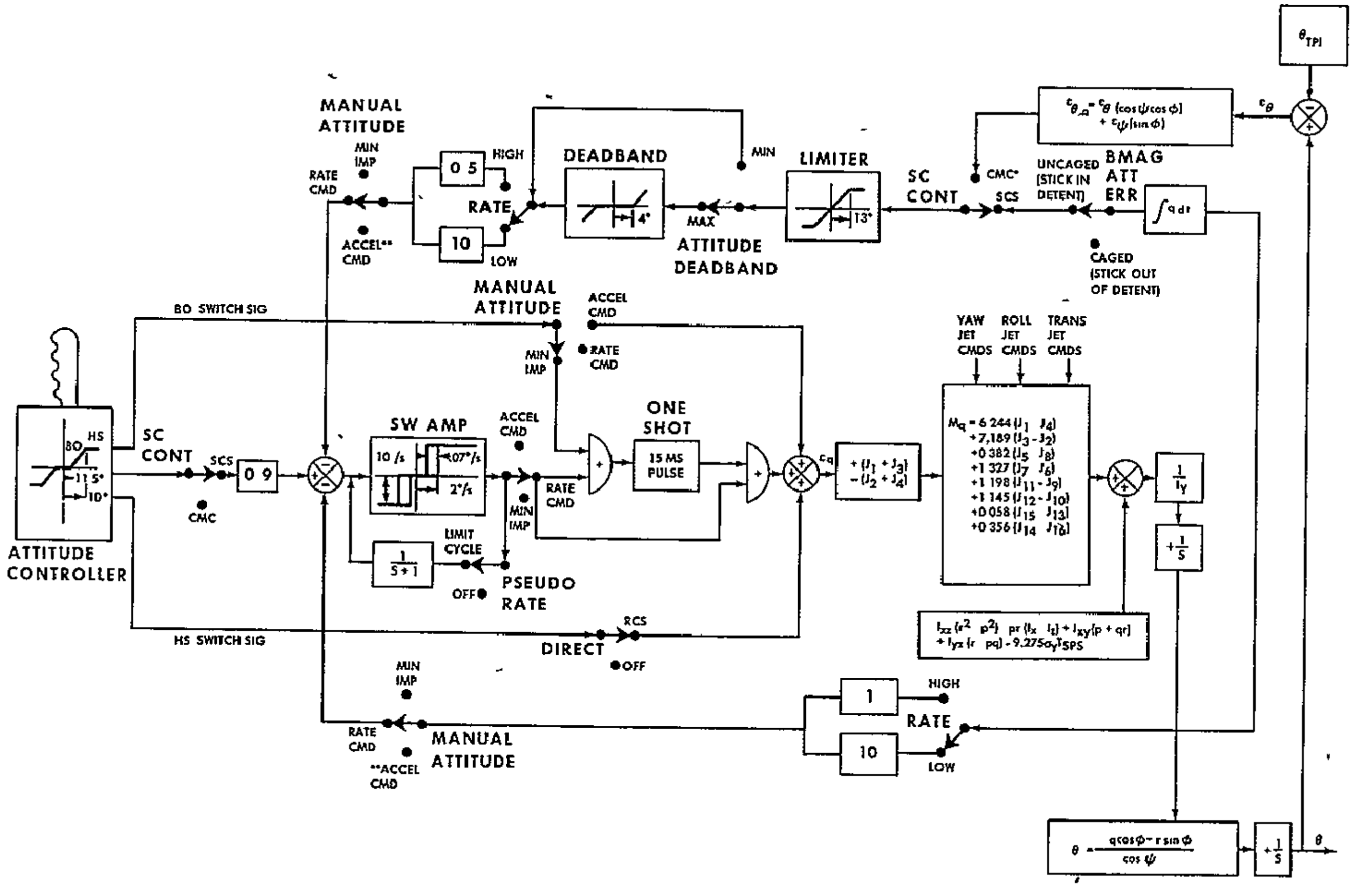


Figure 4a. - Simulated CSM/SCS (pitch channel).

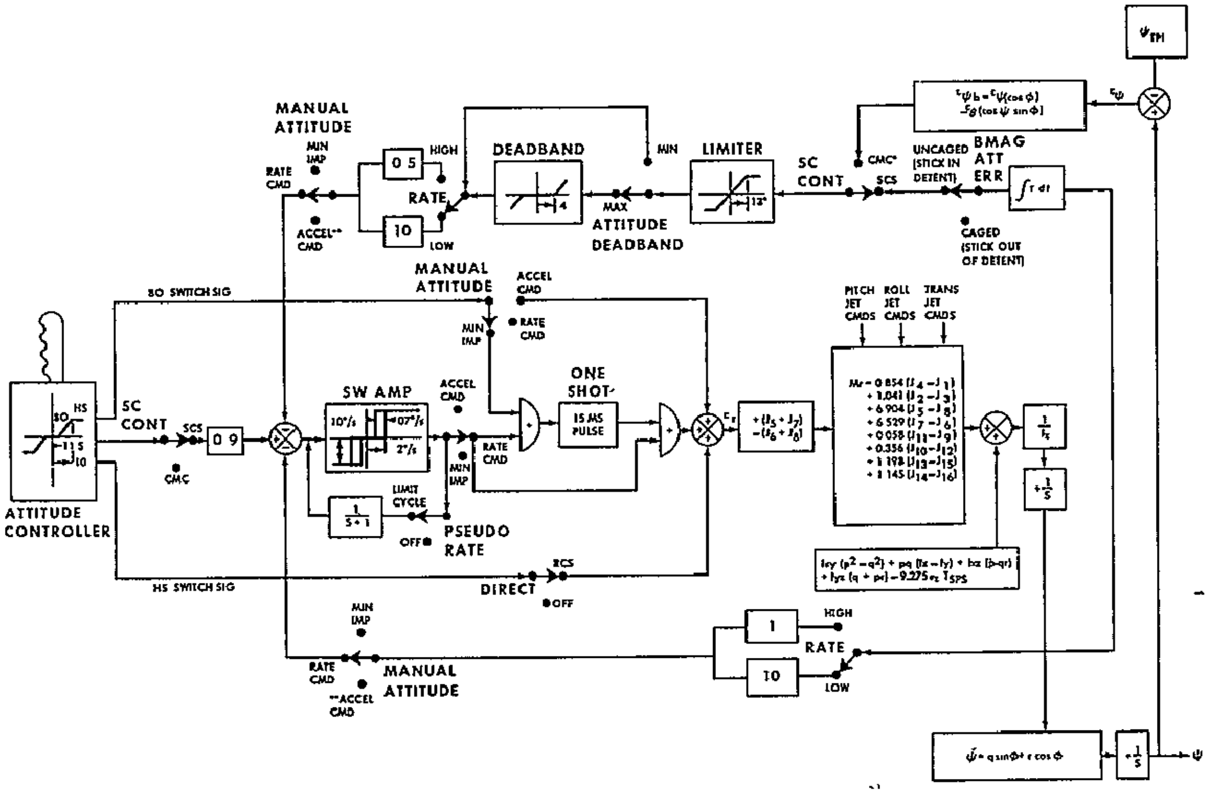


Figure 4b. - Simulated CSM/SCS (yaw channel).

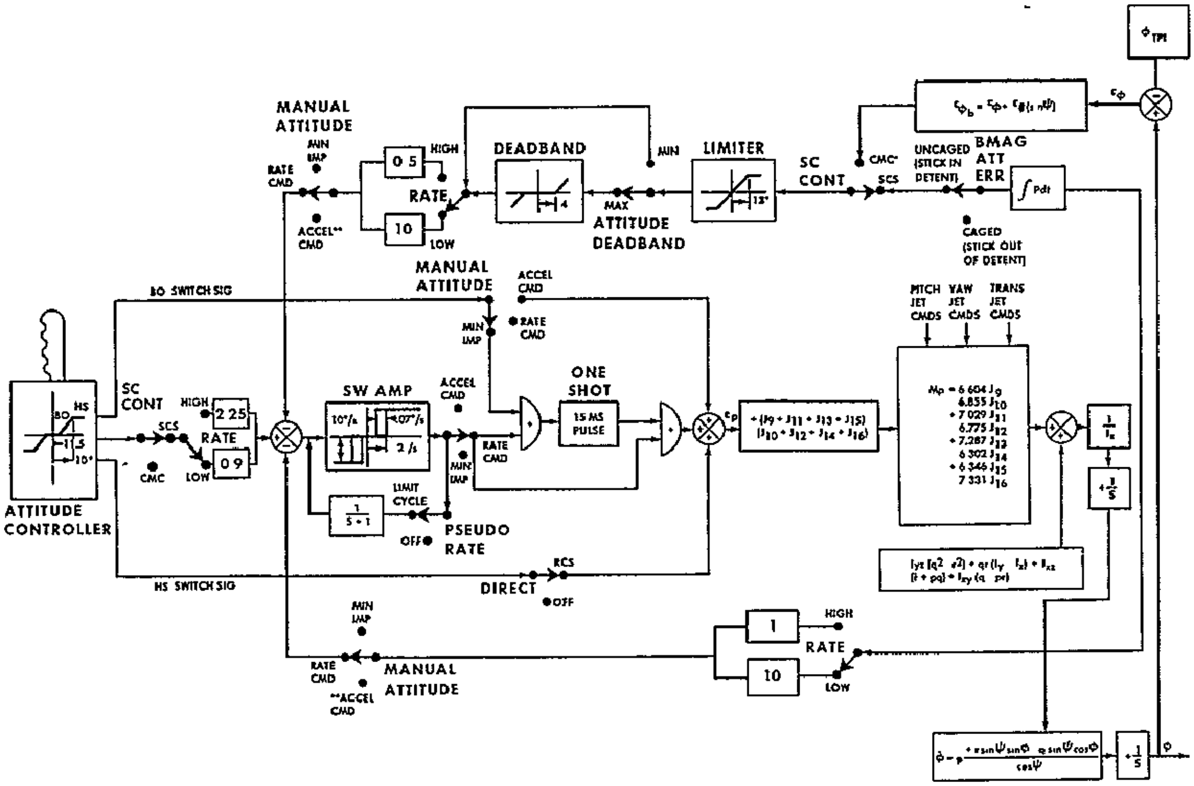


Figure 4c. - Simulated CSM/SCS (roll channel).

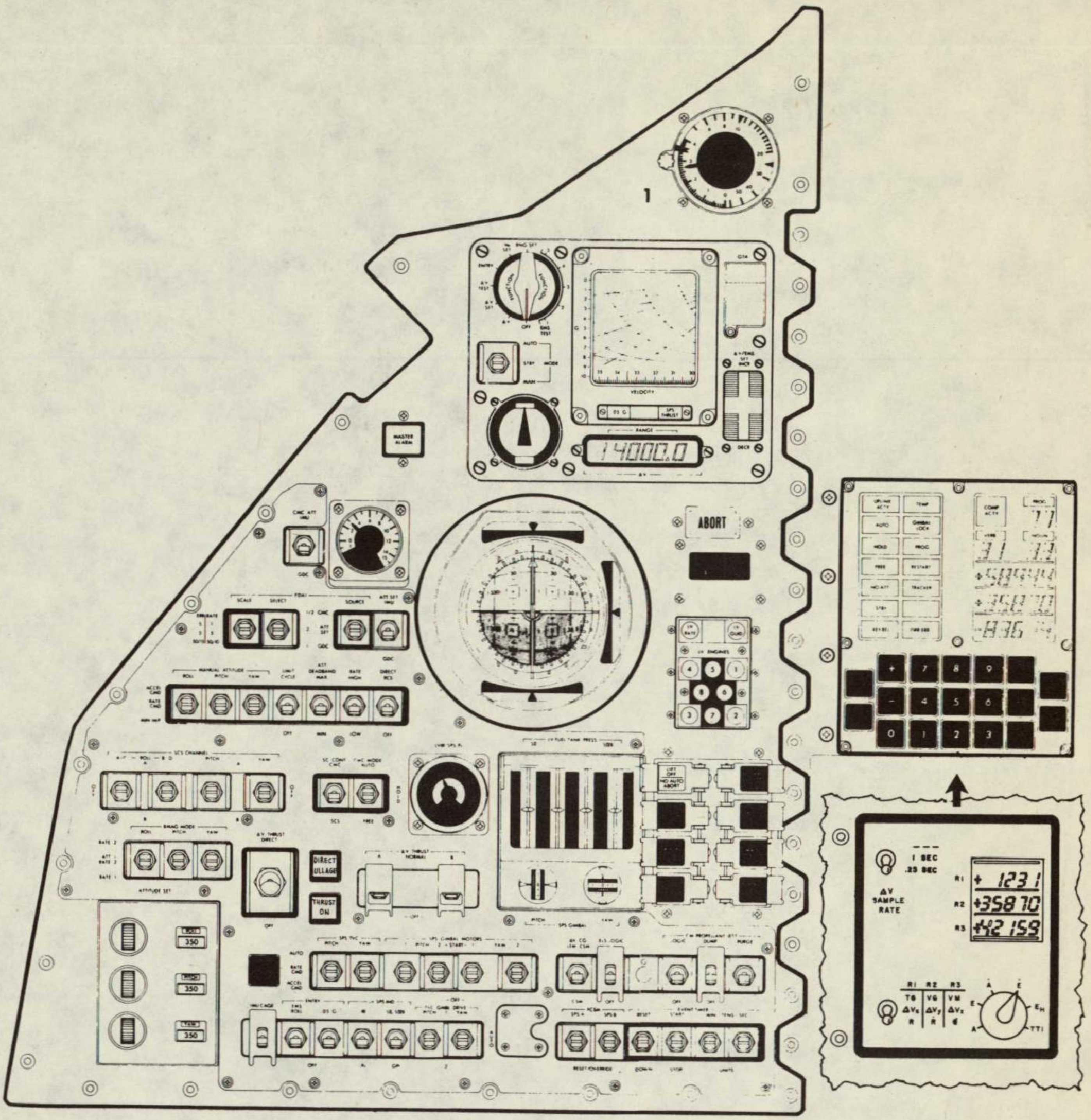


Figure 5a. - Simulated CSM displays and controls.

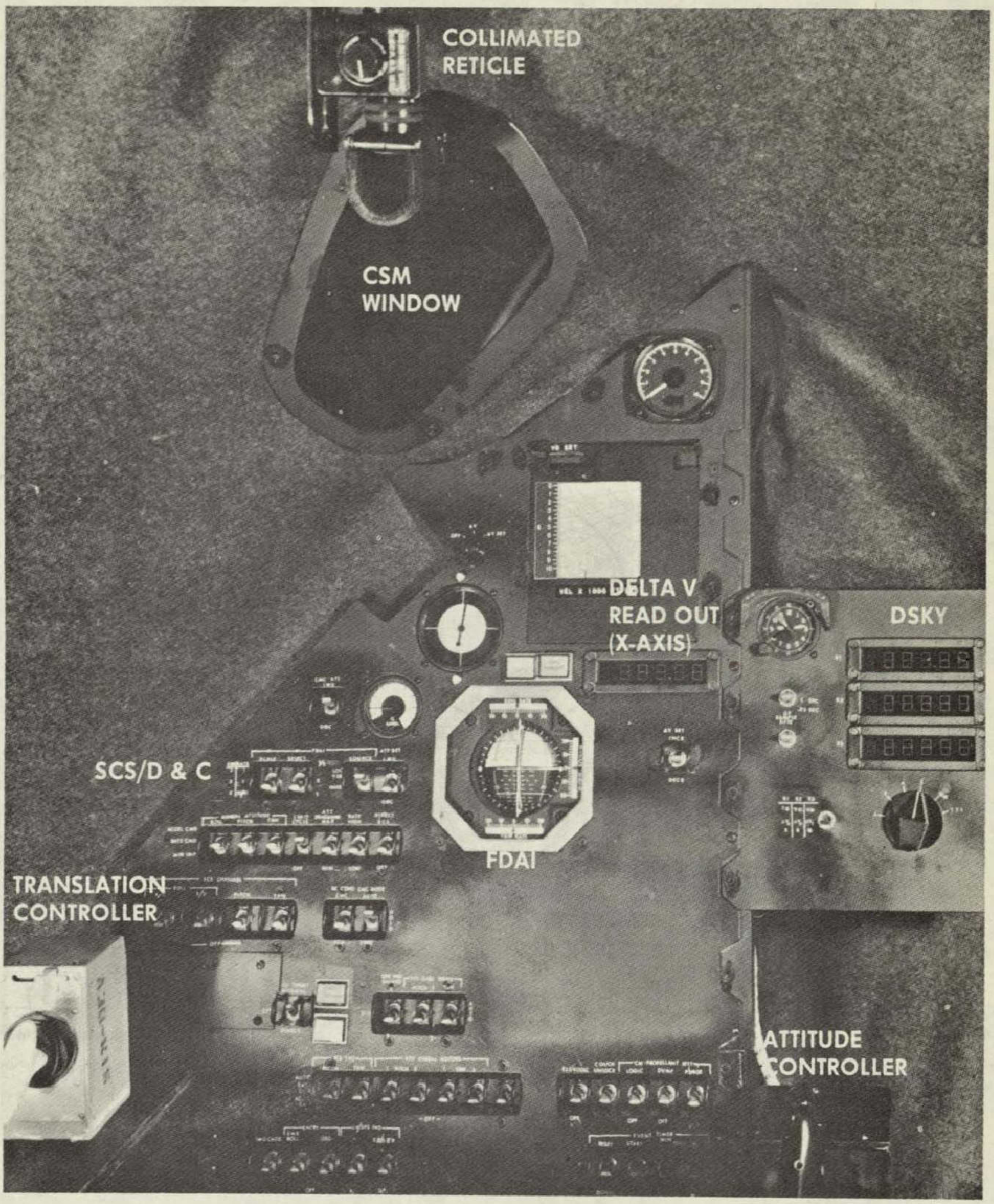


Figure 5b. - Mockup of CSM displays and controls.

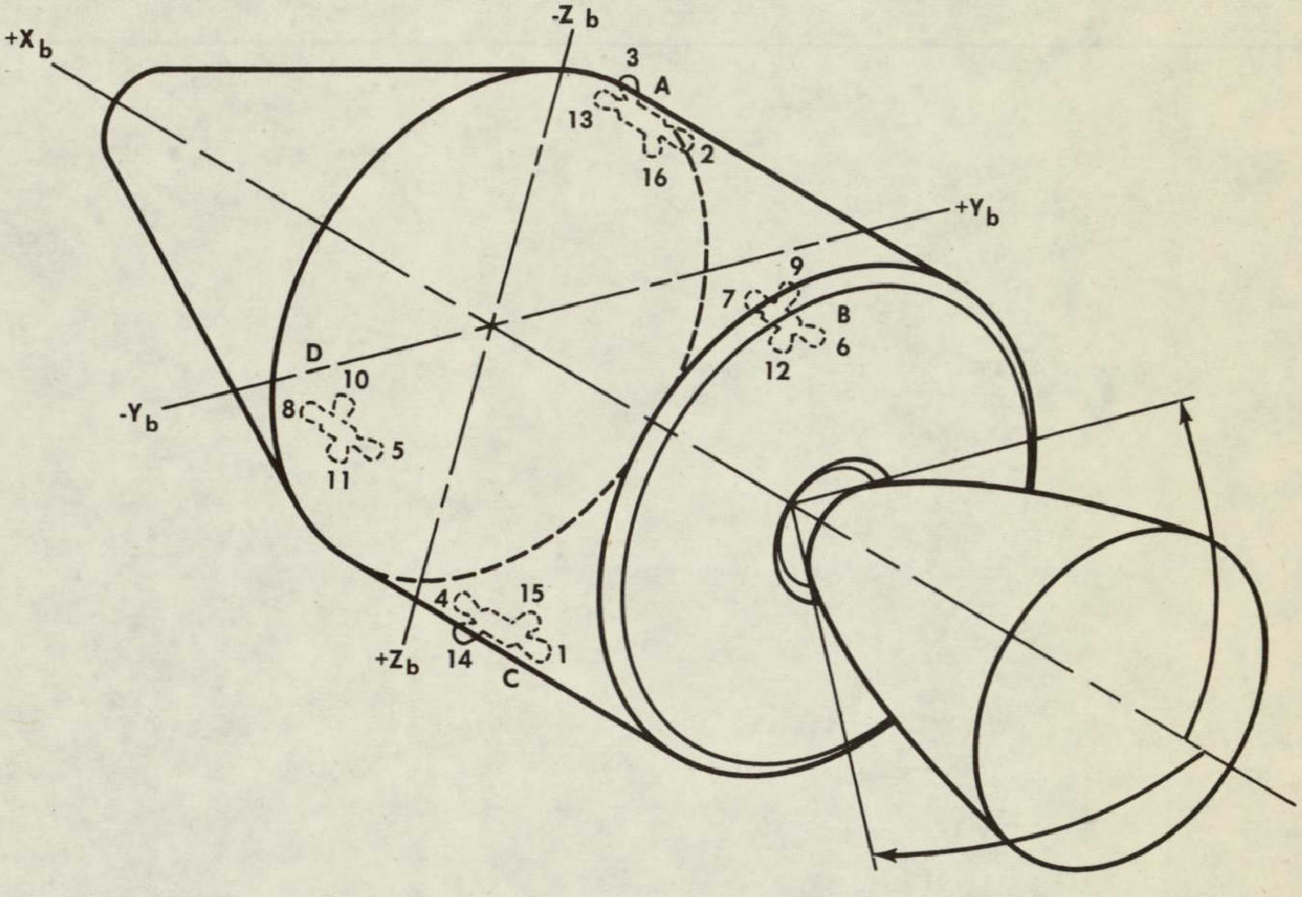


Figure 6a. - CSM geometry (RCS jet quad locations).

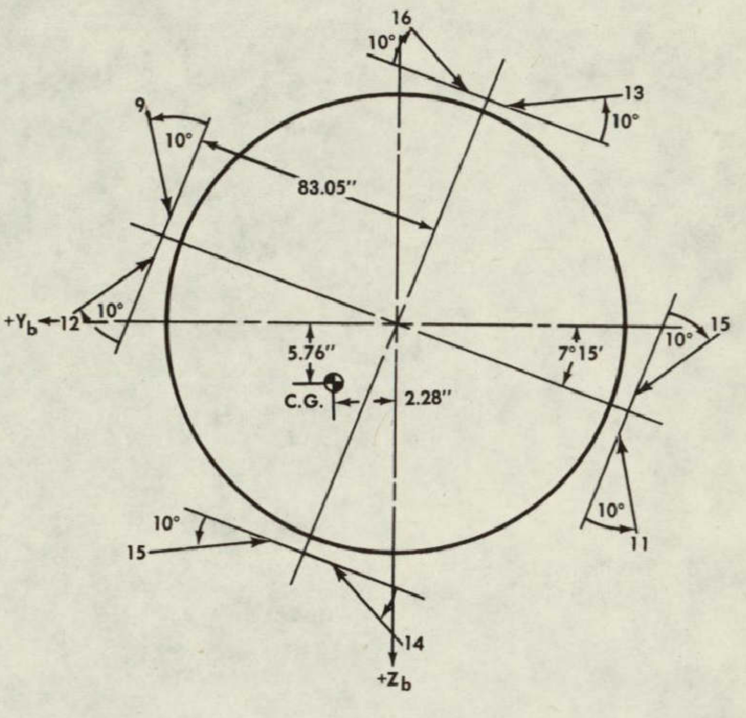


Figure 6b. - CSM geometry (roll jet geometry).

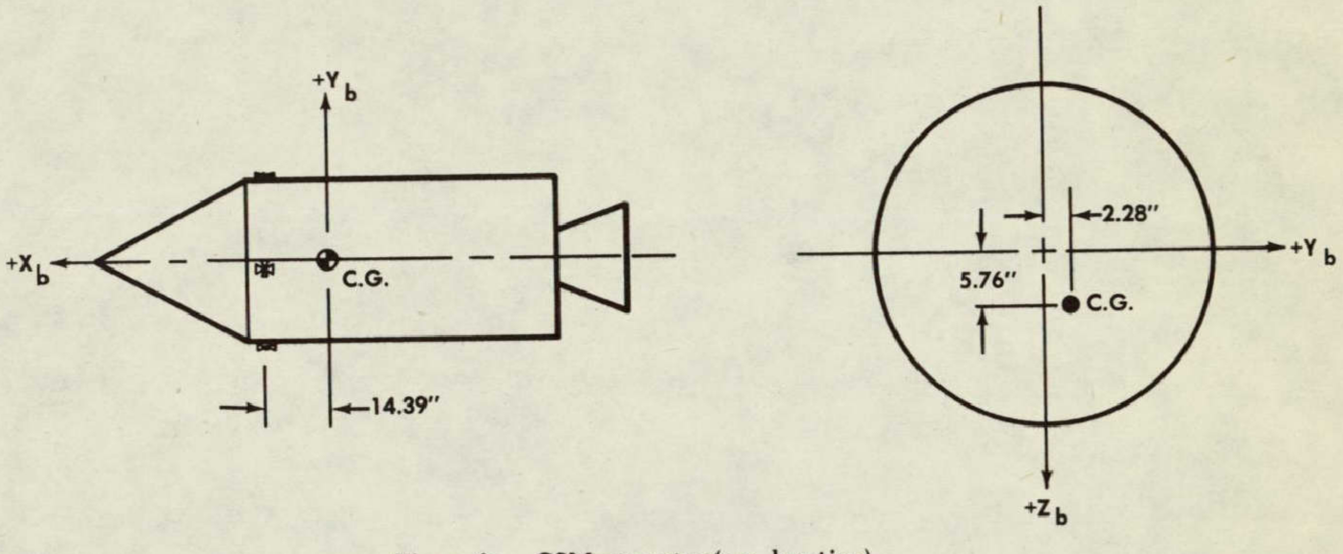


Figure 6c. - CSM geometry (c.g. location).

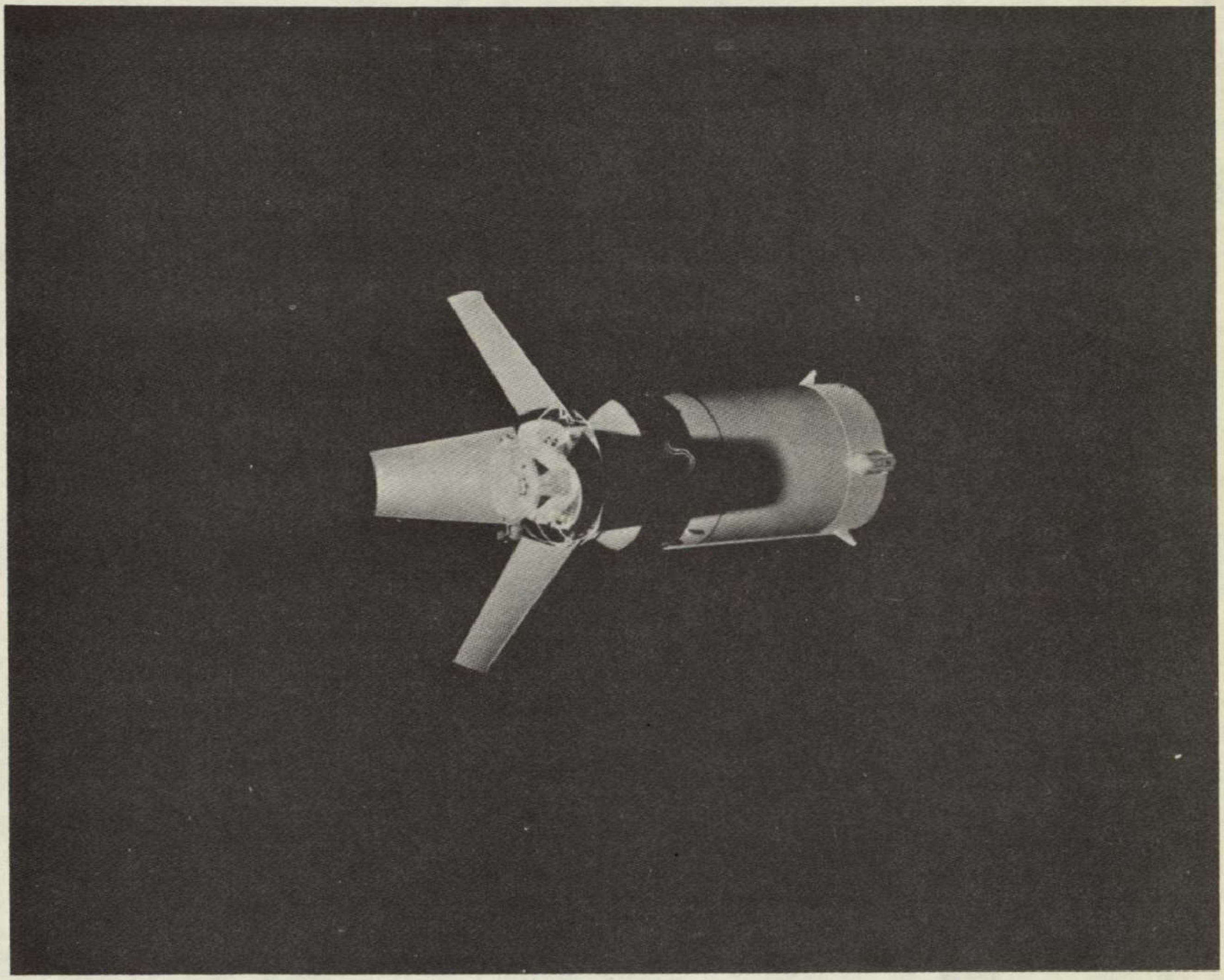


Figure 7. - Simulated LM/S-IVB.

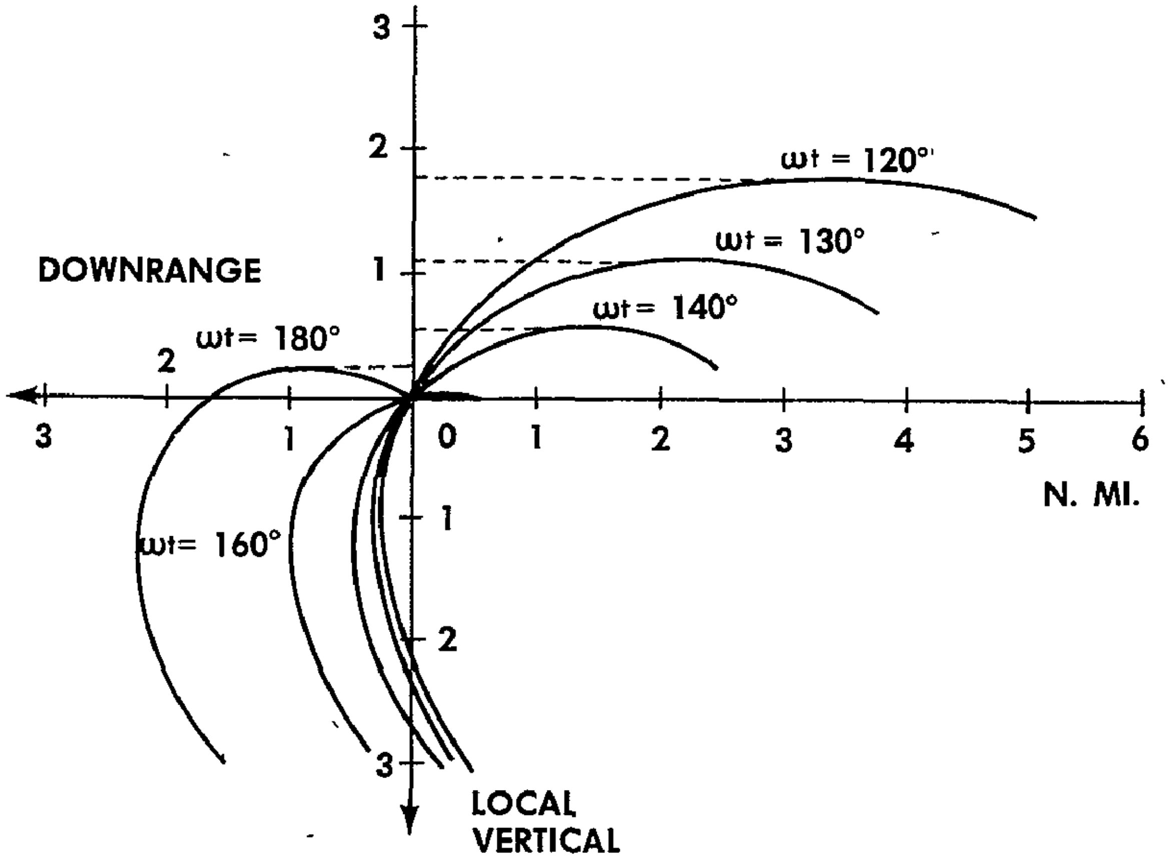


Figure 8. - Coelliptic rendezvous orbit energy for various transfer angles (10-naut-mile ΔH).

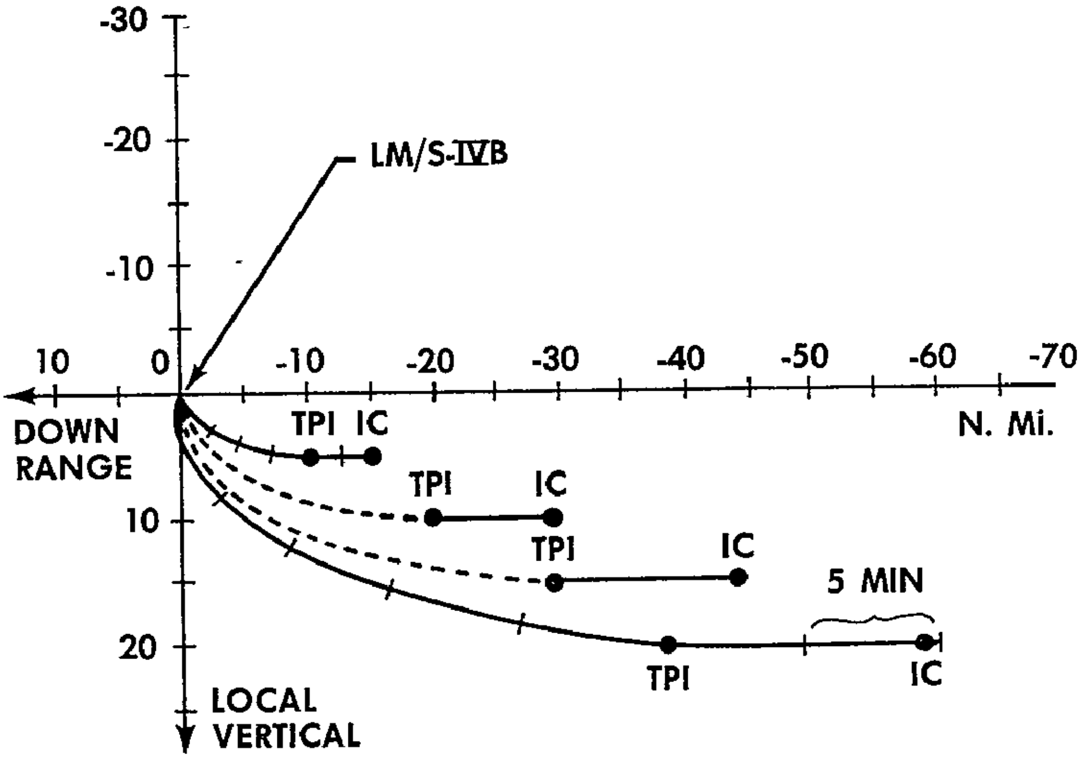


Figure 9a. - Simulated reference trajectories - local vertical plot of CSM position relative to LM.

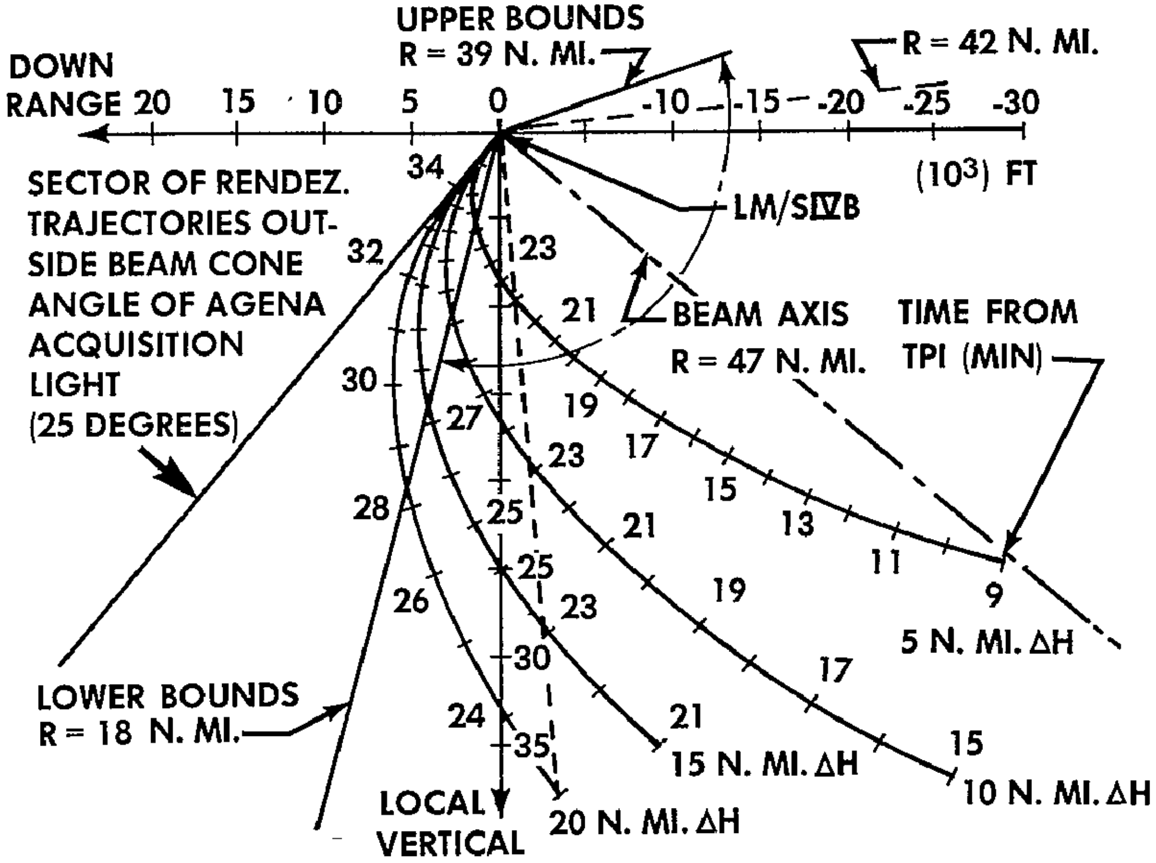


Figure 9b. - Lighting coverage of the Agena acquisition light for SLA panel opening of 50°.

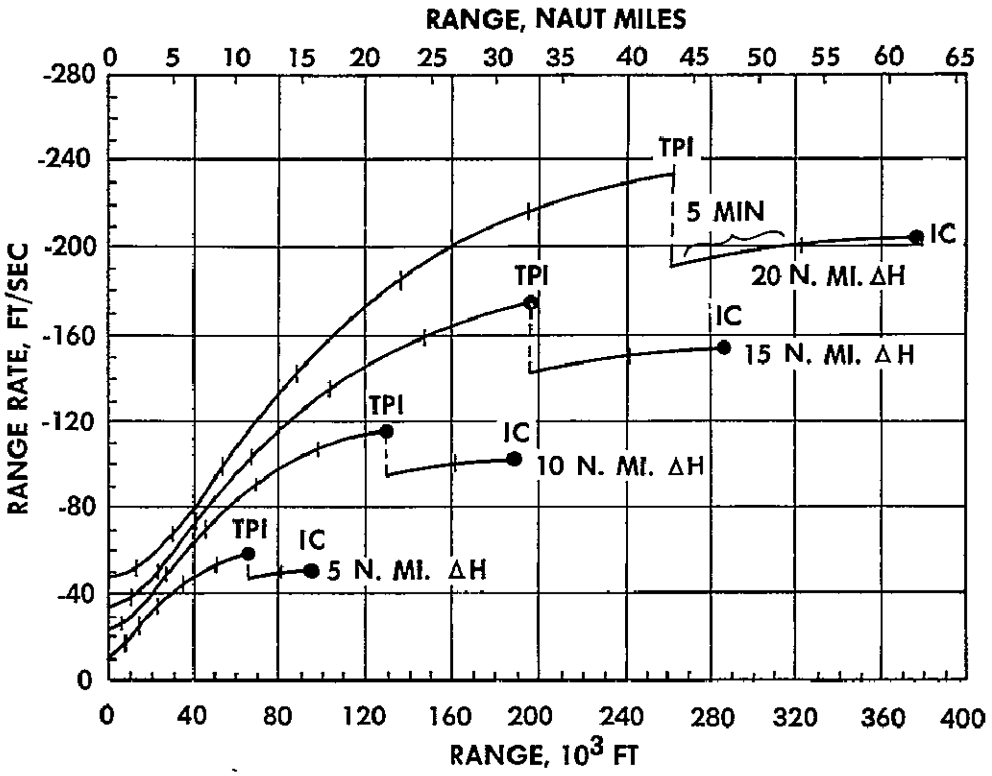


Figure 9c. - Simulated reference trajectories - range rate versus range.

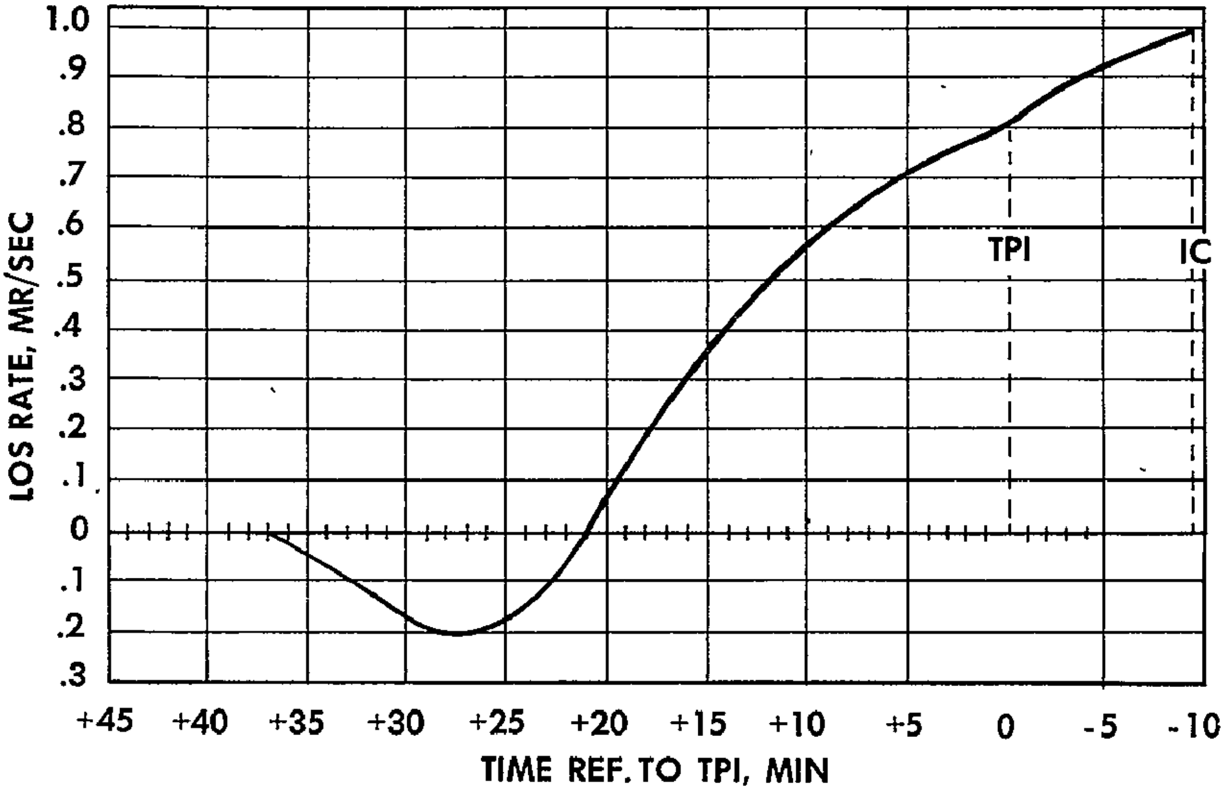


Figure 9d. - Simulated reference trajectories - LOS rate versus time.

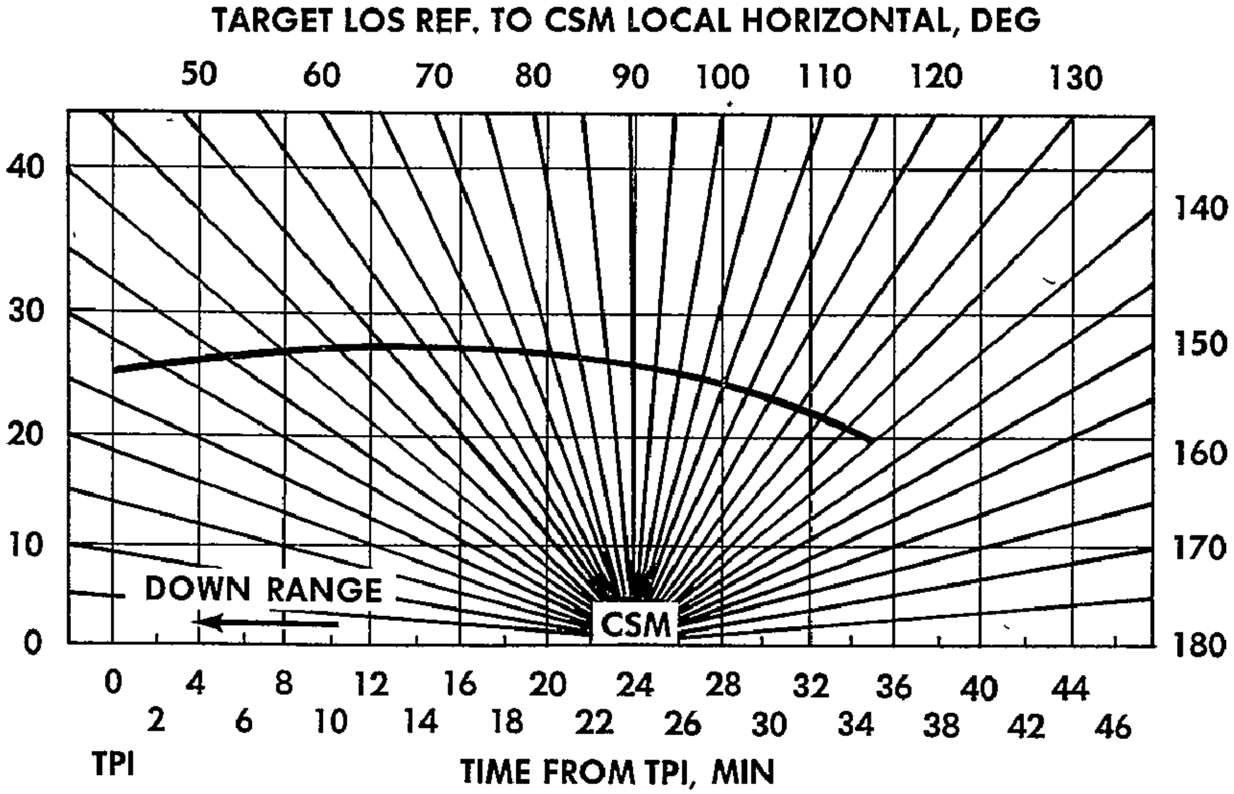
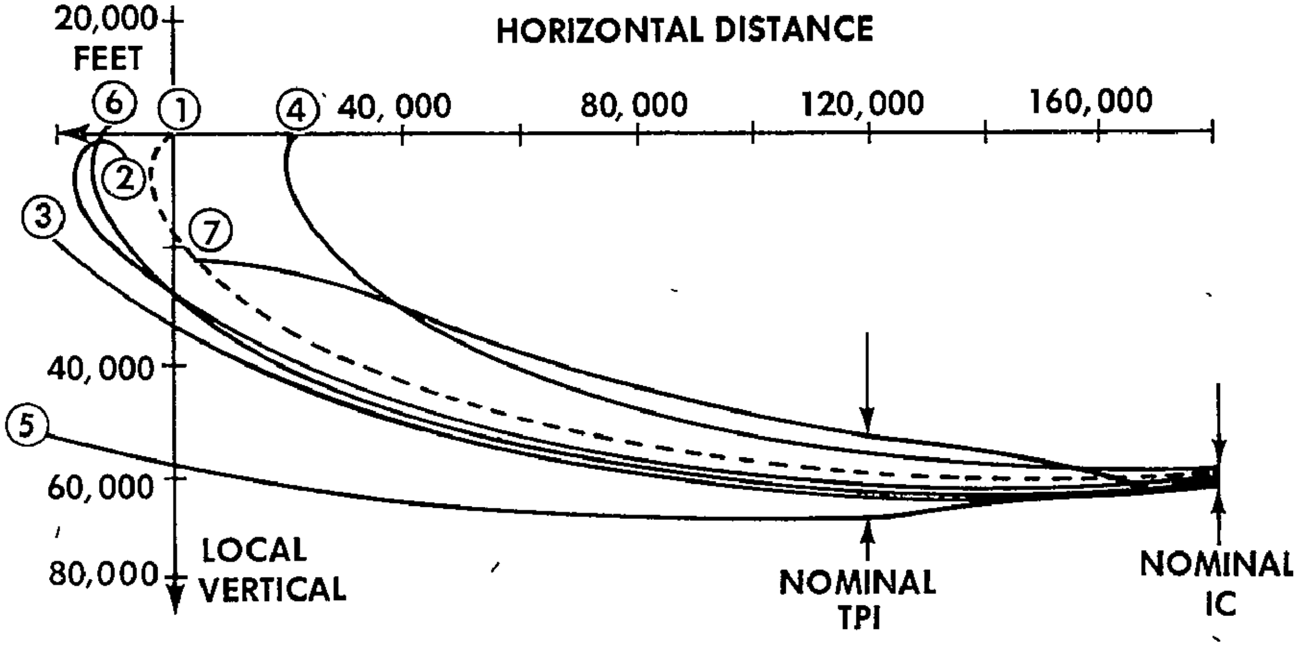


Figure 9e. . Simulated reference trajectories - target LOS versus time.



RECAPITULATION

TRAJECTORY NO.	INITIALIZATION ERRORS
1-10	0 FT / 0 FPS
2-10	650 FT / 1 FPS
3-10	1200 FT / 2 FPS
4-10	2000 FT / 3 FPS
5-10	4000 FT / 6 FPS
6-10	4000 FT / 6 FPS
7-10	9300 FT / 9 FPS

Figure 10. - Nominal and dispersed trajectories for 10-naut-mile Δ H transfer.

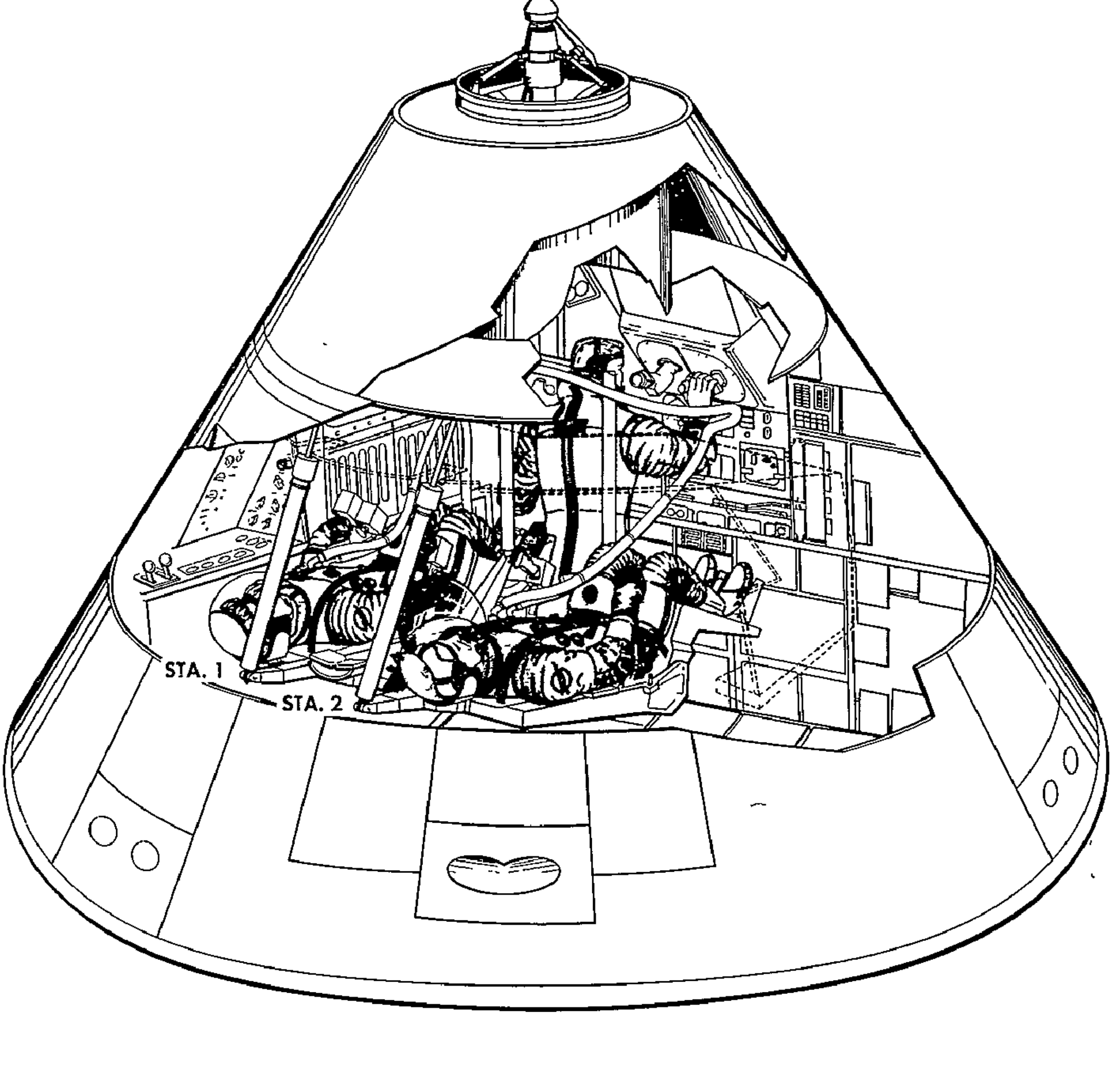


Figure 11a. - Optical sighting

TIME	Θ NOM	R NOM	R NOM	TIME	Θ NOM	R NOM	R NOM
20	70	44	76	27	99	19	45
21	74	39	71	28	104	16	42
22	78	35	65	29	108	14	40
23	82	31	60	30	113	12	38
24	86	28	56	31	118	9	37
25	90	25	51	32	122	7	36
26	95	22	48	33	126	5	36

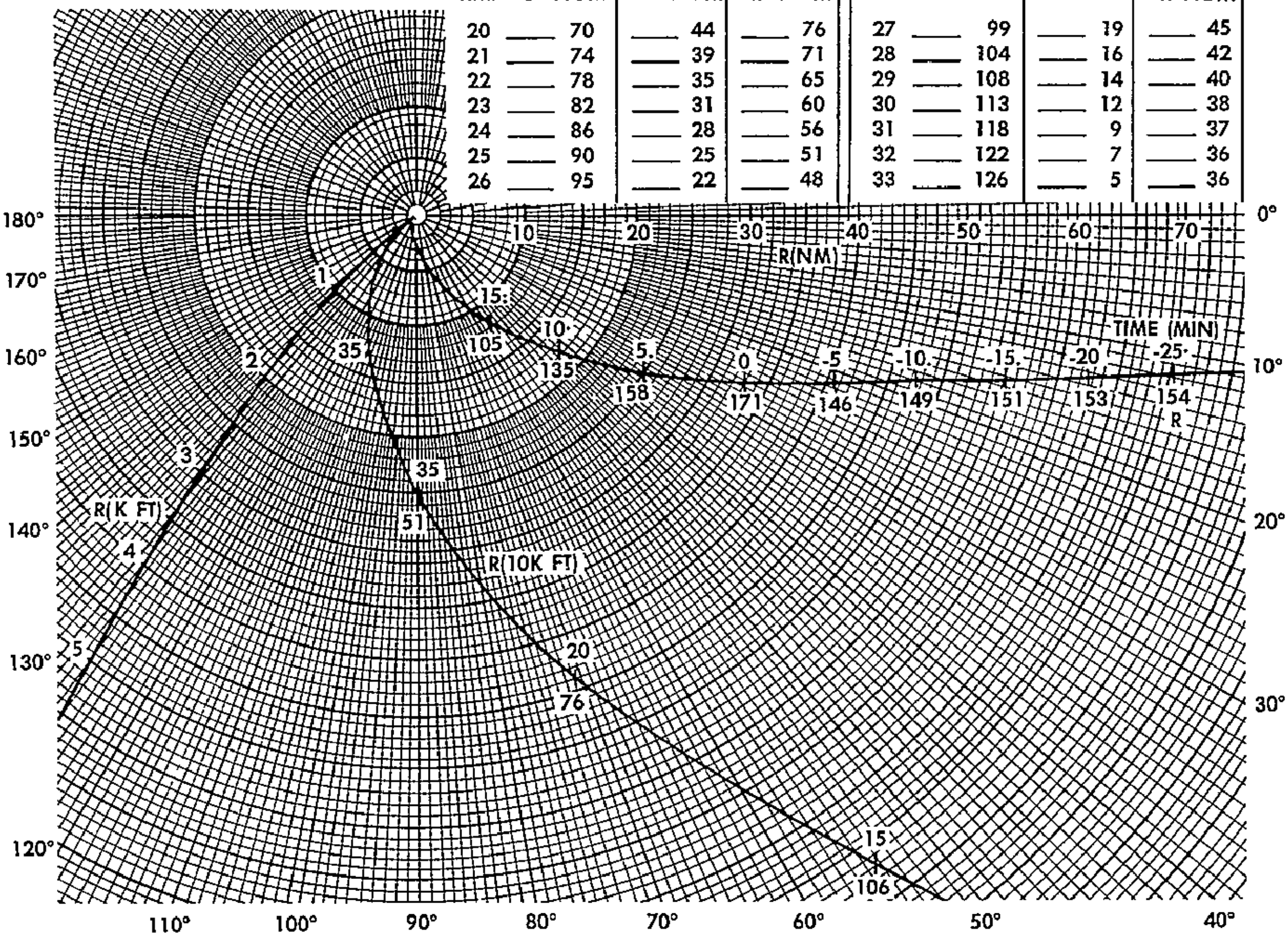
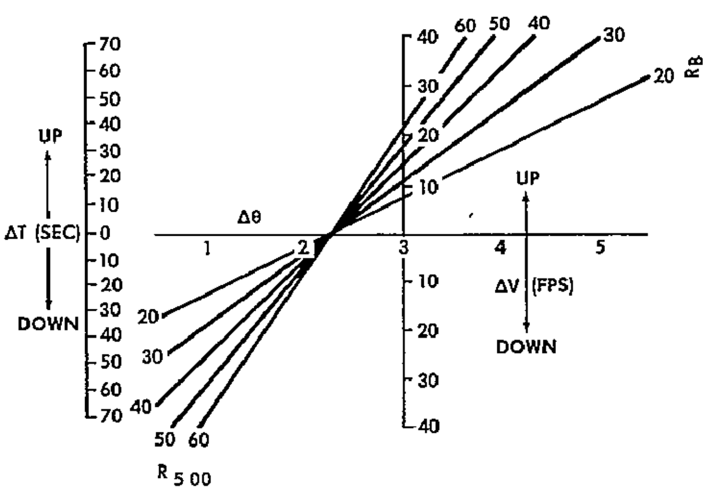


Figure 11b - CSM rendezvous from below relative reference trajectory.

PT	TIME	θ	θ_{NOM}	R	R_{NOM}	R	R_{NOM}
A	-8 00		19 36		44 4		148 2
B	-5 00		-21 63		40 1		146 0
	$-\Delta\theta$		2 27				

TPI 0 00	AXIS	DIR	ΔV	ΔT	USED
	Z	FWD			
	X	U D			



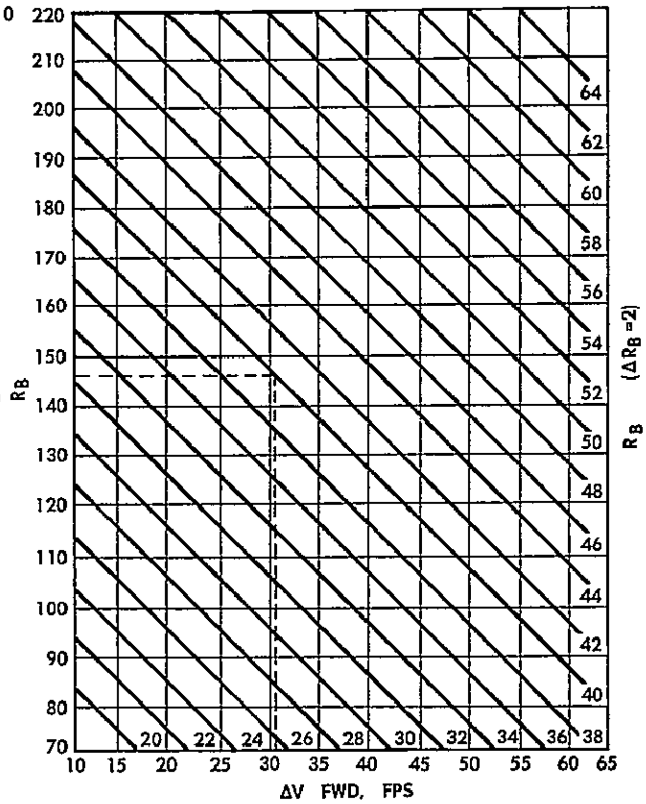
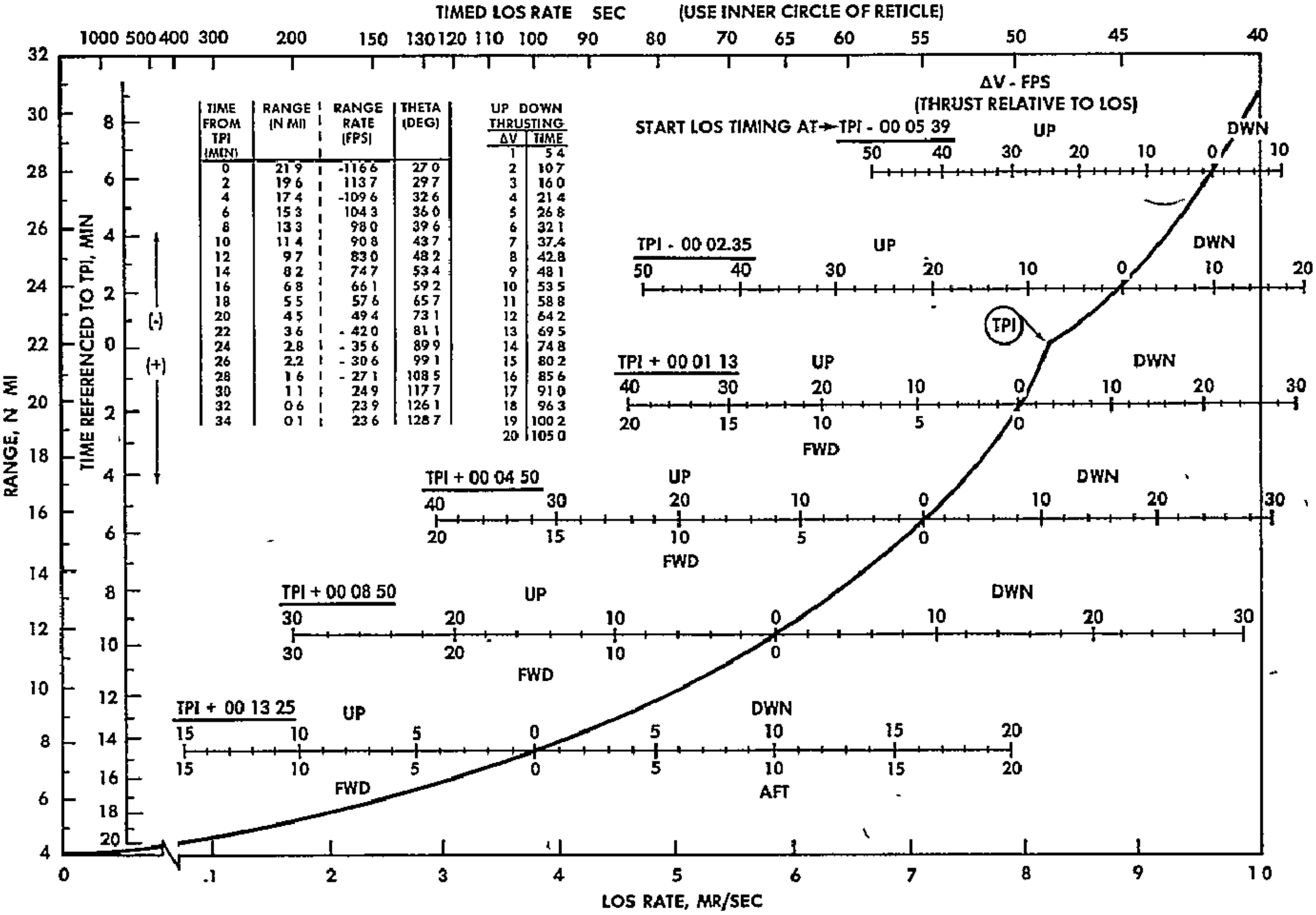


Figure 11c. - Terminal phase initiation (TPI).



TIME FROM TPI (MIN)	RANGE (N MI)	RANGE RATE (FPS)	THETA (DEG)
0	21 9	-116 6	27 0
2	19 6	113 7	29 7
4	17 4	-109 6	32 6
6	15 3	104 3	36 0
8	13 3	98 0	39 6
10	11 4	90 8	43 7
12	9 7	83 0	48 2
14	8 2	74 7	53 4
16	6 8	66 1	59 2
18	5 5	57 6	65 7
20	4 5	49 4	73 1
22	3 6	- 42 0	81 1
24	2.8	- 35 6	89 9
26	2.2	- 30 6	99 1
28	1 6	- 27 1	108 5
30	1 1	24 9	117 7
32	0 6	23 9	126 1
34	0 1	23 6	128 7

UP DOWN THRUSTING ΔV	TIME
1	5 4
2	10 7
3	16 0
4	21 4
5	26 8
6	32 1
7	37.4
8	42.8
9	48 1
10	53 5
11	58 8
12	64 2
13	69 5
14	74 8
15	80 2
16	85 6
17	91 0
18	96 3
19	100 2
20	105 0

Figure 12. - Backup midcourse correction chart.

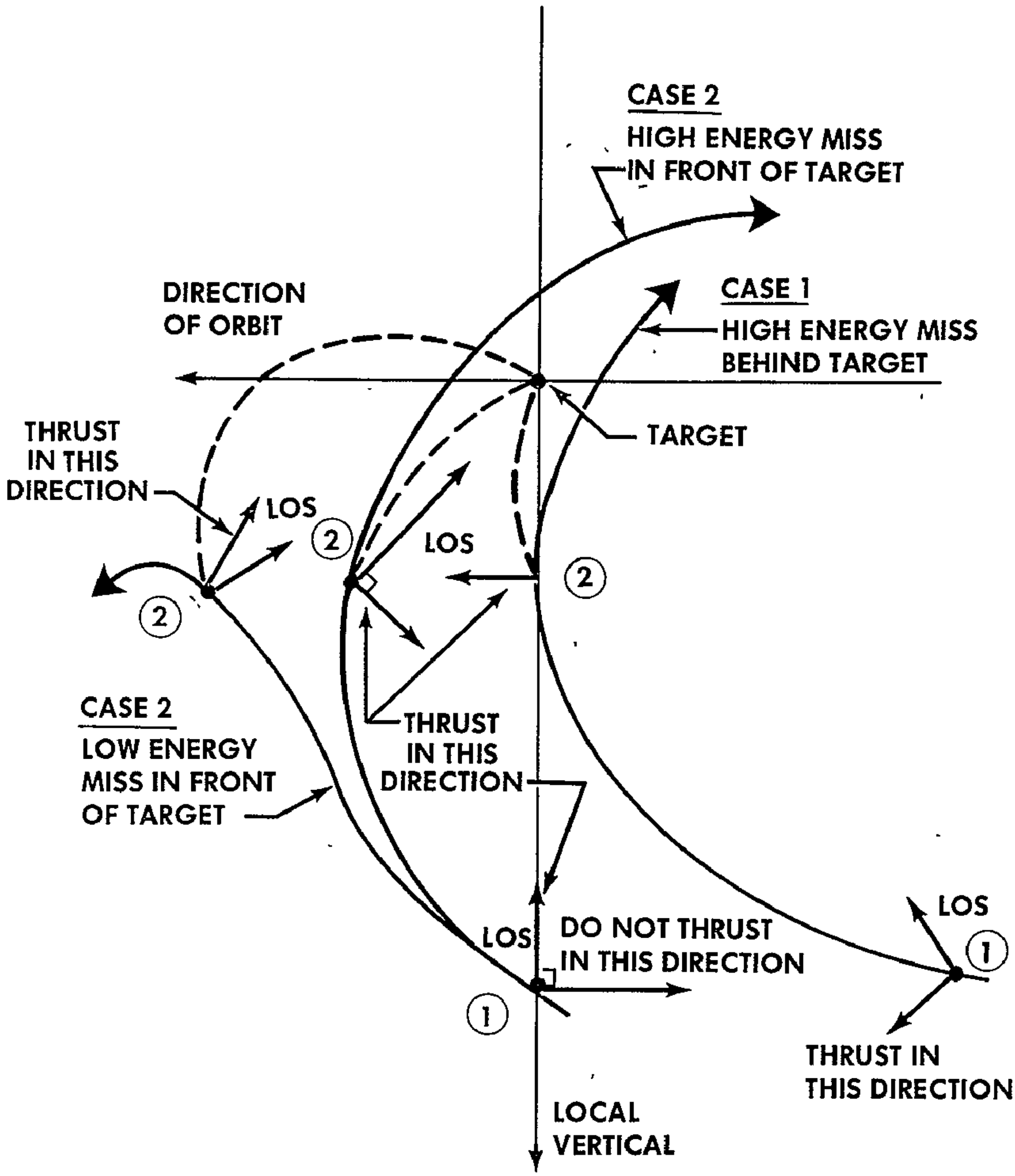


Figure 13. - Terminal LOS rate control.

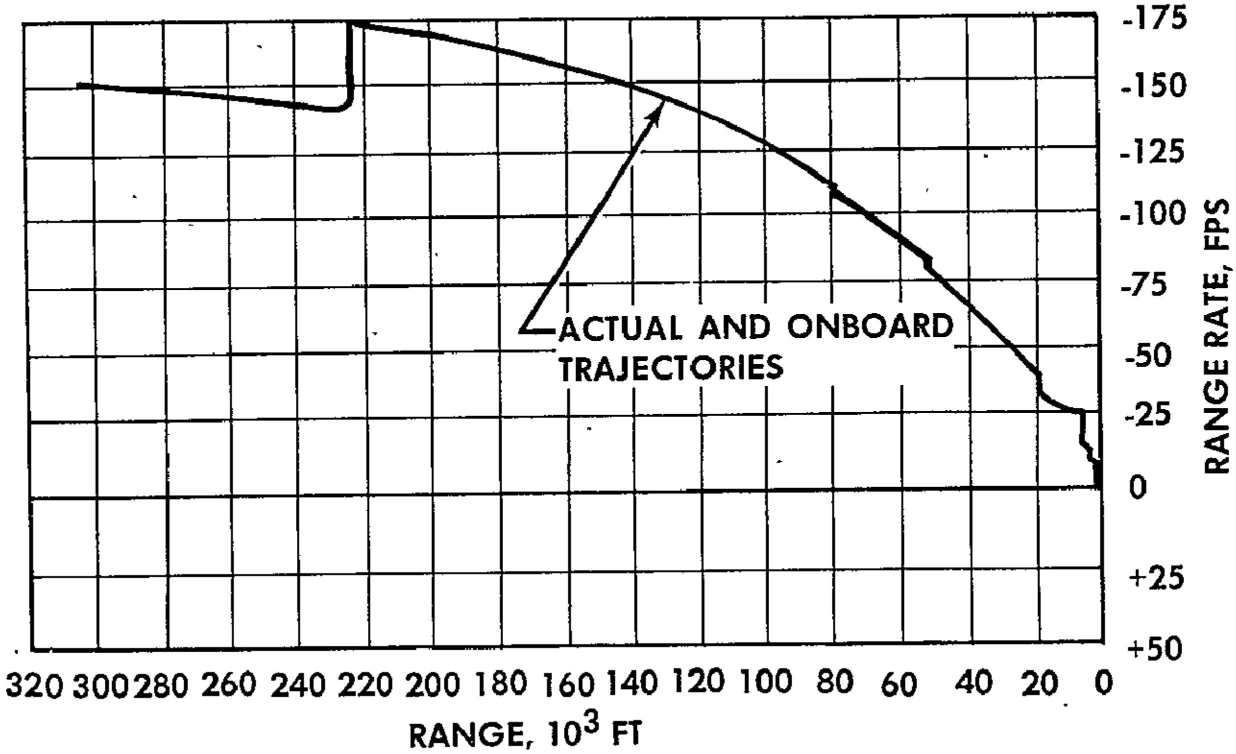


Figure 14a. - Range rate versus range for 15-naut-mile ΔH transfer (no initial errors).

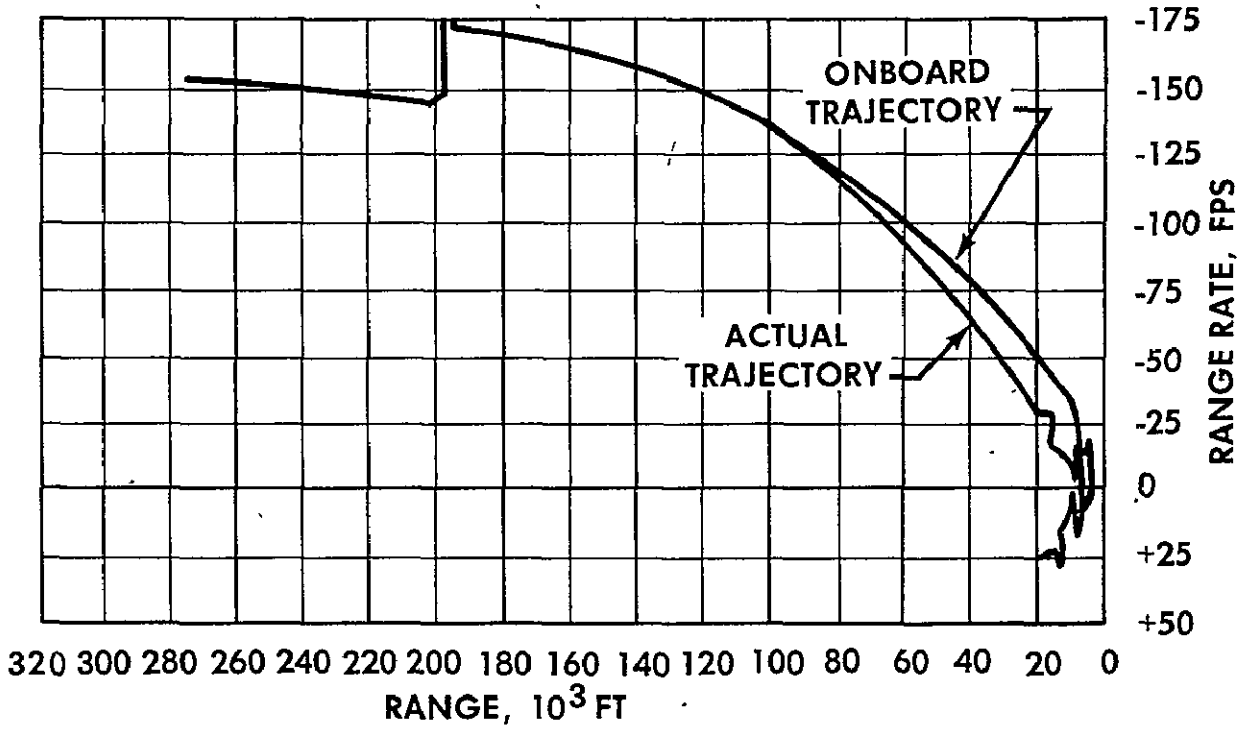


Figure 14b. - Range rate versus range for 15-naut-mile Δ H transfer with initial errors of 650 ft and 1 fps.

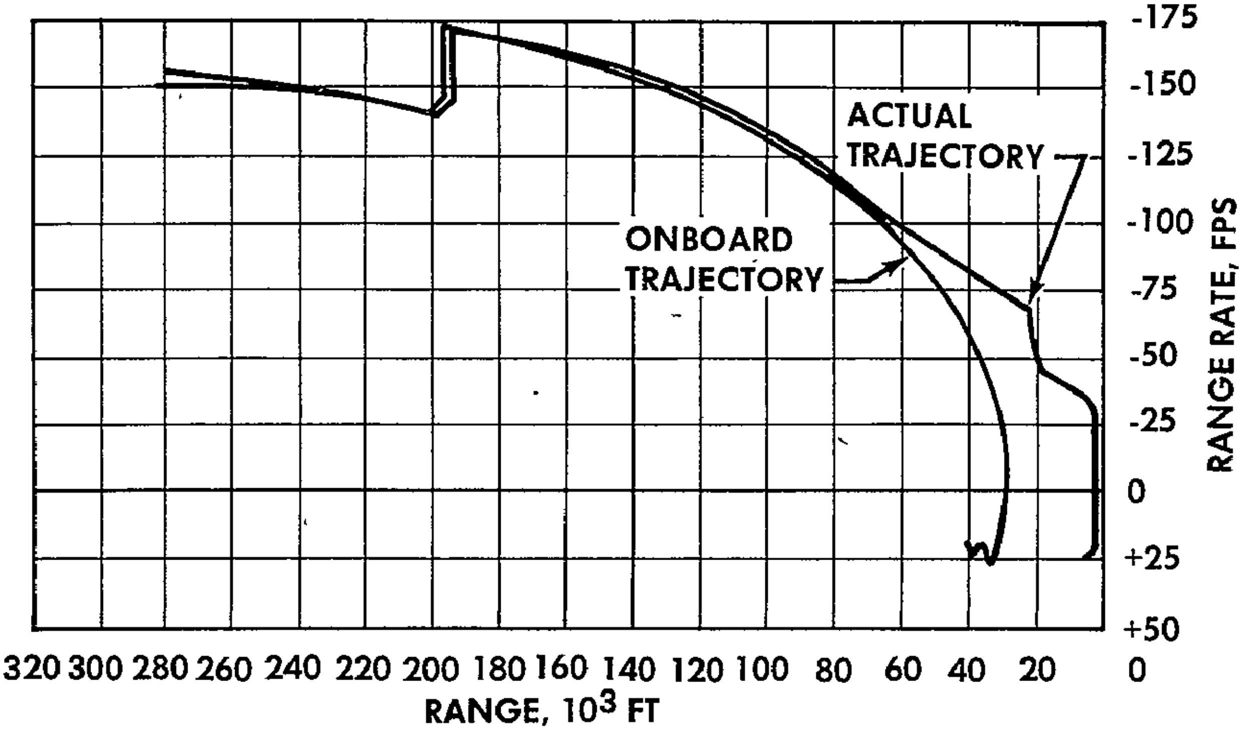


Figure 14c. - Range rate versus range for 15-naut-mile Δ H transfer with initial errors of 2000 ft and 3 fps.

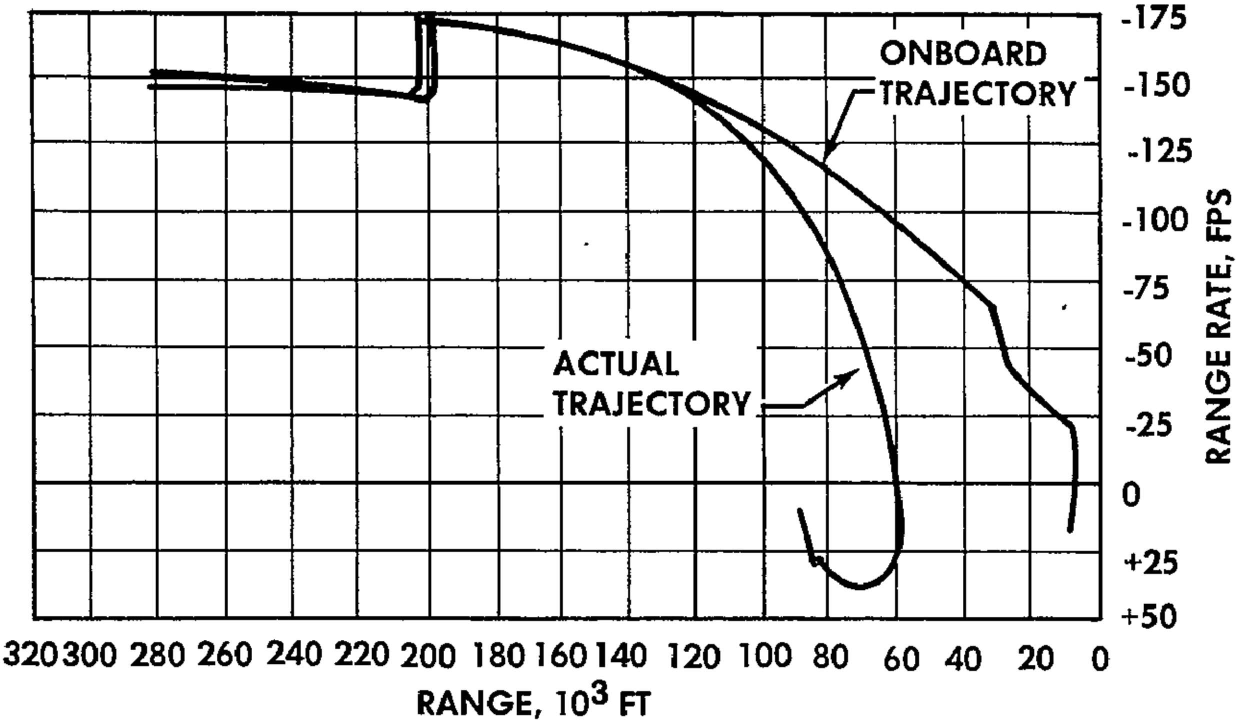


Figure 14d. - Range rate versus range for 15-naut-mile ΔH transfer with initial errors of 4000 ft and 6 fps.

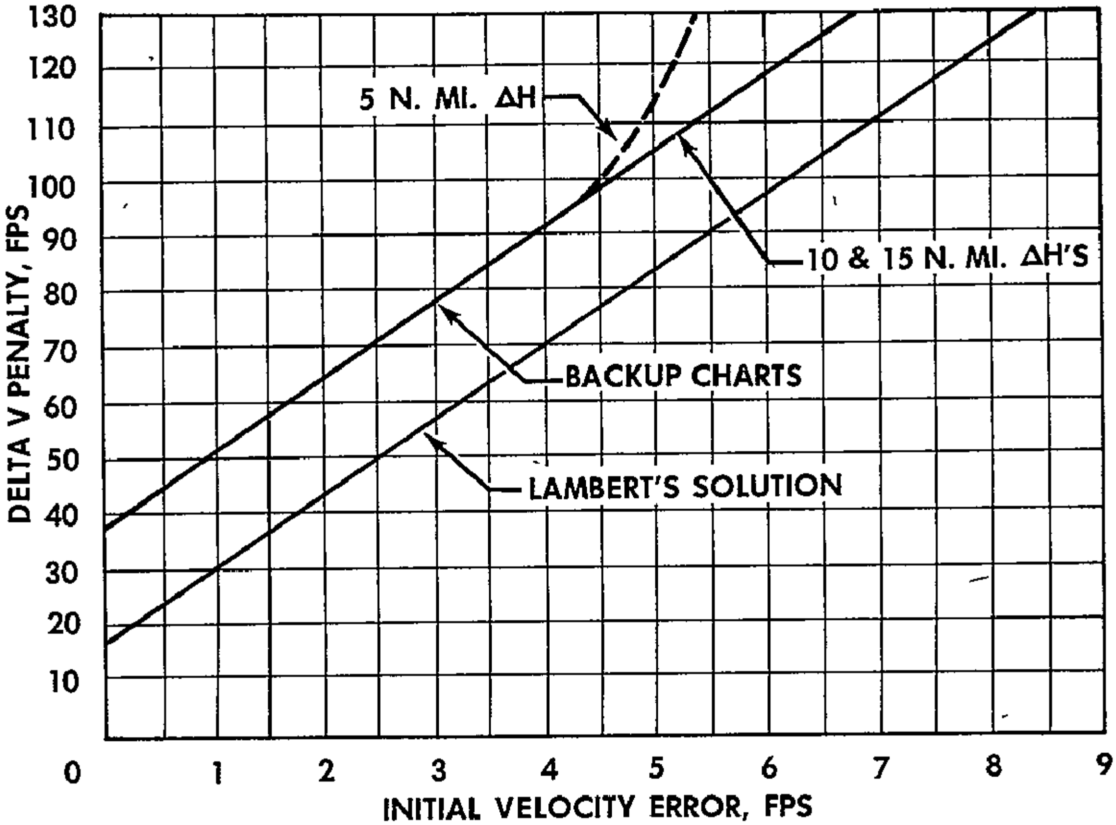


Figure 15. - Delta V penalty required for CSM rescue of the LM.

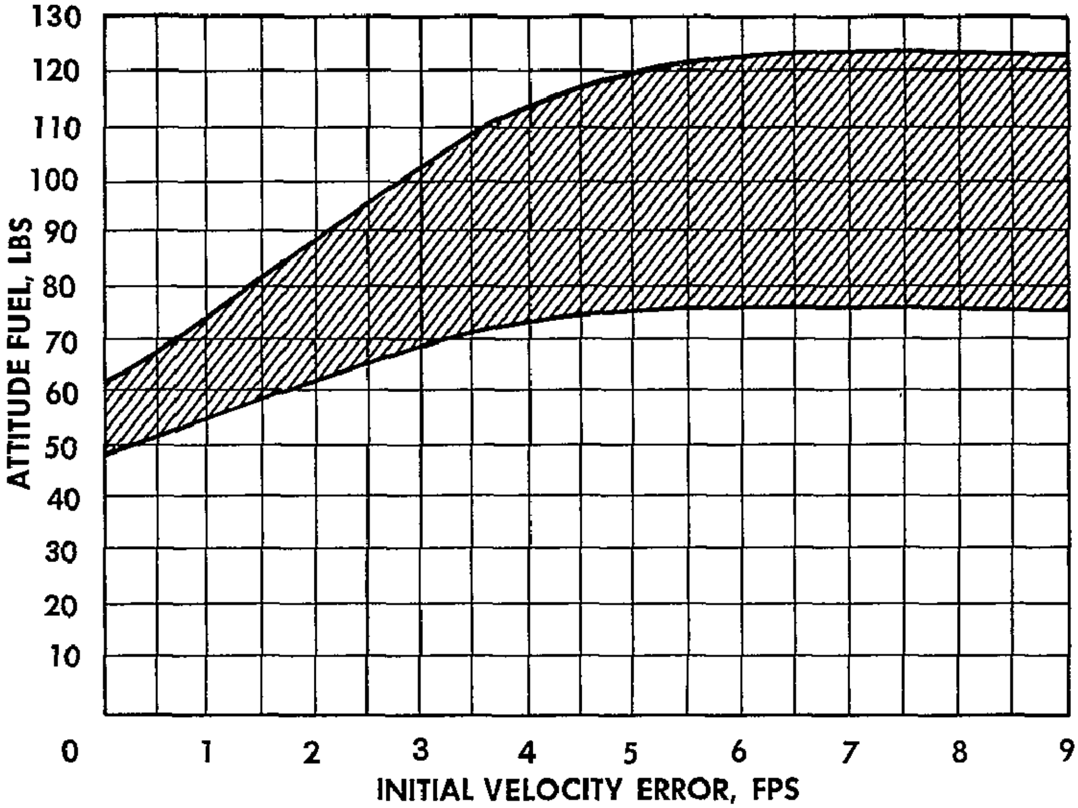


Figure 16. - Attitude fuel required for CSM rescue of the LM.

AS-258

DUAL LAUNCH RENDEZVOUS

SM-RCS REQUIREMENTS	CSM PRIMARY MODE		CSM BACKUP MODE	
ULLAGE - NSR/NCC MANEUVERS	3.8 fps ea.	30 lbs	3.8 fps ea.	30 lbs
ATTITUDE CONTROL (PRE TPI)		30 lbs		30 lbs
TPI MANEUVER	21 fps	84 lbs	5 fps (ULLAGE)	20 lbs
IMPULSIVE TPF MANEUVER	24 fps	96 lbs	24 fps	96 lbs
STA. KEEP./DOCKING	25 fps	100 lbs	25 fps	100 lbs
ATTITUDE CONTROL (POST TPI)		85 lbs		85 lbs
PENALTY FOR 1σ VEL. ERR. (PRIMARY = BACKUP - 3 fps)	56 fps	224 lbs	78 fps	312 lbs
	1σ TOTAL =	649 lbs	1σ TOTAL =	673 lbs
ADDITIONAL PENALTY FOR 3σ VEL. ERR. (PRIMARY = BACKUP = 9 fps)	80 fps	320 lbs	80 fps	320 lbs
	3σ TOTAL =	969 lbs	3σ TOTAL =	993 lbs

Figure 17. - Fuel requirements for AS-258 dual launch rendezvous

AS 258/503 - LM RESCUE

SM-RCS REQUIREMENTS	CSM PRIMARY MODE		CSM BACKUP MODE	
ULLAGE - NSR/NCC MANEUVERS	3.8 fps ea.	30 lbs	3.8 fps ea.	30 lbs
ATTITUDE CONTROL (PRE TPI)		30 lbs		30 lbs
TPI MANEUVER	21 fps	84 lbs	5 fps (ULLAGE)	20 lbs
IMPULSIVE TPF MANEUVER	24 fps	96 lbs	24 fps	96 lbs
STA KEEP/DOCKING	25 fps	100 lbs	25 fps	100 lbs
ATTITUDE CONTROL (POST TPI)		65 lbs		80 lbs
PENALTY FOR 1σ VEL. ERROR (PRIMARY = 1 fps; BACKUP = $2\frac{1}{2}$ fps)	30 fps	120 lbs	71 fps	284 lbs
	1σ TOTAL =	525 lbs	1σ TOTAL =	640 lbs
ADDITIONAL PENALTY FOR 3σ VEL. ERROR (PRIMARY = 3 fps; BACKUP = $7\frac{1}{2}$ fps)	26 fps	104 lbs	65 fps	260 lbs
	3σ TOTAL =	629 lbs	3σ TOTAL =	900 lbs

Figure 18. - Fuel requirements for AS-258/503 CSM rescue of LM

AS 504 - LM RESCUE

SM-RCS REQUIREMENTS	CSM PRIMARY MODE		CSM BACKUP MODE	
ULLAGE - CSI/CDH MANEUVERS	3.8 fps ea	35 lbs	3.8 fps ea	35 lbs
ATTITUDE CONTROL (PRE TPI)		30 lbs		30 lbs
TPI MANEUVER	20 fps	92 lbs	5 fps (ULL)	23 lbs
IMPULSIVE TPF MANEUVER	20 fps	92 lbs	20 fps	92 lbs
STA KEEP/DOCKING	25 fps	115 lbs	25 fps	115 lbs
ATTITUDE CONTROL		65 lbs		70 lbs
PENALTY FOR 1σ VEL. ERR. (PRIMARY = 1 fps; BACKUP = 1.4 fps)	30 fps	138 lbs	58 fps	267 lbs
	1σ TOTAL =	567 lbs	1σ TOTAL =	632 lbs
ADDITIONAL PENALTY FOR 3σ VEL. ERR. (PRIMARY = 3 fps; BACKUP = 4.5)	26 fps	120 lbs	40 fps	202 lbs
	3σ TOTAL =	687 lbs	3σ TOTAL =	834 lbs

NOTE: NO ALLOWANCE MADE FOR OUT-OF-PLANE ERRORS.
CSM WT. = 38,000 lbs

Figure 19. - Fuel requirements for AS-504 CSM rescue of LM

APPENDIX A

EULER ANGLE TRANSFORMATION MATRIX
(θ, ψ, ϕ Rotation)

$$\begin{bmatrix} X_B \\ Y_B \\ Z_B \end{bmatrix} = \begin{bmatrix} B \end{bmatrix} \begin{bmatrix} X_I \\ Y_I \\ Z_I \end{bmatrix}$$

where

$$\begin{bmatrix} B \end{bmatrix} = \begin{bmatrix} \ell_{11} & \ell_{12} & \ell_{13} \\ \ell_{21} & \ell_{22} & \ell_{23} \\ \ell_{31} & \ell_{32} & \ell_{33} \end{bmatrix}$$

$$\begin{bmatrix} F_X \\ F_Y \\ F_Z \end{bmatrix} = \begin{bmatrix} B \end{bmatrix}^{-1} \begin{bmatrix} T_X \\ T_Y \\ T_Z \end{bmatrix}$$

where

$$\begin{bmatrix} B \end{bmatrix}^{-1} = \begin{bmatrix} \ell_{11} & \ell_{21} & \ell_{31} \\ \ell_{12} & \ell_{22} & \ell_{32} \\ \ell_{13} & \ell_{23} & \ell_{33} \end{bmatrix}$$

$$\ell_{11} = \cos\psi \cos\theta$$

$$\ell_{12} = \sin\psi$$

$$\ell_{13} = -\cos\psi \sin\theta$$

$$\ell_{21} = \sin\phi \sin\theta - \cos\phi \sin\psi \cos\theta$$

$$\ell_{22} = \cos\phi \cos\psi$$

$$\ell_{23} = \sin\phi \cos\theta + \cos\phi \sin\psi \sin\theta$$

$$\ell_{31} = \cos\phi \sin\theta + \sin\phi \sin\psi \cos\theta$$

$$\ell_{32} = -\sin\phi \cos\psi$$

$$\ell_{33} = \cos\phi \cos\theta - \sin\phi \sin\psi \sin\theta$$

APPENDIX B

Attitude and Translation Accelerations

$\dot{p} = \pm 8.45\ ^\circ/sec^2$ (4 jets)
$\dot{q} = \pm 1.40\ ^\circ/sec^2$
$\dot{r} = \pm 1.28\ ^\circ/sec^2$

$\ddot{X} = \pm 0.189\ fps^2$ (2 jets)
$\ddot{X} = \pm 0.379\ fps^2$ (4 jets)
$\ddot{X} = + 18.999\ fps^2$ (SPS engine)
$\ddot{Y} = \pm 0.189\ fps^2$
$\ddot{Z} = \pm 0.189\ fps^2$

Minimum Impulse Rate Change

$\Delta p = 0.063\ ^\circ/sec$
$\Delta q = 0.021\ ^\circ/sec$
$\Delta r = 0.019\ ^\circ/sec$

SPS Configuration

SPS thrust location: $X_{sps} = -111.3$ in.
SPS thrust: $T_{sps} = 20{,}000$ lbs.
SPS specific impulse: (see Apollo Mission Data Specification C, AS278A)
Thrust on transport delay: 0.4 second
Tailoff ΔV: 10.3 fps
Tailoff time: 1 second

Characteristics of Simulated CSM

Mass-Inertia Properties

mass (m.): 1052.7117 slugs
c.g. location: $\bar{X} = 0$ in.
$\bar{Y} = 2.28$ in.
$\bar{Z} = 5.76$ in.

moments of inertia: $I_{xx} = 18{,}489$ slug ft^2
$I_{yy} = 55{,}627$ slug ft^2
$I_{zz} = 60{,}940$ slug ft^2

products of inertia: $I_{xy} = -1{,}838$ slug ft^2
$I_{xz} = 523$ slug ft^2
$I_{yz} = 397$ slug ft^2

SM/RCS Jet Configuration

RCS jet plane location: $X_{RCS} = +14.39$ in.
RCS jet thrust: $T_{RCS} = 100$ lbs.
RCS jet specific impulse: (see Apollo Mission Data Specification C, AS278A)

Attitude Control Torques
$M_p = \pm 2727$ ft-lb (4 jets)
$M_q = \pm 1355$ ft-lb
$M_r = \pm 1355$ ft-lb

RCS Jet Logic

$+\epsilon_p$: J_9, J_{11}, J_{13}, J_{15}

$-\epsilon_p$: J_{10}, J_{12}, J_{14}, J_{16}

$+\epsilon_q$: J_1, J_3

$-\epsilon_q$: J_2, J_4

$+\epsilon_r$: J_5, J_7

$-\epsilon_r$: J_6, J_8

$+\delta_x$: J_1, J_2, J_5, J_6

$-\delta_x$: J_3, J_4, J_7, J_8

$+\delta_y$: J_{13}, J_{14}

$-\delta_y$: J_{15}, J_{16}

$+\delta_z$: J_9, J_{10}

$-\delta_z$: J_{11}, J_{12}

TABLE I

Simulated State Vector Errors

Error Case No.	Initialization Error Tracking	Initialization Error Errors	Update Error Tracking	Update Error Errors
1	Perfect	0ft/0fps (nominal	Perfect	0ft/0fps (nominal)
2a	MSFN/SXT	650ft/1fps (correlated)	None	Propagated IC
2b	MSFN/SXT	650ft/1fps (correlated)	SXT	430ft/0.8fps (correlated)
3	MSFN/SXT	1200ft/2fps (correlated)	SXT	1150ft/2.2fps (correlated)
4a	MSFN/SXT	2000ft/3fps (correlated)	None	Propagated IC
4b	MSFN/SXT	2000ft/3fps (correlated)	SXT	460ft/1.2fps (correlated)
5	MSFN	4000ft/6fps (correlated)	None	Propagated IC
6	MSFN	4000ft/6fps (uncorrelated)	None	Propagated IC
7	MSFN	9300ft/9fps (uncorrelated)	None	Propagated IC

TABLE II

Gate No.	Range (N.M.)	Range Rate (FPS)
1	5	50
2	3	35
3	1	15
4	1000 FT	5

H	APPLICABLE GATES
20 N.M.	1, 2, 3, 4
20 N.M.	1, 2, 3, 4
15 N.M.	2, 3, 4
10 N.M.	3, 4
5 N.M.	4