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NO. LED-540-3 DATE: 9 July 1963

BACK-UP GUIDANCE REQUIREMENTS [U] #3

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BACK-UP GUIDANCE

REQUIREMENTS [U] #3

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GRUMMAN AIRCRAFT ENGINEERING CORPORATION

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A. SUMMARY OF CONCLUSIONS AND RECOMMENDATIONS

The study, discussed in the following pages, is aimed at establishing the performance requirements for a back-up guidance system for the LEM. The groundrules under which the study has been performed are:

. the back-up guidance system is to be designed to guide the LEM during nominal ascent from the lunar surface and during abort phases only;

. the performance requirements specified for the system and its components are to be compatible with this limited capability; that is, the equipment characteristics quoted in this report represent the maximum tolerances allowable to perform the limited guidance functions cited;

. system configuration studies are performed only to the extent necessary to verify the feasibility of certain performance requirements. No specific recommendations concerning the hardware implementation of the back-up system are included, although a general discussion of the tradeoffs between platform-mounted and body-mounted inertial components is presented;

. it has been specified that all near moon coasting trajectories must have pericynthion altitudes of greater than 40,000 ft.;

. for the purposes of evaluating the effectiveness of various guidance techniques, it has been assumed that a distance of closest approach of the LEM to the CSM of 10 n. mi. or less would be required on the coasting ascent transfer orbit in order to assure proper initial conditions for the rendezvous phase.

On the basis of these groundrules, the ascent, abort and mid-course phases of the LEM mission have been studied in considerable detail. The results are, however, only preliminary, since considerable additional effort is required to evaluate the proposed guidance concepts and equipment performance tolerances under all possible dynamic situations.

The major conclusions that have been arrived at from the study of the back-up guidance problem thus far are:

. mid-course corrections during ascent coast must be provided to assure rendezvous under the wide range of burnout conditions that can result if ascent or abort is performed on back-up guidance;

. if burn-out of the powered ascent phase occurs at supercircular velocities, the requirements of the mid-course correction phase determine the limiting constraint on the back-up attitude reference. If burn-out occurs at circular orbital velocity, the requirement for clear pericynthion constrains the equipment performance tolerances;

. completely open loop ascent and abort guidance can result in errors at burnout larger than can be handled by the mid-course correction techniques devised for the back-up system. Thrust and weight deviations contribute the major proportion of burnout errors, while gyro drift and initial misalignment represent the most significant equipment errors;

. a comparitively simple ascent guidance technique can be devised to eliminate essentially the errors associated with off-nominal thrust and weight and with thrust vector misalignment. The guidance law corrects a nominal pitch program and the burn-out ΔV as a function of measured accelerations;

. abort from powered descent trajectories can be designed to be similar to nominal ascent trajectories. The same type of pitch program can be utilized, with the parameters of the program being functions of the time of abort. These parameters, three of which are required to specify a trajectory, turn out to be linear or quadratic functions of time;

. a simple mid-course correction technique can be devised, using a perturbation technique in which radar measurements are compared with the nominal values of the measured parameters and the differences, or residuals, combined in linear form to determine the corrective impulse. In-plane and out-of-plane corrections can be combined, provided the corrections are always applied near the line of nodes.

It is recommended that the following procedures be adopted for ascents and aborts on back-up guidance:

. nominal ascent should always take place so that the LEM velocity vector is parallel to the CSM orbit plane at injection. This assures that the line of nodes is 90 degrees central angle down-range from the injection point and allows fixing the time of application of the mid-course corrections regardless of the out-of-plane angle;

. nominal ascents from the lunar surface should always aim for a near Hohmann transfer to the CSM altitude regardless of the out-of-plane angle. This reduces the nominal ΔV requirements and simplifies the guidance scheme;

. direct ascents to supercircular velocity are always preferred since the tolerance of pericynthion altitude to burn-out errors in flight path angle and velocity are reduced. For those cases where stay in a low altitude parking orbit is required, a velocity bias of the burn-out conditions should be applied so that a slightly elliptical rather than circular parking orbit is attained;

. since the mid-course correction technique requires a nominal coasting ascent trajectory as a reference, a minimum of reference trajectories should be used for ascent and abort in order to minimize the number of constants to be stored. It is recommended that in addition to the Hohmann trajectory for ascent from the surface, two other nominal trajectories be designed for abort. One of these, a 267 degree central angle (synchronous) orbit, is used for early aborts, and the other, a 235° central angle orbit, for aborts occurring later in the powered ascent. These orbits were selected on the basis of minimizing the total ΔV for abort and to reduce the maximum stay time in a parking orbit following abort to about one half-hour;

. the nominal burnout conditions for an abort from powered descent should always be those corresponding to a slightly super-circular velocity at 50,000 ft. altitude. Insertion into the nominal ascent trajectories then takes place after a stay time in the low altitude orbit which is a function of the time of abort;

. delays in launching from the lunar surface should be handled by again inserting the LEM into a slightly elliptical, low altitude parking orbit and transferring to the Hohmann ascent orbit when the proper phasing conditions are attained;

. for the worst case of lunar surface launch, the ellipticity of the parking orbit would increase the maximum synodic time for phasing by about an hour. However, it is recommended that the mission design plan not be altered to reflect this increase, since this situation can only occur as a result of a double failure situation; one failure which requires an immediate launch and a concurrent failure of the primary Navigation and Guidance system.

An analysis of the LEM mission success and of crew safety reliability with the primary Navigation and Guidance system, and the resultant reliability requirements upon the back-up guidance system, as well as a configuration analysis of the proposed back-up equipment, are presented in the following documents:

LED-290-3; Enclosure I, "Reliability Configuration Analysis of LEM Guidance."

Enclosure II (GAEC Rpt. no. RC-G323A-1.0), "Techniques for Finding Reliability of Complex Networks."

LED-550-1; "Reliability Configuration Analysis of a Four-Gimbal Platform vs. a Strapped-Down System."

B. STUDY PLAN FOR THE DETERMINATION OF BACK-UP GUIDANCE REQUIREMENTS

Design Concepts

The study of the performance requirements and system configuration of the LEM back-up guidance equipment has been directed primarily towards satisfying crew safety reliability established by NASA for the LEM mission. It has thus been stipulated that the back-up guidance system shall achieve rendezvous with the CSM from any mission phase in the event of a failure in the primary navigation and guidance equipment. In evaluating the crew safety probability attainable with a given implementation of a back-up guidance system, the ability of the CSM to aid the LEM, either by providing guidance data or by performing the rendezvous and docking phases, has been an important consideration.

Throughout the studies described in this report, as well as in the studies performed to define the use of and the requirements for the LEM radar sensors,[1] it has been a fundamental rule that in order to minimize equipment complexity, the design of the back-up guidance equipment and operation shall be compatible with the crew's ability to monitor and perform as many guidance functions as possible.

Once the basic design philosophy and design concepts for the development of the back-up guidance subsystem had been determined, the following study program was undertaken:

1. A comprehensive study of the nominal mission and of combinations of failures of the primary guidance equipment during the various mission phases in order to define the functions of the back-up equipment;

2. An investigation of the implications of the nominal design trajectories upon possible abort procedures and of the feasibility of aborting from any point in the nominal mission;

3. Analytic studies of applicable guidance techniques to accomplish the various guidance functions and of the design, performance and accuracy requirements for the back-up equipment;

4. Detailed hardware studies of various equipment configurations to determine component performance, accuracy, and reliability requirements;

5. Investigation, through piloted simulations, of the flight crew operational requirements and performance capabilities in the back-up modes to determine to what extent the crew can monitor and take over the guidance function with appropriate control aids and instrumentation;

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[1] "Radar Requirements Report", LED-540-1

6. Reliability analysis of the various back-up guidance modes and types of equipment to determine the degree to which mission safety and success are enhanced through the use of the back-up equipment.

The present document represents a preliminary report on the results of these studies. The simulation effort is in the process of being set up and no results are available from their aspect of the study. Further trajectory and guidance investigations are required to firm up the guidance and operational concepts developed thus far and establish in detail their feasibility.

C. BACK-UP GUIDANCE DESIGN CONCEPT

The back-up guidance mode is defined as the mode in which back-up equipment and/or extensive crew participation in the guidance operation are required because of a failure in the primary system. The following system design concepts have been adopted for the back-up guidance mode.

1. Crew safety is the primary design consideration.

   This concept leads to the requirement for abort capability from any point in the mission. However, it is possible that mission safety can be increased under specific circumstances by continuing the mission to a lunar landing despite a failure in the primary guidance system, and thus the back-up guidance system should be designed to perform this function where feasible.

2. The back-up guidance system must be simpler and more reliable than the primary system.

   This requirement ensures that the reliability goals for crew safety be met without paying a large weight penalty for carrying standby or redundant equipment that is fully as complex as the primary system. With respect to the AGC, (the Apollo Guidance Computer), there is no intention of duplicating its functions in the back-up equipment, but rather to accomplish the abort function with minimum equipment complexity. Radar sensors will be designed so that one can provide back-up for the other, rather than include a standby radar on the LEM specifically for an altimeter or a rendezvous back-up function.

3. Manual modes will be employed where their incorporation increases crew safety.

   The use of the crew's capabilities wherever possible to increase the probability of mission safety and success is the major design goal. Even when the primary guidance system is operating properly, the judgment and versatility of the human operator will be most helpful during final descent, touchdown on the lunar surface and during rendezvous of the LEM with the CSM. The back-up guidance system will be designed to permit crew participation to the maximum extent commensurate with safety goals.

4. There will be a direct display of sensor data where possible.

   A direct display of information from such sensors as the rendezvous and altimeter radar data, the IMU gimbal angles, and body-mounted rate gyros will permit this information to be monitored

and used by the crew before degradation or loss through failure of any processing equipment (such as the primary guidance computer). The concept of crew operations during descent and landing assumes that with directly displayed data from the altimeter radar (between 20,000 and 15,000 ft. altitude good altimeter data becomes available), the crew can take over attitude and thrust control of the LEM and perform a safe terminal descent and landing within the ΔV limitations of the descent stage. Similarly, with directly displayed data from the rendezvous radar, the terminal homing technique recommended by GAEC for the rendezvous guidance phase can be performed by the crew without the use of the primary guidance computer.

5. The automatic guidance modes should be compatible with manual monitoring and override when the alternate manual modes are available.

   This requirement is necessary to ensure that the maneuvering and control commands generated by the primary guidance system are compatible with the commands that would be necessary if generated in the takeover from the automatic mode. If required either by the crew or by failure of some primary equipment, the transition will then be relatively smooth and continuous.

D. MISSION ANALYSIS

The Primary System

The guidance functions provided throughout the mission by the primary system require the following major elements:

1. IMU

   The Inertial Measurement Unit is a three gimbal, gyro stabilized platform mounting three integrating accelerometers. LEM orientation and acceleration information generated in the IMU is sent to the other elements of the primary guidance system for processing and vehicle control.

2. AGC

   The Apollo Guidance Computer accepts the acceleration measurements from the IMU, computes LEM velocity and position, and generates steering attitude commands and velocity corrections.

3. CDU

   The Coupling and Display Unit accepts the steering attitude commands from the AGC and sums them with present attitude from the IMU gimbal angle transmitters. The outputs of the CDU's are steering attitude error signals which provide commands to the SCS (Stabilization and Control System).

4. OMU

   The Optical Measuring Unit in the LEM will consist of either a Scanning Telescope (SCT) or an Alignment Optical Telescope (AOT). The former is a two-degree-of-freedom articulated optical device with automatic gimbal readout. The latter may be a fixed or a single-degree-of-freedom articulated telescope with a movable reticle. The AOT would provide only manual readout of reticle position.

5. Rendezvous Radar

   The rendezvous radar provides outputs of range, range rate, angles, and angular rates with respect to the CSM which are displayed and transmitted to the AGC for processing.

6. Altimeter Radar

   The altimeter radar (a three-beam Doppler configuration) provides altitude and velocity components in body coordinates to both the displays and the AGC.

The failure of any combination of these major functional elements during the mission will require back-up equipment dependent upon the mission phase and the guidance technique employed. The means of determining the operational status of the primary system will be discussed in detail in the section on crew operational requirements.

Back-up Requirements by Mission Phase[2]

a. Synchronous Coast Phase

This phase starts with the transfer of the crew from the CSM to LEM, and the check-out of all systems. The reaction jet system will be exercised, and the two vehicles will be separated by about 100 feet to permit check-out of the radar transmitters and beacons in both the LEM and CSM. The back-up system will be aligned and checked. If a primary system failure is detected prior to injection into synchronous orbit and an abort decision is made, the docking maneuver can readily be accomplished by observation of the CSM, and by the use of either the back-up attitude reference for an attitude hold mode, or the body mounted rate gyros of the SCS (Stabilization and Control System) in the rate command mode.

b. Injection into Synchronous Orbit

The LEM is oriented so that the main engine is pointed roughly along a radius vector away from the moon, and, at a precomputed time, the engine is fired for a short period to obtain transfer into the synchronous orbit (Figure B-1). The back-up attitude reference is used before application of thrust to monitor the attitude change and serves to indicate primary system operational status.

c. Synchronous Coast to Pericynthion

After injection, the LEM will perform radar and optical measurements with respect to the CSM. These measurements can be used to confirm the nominal descent orbit and to detect errors, if any. If the primary system fails subsequent to injection, the abort procedure consists of continuing the coast through to rendezvous. If significant errors exist due to primary system failure during the thrust period, corrective action must be based upon the magnitude of the errors. The midcourse guidance and rendezvous procedures, described later in this section can, in general, be used to account for a fairly wide range of off-nominal conditions. These procedures will have to be adopted as early as possible during the coasting orbit, if a grossly off-nominal condition exists. For example, if measurements indicate a significantly lower pericynthion than nominal, corrective action must be taken immediately.

[2] "LEM Basic Design Mission", LMO-540-59

About 10 minutes prior to reaching pericynthion (approximately 70,000 feet altitude), the altimeter radar sensor will be checked against the lunar surface. A three beam Doppler radar, as is being considered for the altimeter, will provide altitude as well as velocity in body coordinates. It is not expected that high quality information can be obtained at these extreme altitudes but even a gross check on performance will be worthwhile.

d. Powered Descent to 20,000 Ft.

Powered descent is initiated at pericynthion (50,000 ft.) of the synchronous transfer orbit. During the early portions of the descent, data for the guidance and control system are derived from the inertial sensor portion of the primary system. For any reasonable descent trajectory, the pitch attitude of the LEM remains within ±10 degrees of the horizontal for approximately the first 200 seconds of descent. Although accurate thrust vector control is required throughout this phase, a degree of freedom (about the thrust axis) remains. The LEM vehicle may thus be rolled about the thrust axis to orient the forward-looking windows so that they face either the stars or the lunar surface; the latter has the advantage of providing a view of lunar landmarks which may serve to determine visually if gross errors in the plane of the descent trajectory exist and to provide a gross check on range-to-go. The rendezvous radar can track the CSM in either orientation. The LEM and CSM radars, visual observations of landmarks, and the back-up attitude reference can monitor the performance of the primary system.

In the event an abort is required because of a failure of either or both the IMU and AGC, guidance commands must be immediately provided by the back-up system to initiate the appropriate abort trajectory. (Both the abort trajectories and the guidance techniques will be discussed in detail in the following sections.) In general, the abort guidance procedure establishes a simple, constant thrust, pitch angle profile which is basically the same for all abort situations but the specific parameters of which are a function of the initial conditions at the time of abort.

This report defines the back-up equipment necessary for the abort function. Later simulation studies will investigate the information display, equipment, and control procedures needed to continue the mission in a back-up mode.

e. Powered Descent From 20,000 Ft. to Hover

The inertially observed position and velocity data during descent are subject to various errors; errors in initial conditions due to inadequate knowledge of the lunar geoid, cumulative time varying errors due to misalignment and drift and, finally, terminal errors because of terrain variations.

For these reasons, direct sensing of LEM position and velocity through the altimeter radar is necessary for successful descent and landing. The radar sensor updates the computed knowledge of LEM position and velocity relative to the surface and, in combination with the inertial sensors, optimizes the knowledge of trajectory and dynamic control parameters. As the quality of the altimeter radar data improves with decreasing altitude, the relative weighting of the inertial and non-inertial data in the AGC begins to favor the radar input until between 20,000 and 15,000 ft., the altitude data are used exclusively. Updating of altitude rate information is similarly initiated between 7,000 and 5,000 ft. If a radar beacon has been placed at the landing site, the rendezvous radar will search for and lock onto the beacon during this phase and provide direction and slant range information to the AGC for further updating. In the absence of a beacon, the scanning telescope (SCT) or (AOT) will permit a visual check on the quality (with respect to surface obstacles and irregularities) of the computer selected landing site. Initially, SCT or AOT viewing of the landing site will provide only gross information which probably will be insufficient to decide upon a change in landing site. It is expected that not until the range-to-go approaches 10 n. mi. (at an approximate altitude of 10,000 ft.) does the visual information on the landing site become useful. Therefore, it is not necessary that the LEM orientation about the thrust axis be altered to permit viewing the landing site until shortly before this altitude is reached. If, on the basis of visual observation of the site, it is decided to change from the intended landing site to a more favorable location, the crew can use the optical equipment to enter the new location into the AGC which will automatically modify the descent trajectory accordingly. The rendezvous radar, while not expressly designed for the purpose, will have a limited capability for measuring range to the lunar surface at lower altitudes.

The performance of the primary guidance system may be monitored during this phase by comparing inertially computed altitude and velocities with those measured by the altimeter radar and those predicted for the nominal trajectory. The further check provided using the rendezvous radar in an altimeter mode permits additional means of deciding between the altimeter and the IMU/AGC if they should display significantly different values of altitude and velocity. The back up attitude reference can monitor the IMU attitude outputs.

Even if either the IMU or the AGC, or both, fail, the altimeter radar will still provide velocity and altitude information and display them directly to the crew. Below 15,000 ft., depending upon the circumstances of failure and the confidence the crew has in the altimeter data, the mission may be continued in the back-up mode making use of the altimeter and back-up attitude reference data. Sufficient study and simulation has not yet been undertaken to indicate if LEM velocity data relative to body coordinates are adequate to permit successful landing or if resolution into local vertical and horizontal coordinates will be required. The accuracy within which a particular landing site can be selected will be considerably reduced without the use of the primary system, but the SCT can still inspect the estimated landing area and if in the judgment of the crew the area is suitable, descent can continue to the hover phase.

If the altimeter radar alone should fail subsequent to updating

the primary system, the descent may continue with the inertially computed position and velocity data, provided that analysis of the propagation of errors shows that their magnitudes remain within acceptable limits.

f. Hover and Letdown

This phase extends from hover altitude (approximately 1,000 ft.) to touchdown. With the primary system and altimeter radar working properly, the crew will have the option of landing automatically with modification of the touchdown point and descent rate on a momentary override basis or to take over the thrust and attitude control of the vehicle and perform the final phase with semi-automatic or manual control using appropriate displays. The major guidance function in this phase is performed by the crew in either mode of operation in the sense of determining the final landing site. For this reason, the crew must have the ability to scan the intended landing site prior to hover and to translate the LEM over a range wide enough to ensure a high probability of finding a suitable landing site. Descent from hover and translation of the LEM over to a new landing site can be done with altitude and velocity information from either the inertial portion of the primary system, the altimeter radar, or, possibly, the rendezvous radar.

In the event of a failure in the primary system or in the altimeter radar during or just prior to the hover phase, mission safety may be increased if the landing is completed rather than aborted. At some point prior to touchdown, (depending upon the fuel remaining in the descent stage) the decision to abort will require staging of the descent engine and starting of the ascent engine. The loss of altitude during this operation, with an adequate safety margin applied, will determine the minimum altitude at which abort is still feasible. The back-up attitude reference, in conjunction with radar and visual data, will be used to land in the back-up mode.

g. Pre-Launch

On the lunar surface, the back-up attitude reference must be aligned and checked out as part of the pre-launch procedure. In addition, it is necessary to establish LEM location with respect to the CSM orbit and the location, in time, of the CSM in its orbit in order to determine the initial conditions for the ascent trajectory. The SCT and rendezvous radar on the LEM can both be used to track the CSM during the portion of its orbit above the LEM's horizon; in addition, the CSM can communicate its orbital parameters and relative position to the LEM. Since the guidance equipment will be turned off during the lunar stay, LEM touchdown position and attitude must be stored in either a non-destructive portion of the equipment, manually recorded from the displays before shutdown, or recomputed upon equipment activation. It is assumed that a precision clock is kept operating so that the CSM position, as a function of time, may be determined whenever necessary.

In case of failure of primary system on the lunar surface, the back-up attitude reference must be aligned and CSM-LEM relationships determined prior to ascent. As shown, the LEM location relative to the CSM orbit can be determined by the use of the rendezvous radar, SCT, or from the CSM via the communications link. The back-up system can be aligned to a local vertical (as in the case of the primary system) by appropriate processing of back-up accelerometer outputs. Azimuth alignment will require use of star sightings or CSM tracking.

h. Powered Ascent

For nominal ascent the LEM follows a pre-programmed constant thrust attitude profile which is initiated when the proper LEM-CSM phase angle is attained. The rendezvous radar, the SCT, or the precision clock can monitor the time of thrust initiation. During the thrusting period, the back-up attitude reference and accelerometers may monitor the performance of the primary system by comparing the LEM attitude and integrated acceleration with the nominal expected values. In the event of ascent with the back-up system only, the identical attitude profile is commanded. The attitude reference, programmer and accelerometer will provide the guidance commands necessary to achieve the ascent trajectory. In the back-up mode, the line of sight to the CSM can serve a monitoring function with respect to back-up system attitude reference performance by comparing measured radar gimbal angles and angular rates of the LOS with respect to LEM body axes with the nominal values of these parameters.

If the primary system fails during the powered ascent phase, the crew must immediately revert to back-up system operation to avoid the buildup of catastrophic errors in the trajectory. For this reason, the guidance steering commands for nominal ascent must be simple and practically insensitive to transients and small errors in thrust vector control. Assuming that the decision to switch from primary to back-up guidance can be made rapidly, the latter must be able to take over the ascent guidance function instantly during this phase.

i. Ascent Coast and Midcourse Correction

If LEM has been launched at the proper time, the initial thrusting phase will inject LEM into the proper transfer orbit. However, if there is a launch delay or if an emergency takeoff is required because of poor LEM/CSM phase relationship, the LEM must enter a parking orbit and await the proper injection conditions for transfer to the CSM. These conditions can be ascertained from data obtained either by the LEM tracking the CSM with the rendezvous radar or the SCT or by the CSM performing this function and transmitting the data to the LEM. The ascent coast phase carries the LEM from burnout altitude of powered ascent to within homing rendezvous range of the CSM. If the primary guidance system is functioning properly, the midcourse correction (which makes use of LOS angle and range rate information from the rendezvous radar) may or may not be essential, depending upon the magnitude of errors accumulated during the ascent. The effect of midcourse corrections

will then be reduction of ΔV penalties during homing rendezvous due to ascent burnout errors.

If the primary system (IMU or AGC) fails so that all or a portion of the ascent has been carried out in the back-up guidance mode, the resulting burnout errors can be quite large. A capability of performing midcourse corrections in the back-up mode is, therefore, a necessity. The guidance techniques studied for application in this phase are described in considerable detail in Section F. The rendezvous radar of the LEM plus appropriate processing equipment can serve the basic function of measuring the relative positions and velocities of LEM and CSM. Since the coasting ascent period is relatively long, there is sufficient time to use tables for manually determining midcourse correction commands. The CSM radar can also track the LEM during this phase and via the communications link, check these computations or supply a primary determination of them, if necessary. The SCT on the LEM can be used for monitoring the performance of the rendezvous radar or as a prime source of information if both radars fail. The back-up attitude reference will provide the necessary coordinate reference for computing midcourse corrective thrusts. Several midcourse corrections may be required to ensure that the terminal or homing rendezvous phase can be accomplished within the remaining ΔV capability of LEM.

Whether the powered ascent in the back-up mode is undertaken from either the lunar surface or from an abort of the powered descent, the nature of the failures may have been such that only a clear pericynthion orbit can be achieved. Once the crew is assured of the safety of this low altitude orbit (through the use of available sensors, or sufficiently accurate tracking by the CSM), the completion of the mission will depend upon either repair of the necessary equipment or a rescue of the LEM by the CSM. The degree to which the LEM can cooperate in this rescue will vary greatly depending upon the control and guidance capability remaining. A completely passive LEM will require that the CSM compute the necessary descent trajectory to rendezvous with the LEM, and perform the rendezvous and docking maneuvers.

j. Rendezvous

The rendezvous portion of the mission begins when the ascent trajectory of the LEM (after suitable midcourse corrections) has brought the LEM to within approximately 20 n.m. of the CSM. A homing guidance technique is recommended as the basic rendezvous guidance mode primarily to allow the design of alternate and backup modes which provide for maximum crew participation in the operational procedure. It is desirable that the fully automatic primary system perform the homing rendezvous in exactly the same operational steps as the crew would perform in the alternate manual and back-up modes. The crew can then monitor the progress of the rendezvous phase and determine if it proceeds properly; the dynamic conditions existing at any time are suitable for change-over to the alternate or back-up mode so that the crew is prepared to take over the operation and continue it to a successful conclusion without requiring any psychological reorientation, and the crew need only be trained for one basic rendezvous maneuver. A direct display of radar gimbal angles and rates as well as range and range rate will make monitoring possible for the crew during primary system

operation and provide basic data for guidance in the back-up mode. The CSM radar can also be used for either monitoring or for providing guidance data to the LEM via the communications link if the LEM radar fails. If the AGC or IMU fail, the LEM radar can be used for rendezvous. If additionally the LEM radar fails, the CSM radar and the LEM SCT can accomplish the rendezvous.

Summary of Mission Guidance and Control Requirements

Table D-1 presents a summary of guidance and control requirements by phase giving a brief description of the task, constraints, initial and terminal conditions based upon the known parameters - docking and touchdown design conditions, and orbital relationships of both LEM and CSM.

TABLE D-1

GUIDANCE AND CONTROL MISSION REQUIREMENTS

MISSION PHASE FUNCTIONAL REQUIREMENTS	DESCRIPTION OF TASK	CONSTRAINTS	INITIAL CONDITIONS	TERMINAL CONDITIONS
1. Synchronous Coast a) Separation b) Injection c) Synchronous Coast	. Perform docking maneuver - range, range rate, attitude control . Perform orbit adaptation maneuver - thrust vector control . Perform rendezvous terminal homing maneuver (see Mission Phase Functional Rqmts. Item 8)	. Dependent upon situation at failure of primary system		Misalign.....1 ft. Radial, Impact Vel.....2-0 fps. Ang. Vel.....5-0 °/sec. Pitch, Yaw, Roll.....± 10°
2. *Powered Descent to 20,000 ft.	. Fly nominal altitude, altitude-rate, range profile to 20,000 ft. . Observe and correct out-of-plane errors ("Belly-Down" attitude around roll axis to observe landmarks)	. ΔV budget as below	. Nominal 50,000 ft. synchronous orbit . 1.8 deg. central angle LEM leading CSM	Altitude.....20,000 ft. "Belly-Down" roll orient nominal trajectory conditions
3. Descent From 20,000 Ft. to Hover	. Fly nominal altitude, altitude-rate, range profile . Achieve nominal hover conditions . Roll 180° to "Belly-Up" attitude . Inspect and make gross choice of landing site	. 6735 fps ΔV nominal for total descent . 10% reserve . Ground slope to 3 deg.	. Dependent upon nominal trajectory . "Belly-Down" roll orientation . Range.....≈20 n.m.	Horiz. Vel.....75-0 fps Vert. Vel.....5-0 fps Altitude.....1000 ft. Attitude.....Vertical and Downrange
4. Hover and Letdown	. Accurate determination of altitude, altitude rate, horizontal velocity . Manual control of letdown . Coarse attitude control . Roll capability through 360° for landing site search	. Touchdown A. S. A. P. (< 2 min.) . Reasonable attitude and attitude rates . Ground slope to 15 deg.	Horiz. Vel.....75-0 fps Vert. Vel.....5-0 fps Altitude.....1000 ft. Attitude.....Vert. and downrange	Touchdown Vert. Vel...10-0 fps Touchdown Horiz. Vel..5-0 fps Angle.....±5 deg. Angle Rate.....±5 deg/sec. Range to landing site.0.5-0 n.mi.
5. Pre-launch	. Align to local vertical, azimuth . Determine plane of CSM orbit rel. to LEM . Determine position of CSM in orbit	. Lunar surface attitude constraints . Maximum allowance alignment and warmup time . Limited CSM tracking time	. All systems off-coarse alignment of LEM known from touchdown conditions . CSM orbit and rel. position known via communication link	Vertical Alignment Azimuth Alignment CSM orbit plane uncertainty CSM altitude uncertainty CSM central angle uncertainty
* Abort From Powered Descent	As a function of time in descent . Decide when to stage descent engine . Initiate rapid pitch-over to initial ascent attitude . Follow pre-programmed thrust-pitch profile to first burnout . Perform transfer to rendezvous point	. Achieve clear pericynthion and bounded apocynthion orbit conditions at first burnout . 6225+fps ΔV is goal for capability to continue to rendezvous . 7079+fps ΔV max. to achieve safe orbit and await rescue (+ not including descent fuel remaining)	. Dependent upon initial conditions at time of abort - detectable deviation from nominal trajectory	. LEM-CSM relationship dependent upon time of abort from descent . Velocity, Altitude, Flight path angle same as for powered ascent (Item 6)

TABLE D-1 (CONTINUED)

GUIDANCE AND CONTROL MISSION REQUIREMENTS

MISSION PHASE FUNCTIONAL REQUIREMENTS	DESCRIPTION OF TASK	CONSTRAINTS	INITIAL CONDITIONS	TERMINAL CONDITIONS
6. Powered Ascent	. Roll to proper ascent pitch plane, fly nominal pitch-thrust time profile . Achieve nominal burnout conditions . Initiate launch within launch window	. Achieve bounded apo-pericynthion burnout conditions . 6025 fps ΔV nom. 6225 fps ΔV max. . Launch window	. Lunar surface . Launch time phase angle between LEM-CSM . Range (Hohmann Ascent) .....98 n.mi. . LOS ∢.....56° (from horiz.) . Any roll attitude angle	Range to CSM.....193 n.mi., LOS ∢ = 22° (from horiz.) Nom. Vel.....5580 fps Nom. Flt. Path ∢.....0° Nom. Altitude.....50 k ft.
7. Midcourse Corrections	. Differential correction technique using two point measurements of line of sight angle and range (or range rate) . First measurement made at burnout, second measurement and correction made approximately 1800 sec. . Multiple corrections may be made if necessary	. Perform with RCS . 50 fps ΔV nom. 100 fps ΔV max.	Nom. Vel.....5580 fps Nom. Flt. Path ∢.0° Nom. Altitude....50 k ft.	Range.....30-20 n.mi. Range Rate.....300-100 fps Attitude.....Any ∢ around LOS axis
8. Rendezvous Terminal Homing Maneuver	. Null line of sight rate - (within 0.2 mr/sec) . Reduce range rate as function of range to CSM	. Maintain visual contact . Perform with RCS . 335 fps ΔV nom. 570 fps ΔV max. . Perform within time schedule	. Range.....30-20 n.mi. . Range Rate.....300-100 fps . Attitude.....Any ∢ around LOS	Range.....1000-500 ft. Range Rate.....±10 fps Attitude.....Any ∢ around LOS axis
9. Manual-Visual Docking Maneuver	. Control LEM attitude with respect to CSM . Control LEM range and range rate with respect to CSM	. Maintain visual contact . Perform with RCS . 25 fps ΔV nom. 50 fps ΔV max.	. Range.....1000-500 ft. . Range Rate.....± 10 fps . Attitude.....Any ∢ around LOS axis	See Terminal Conditions for Item 1

E. NOMINAL DESCENT TRAJECTORIES[1]

General Discussion of Powered Descent

Following separation from the CSM the LEM descends in a synchronous (equal period) transfer orbit, from a CSM circular parking orbit to a pericynthion altitude of 50,000 feet. The powered descent phase starts at or near pericynthion and ends at LEM touchdown on the lunar surface. For convenience of discussion, this phase of the mission is subdivided as follows:

1. Initial powered descent - begins at pericynthion of the transfer orbit and ends when the LEM is approximately 20 n.mi. from the intended landing site. The trajectory for this phase of the descent is based on a constant thrust pitch program which is optimized for minimum fuel (minimum ΔV). When the quality of radar information from the lunar surface exceeds that of the IMU, these data will update the computed position and velocity coordinates to correct the terminal portion of the trajectory. The criteria for the exact termination conditions of this phase will be based primarily on visibility, ΔV and control requirement considerations.

2. Final powered descent - As presently conceived by GAEC this phase will employ one of the following guidance techniques:

   Proportional navigation: With this procedure, the rotation of the line of sight to the target is nulled so that the LEM approaches the hover point at a near constant flight path angle. There will be a provision for engine throttling to obtain the proper rate of velocity reduction.

   Constant thrust-to-weight ratio: A second procedure which seems to hold promise is based on automatic throttling of the descent engine to maintain a constant thrust-to-weight ratio. This may be combined with a constant attitude pitch program to produce a final descent phase which is compatible with a manual mode of control.

3. Flare - The term flare maneuver is used to designate the final phase in the attitude control program just before the LEM arrives at the hover point in order to bring the LEM to the vertical attitude required. At present, the flare maneuver has not been thoroughly investigated since it is so strongly dependent on the final descent procedure. There

[1] "Trajectory Characteristics During the LEM Mission, II" LMO-500-48

is a possibility that it can be incorporated into a proportional navigation scheme rather than remain as a separate phase of the descent. The problem is further complicated by the desirability of maintaining visibility to the landing site throughout the final descent to the hover point.

4. Hover to landing - The LEM will reach the hover point, at an altitude of about 1000 ft., with zero vertical velocity. As a result of the flare maneuver, the LEM should be at a vertical attitude, with the thrust level reduced to equal the LEM lunar weight. There may be a residual horizontal velocity of up to 100 ft/sec. The pilot must then descend toward the lunar surface, select a landing site, and touchdown, in a period of two minutes. There is provision for a horizontal translation capability to aid in landing site selection. It is expected that the instrument displays during this phase can use information from the IMU as well as directly from the radar.

Study Areas

The descent phase is divided into three study areas; initial descent, final descent to hover, and descent from hover to touchdown. The flare maneuver and the final descent are grouped together for this study.

1. Initial descent - The original study effort concerns itself with the development of optimum trajectories (with respect to ΔV) which have been computed for a set of initial altitudes and velocities.[2] However, the initial conditions of interest are for descent from the 50,000 ft. pericynthion of a synchronous transfer orbit. The optimum trajectories studied were all based on constant thrust during the initial descent phase. The required final conditions were 0 velocity at the 1,000 ft. hover point. The results have shown that for these conditions, minimum ΔV descent is obtained with an initial T/W ratio of 0.6. This is not a strong minimum however, and an initial T/W ratio of 0.4 is being used as the nominal, resulting in a ΔV penalty of about 50 ft/sec. The lower thrust level is favorable for engine sizing, requires less thrust range to obtain the minimum thrust level required, and improves the LEM/CSM phasing relation. Since the actual descent will depart from the optimum well before the hover point, it will be necessary to rerun the optimums for a different set of end conditions. Nevertheless, the optimum trajectory to the hover point is extremely valuable as an ideal case against which other descent regimes may be judged. Some off-nominal descents have been run and they indicate that some alteration of the optimum is possible without a large ΔV penalty. The true optimum approaches the hover point with a shallow flight path angle and a near horizontal pitch attitude. This results in difficulties with pilot visibility of the touchdown area and requires

[2] Powered Descents from Various LEM Pericynthion Conditions, LMO-500-8

an extreme flare maneuver. It seems, therefore, desirable to attempt reshaping the initial descent so that the transition to final descent occurs at a steeper path angle. In the optimum trajectory, the pitch rate varies continously but not rapidly. This suggest the possibility of replacing the optimum with a constant pitch rate program.

A complete error analysis must be carried out for all phases of the descent trajectory, since for an abort situation during descent any errors in initial conditions for abort must be added to errors arising during the abort in order to determine realistic back-up guidance requirements.

2. Final descent to hover - At present, the entire final descent phase is being studied, with particular emphasis on choice of guidance techniques to be used in the error analysis.

The two most promising procedures appear to be proportional navigation, and constant attitude descent. The proportional navigation process has been used to develop the present nominal trajectory. In this procedure the primary reference is the line of sight (LOS) from the LEM to the hover point. Since the location of the hover point is calculated, rather than observed directly, this is not a true "homing" procedure but rather a method for approaching final conditions which allows for constant correction of errors which may develop. The basic procedure is to control LEM attitude so that the rate of rotation of the line of sight is nulled. Concurrently the thrust is reduced, so that hover is not approached at too rapid a rate of deceleration. If the process were followed exactly, the LEM would descend to the hover point along a straight line in lunar coordinates. In actual practice, the guidance scheme does not null the LOS rate of rotation immediately, but in a manner which is dependent on the "gains" used in the guidance computation. For a particular set of values of these gains, the actual shape of the homing trajectory is a function of the LEM position and velocity vectors relative to the intended hover point at the start of the final descent phase. This means that deviations from the nominal during initial descent will affect the parameters of the final descent trajectory, in particular the flight path angle at the approach to the hover point.

In the constant attitude descent, the LEM is held at a constant inertial attitude, and the engine is throttled continuously to maintain a constant thrust-to-weight ratio. The two degrees of freedom represented by the attitude and thrust-to-weight ratio are sufficient to bring the LEM to a specified velocity, path angle and altitude to initiate the flare. However, if range control is also desired it will be necessary either to vary the time of initiation of the constant attitude

phase or introduce a flare from variable initial conditions. Both these procedures are difficult, so that it may be necessary to relinquish firm range control if constant attitude final descent is used.

A point in favor of the constant attitude final descent is the relative ease with which it could be flown in a manual mode. Whereas if proportional navigation is used in the nominal it will probably be necessary to revert to a simpler procedure if a switch is made to the manual mode.

Abort Prior to Hover

The question of which engine is used for the abort is more critical during the powered descent, since the descent engine will be available for more than just a "token" effort, as in the case of abort from hover; the descent fuel remaining and ΔV required to achieve an abort ascent is shown in Figure E-1. Most study effort to date has been devoted to abort on the ascent engine, since descent engine failure is the most likely cause of an abort before the hover point.

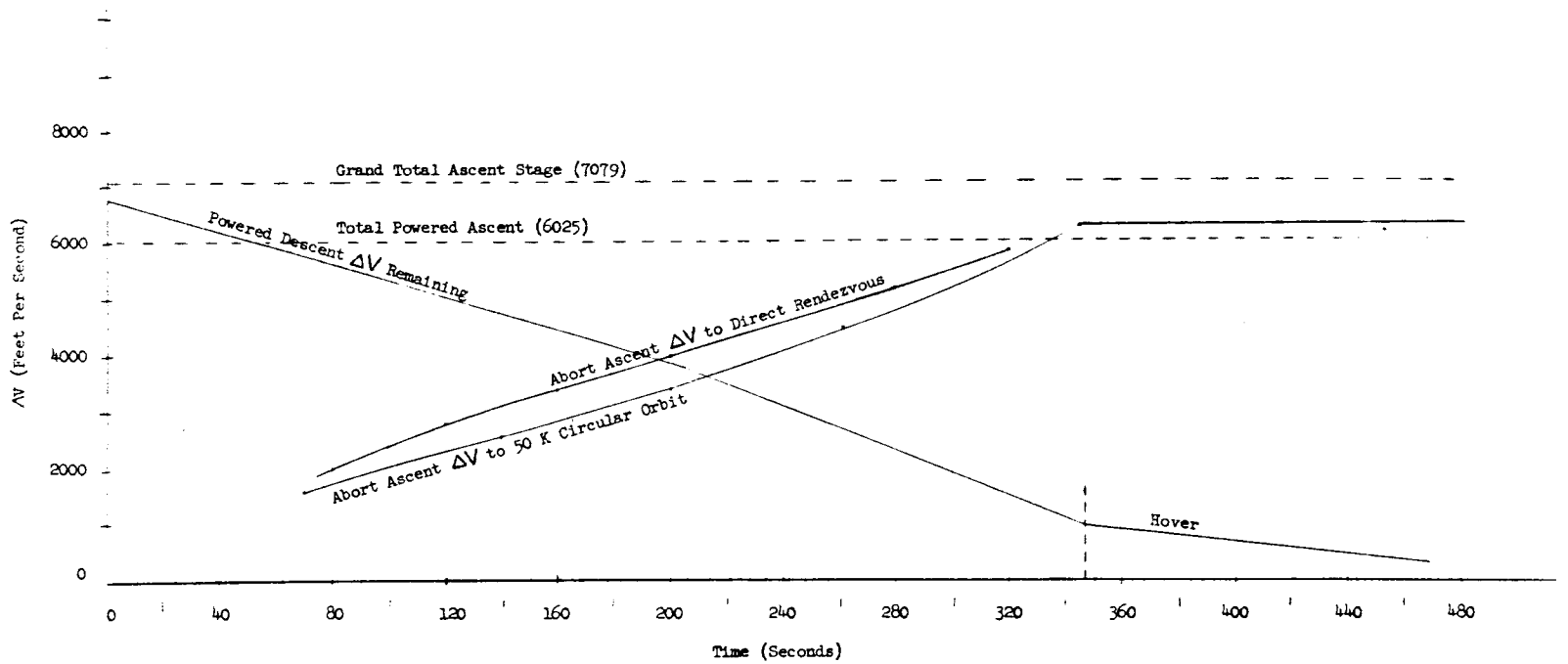
One of the major constraints involved in the development of trajectories for abort from different points on the powered descent trajectory is the limitation on back-up computing (or programming) capacity. At the hover point, there exists a single set of initial conditions for which a single nominal abort trajectory with simple corrections for small initial condition errors can be devised. During powered descent, the possible initial conditions for an abort vary with time, and pass through a wide range of values. Hence, even though the abort trajectories may be defined by the same set of parameters described in the previous sections, these parameters will now be continuous functions of time.

A series of abort trajectories has been developed, based on the scheme used for the hover point aborts: a vertical thrusting phase, a rapid pitch maneuver, and a low pitch rate phase. As mentioned in the hover point case, three of the four parameters (vertical rise time, rapid pitch angle, low pitch rate, and burn-out time) are required to fix burn-out conditions. This leaves one parameter free for trajectory selection. Figures E-2(a) through 2(e) show the characteristics of a set of such aborts. They start from different points on the nominal descent trajectory. All curves are shown as a function of time along the powered descent path. For these aborts, one of the parameters, low pitch rate, is fixed at -0.13 deg/sec. Since some freedom is allowed in the selection of a burnout altitude, a second parameter, vertical rise time, is varied to give this additional degree of freedom. Figure E-2(d) shows the burn-out altitude variation with vertical rise time. Figures E-2(a) and 2(b) show the values of the remaining two parameters, rapid pitch and burn-out time, corresponding to the vertical rise times. Figures E-2(c) and 2(e) show the ΔV required to obtain circular orbit and the central angle traversed during the abort.

FIGURE E-1

Reference: LMO-500-29 (Figure 9 and 14)
LMO-500-37, Revision A (Table 1 and 2)
LMO-500-48 (Figure 8.2.3-2)

CSM Orbit Altitude 80 n. mi.
LEM Central Angle to Powered Descent 94°
50,000 feet Synchronous Orbit Percynthion
Descent Engine (T/W_o) 0.396
Ascent Engine (T/W_o) 0.48



ΔV FOR ABORT ASCENT

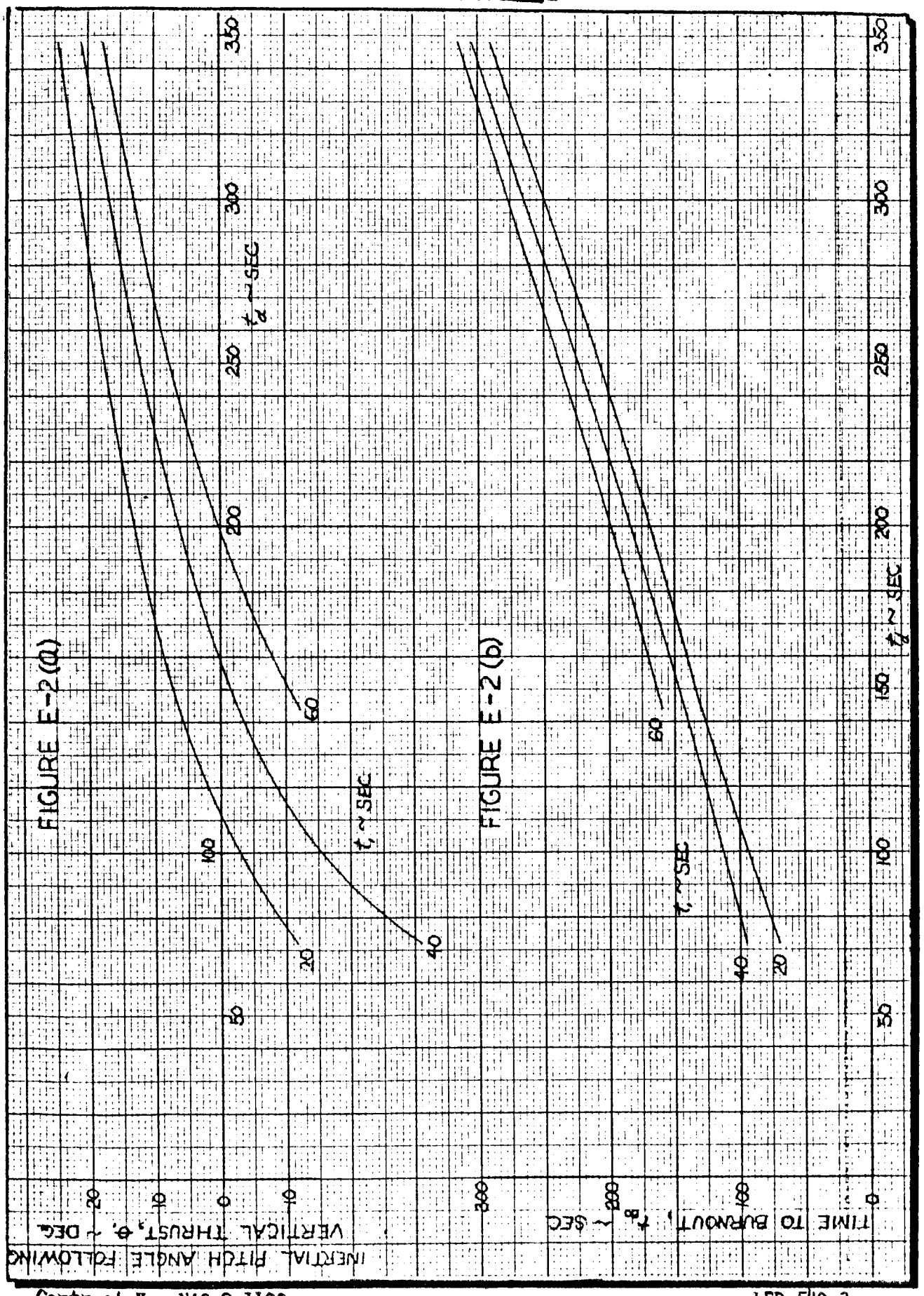


FIGURE E-2(a)
INERTIAL PITCH ANGLE FOLLOWING VERTICAL THRUST, θ_v ~ DEG
20
10
0
10
20
50
100
40
60
t_α ~ SEC
250
300
350
FIGURE E-2(b)
TIME TO BURNOUT, t_{bo} ~ SEC
300
200
100
0
20
40
60
t_α ~ SEC
150
200
250
300
350
50
100

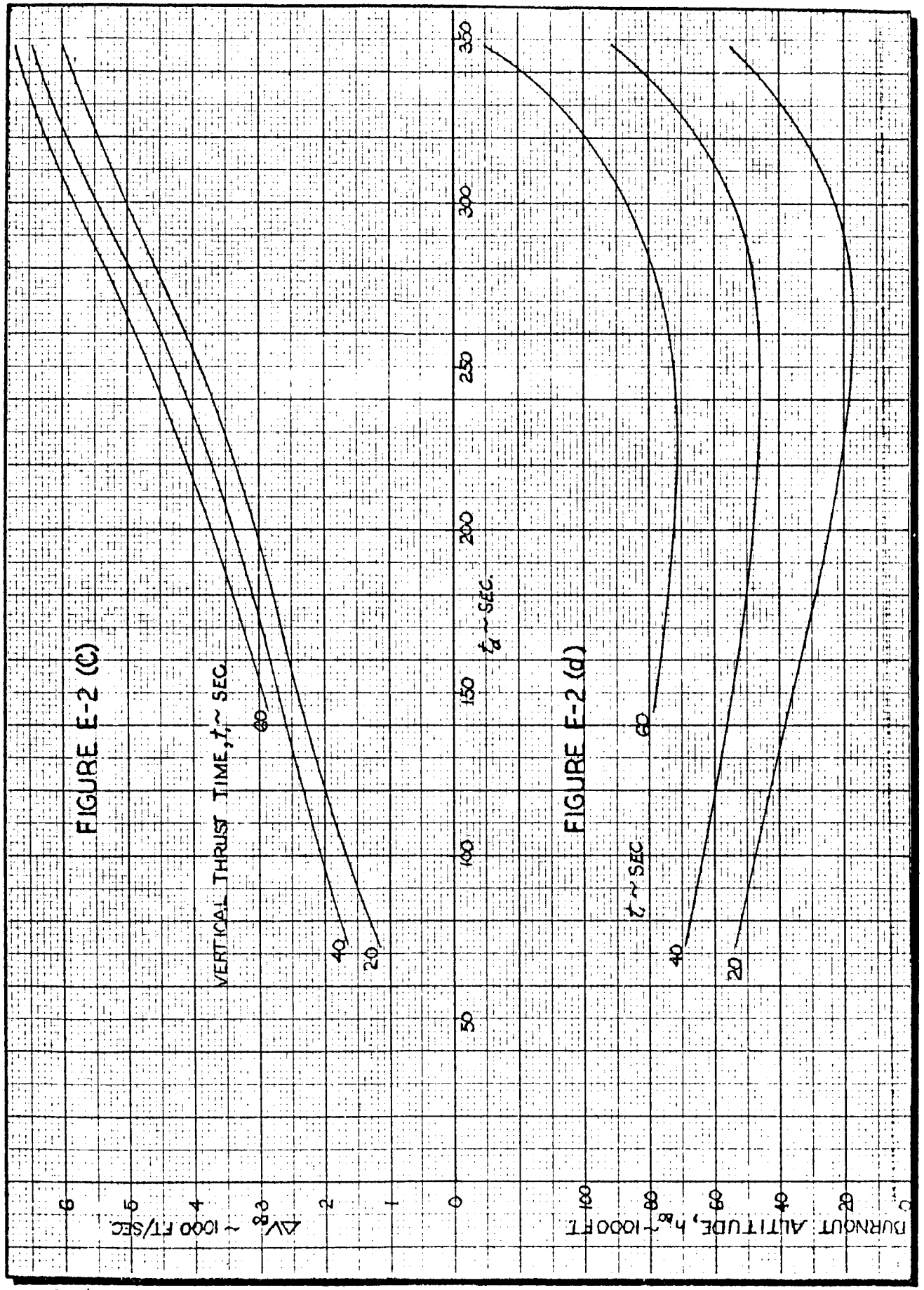


FIGURE E-2 (c)
VERTICAL THRUST TIME, t_v ~ SEC.
20
40
60
ΔV_{BO} ~ 1000 FT/SEC
6
5
4
3
2
1
0
FIGURE E-2 (d)
t_d ~ SEC.
t_v ~ SEC
20
40
60
BURNOUT ALTITUDE, h_{BO} ~ 1000 FT
350
300
250
200
150
100
50
100
80
60
40
20
0

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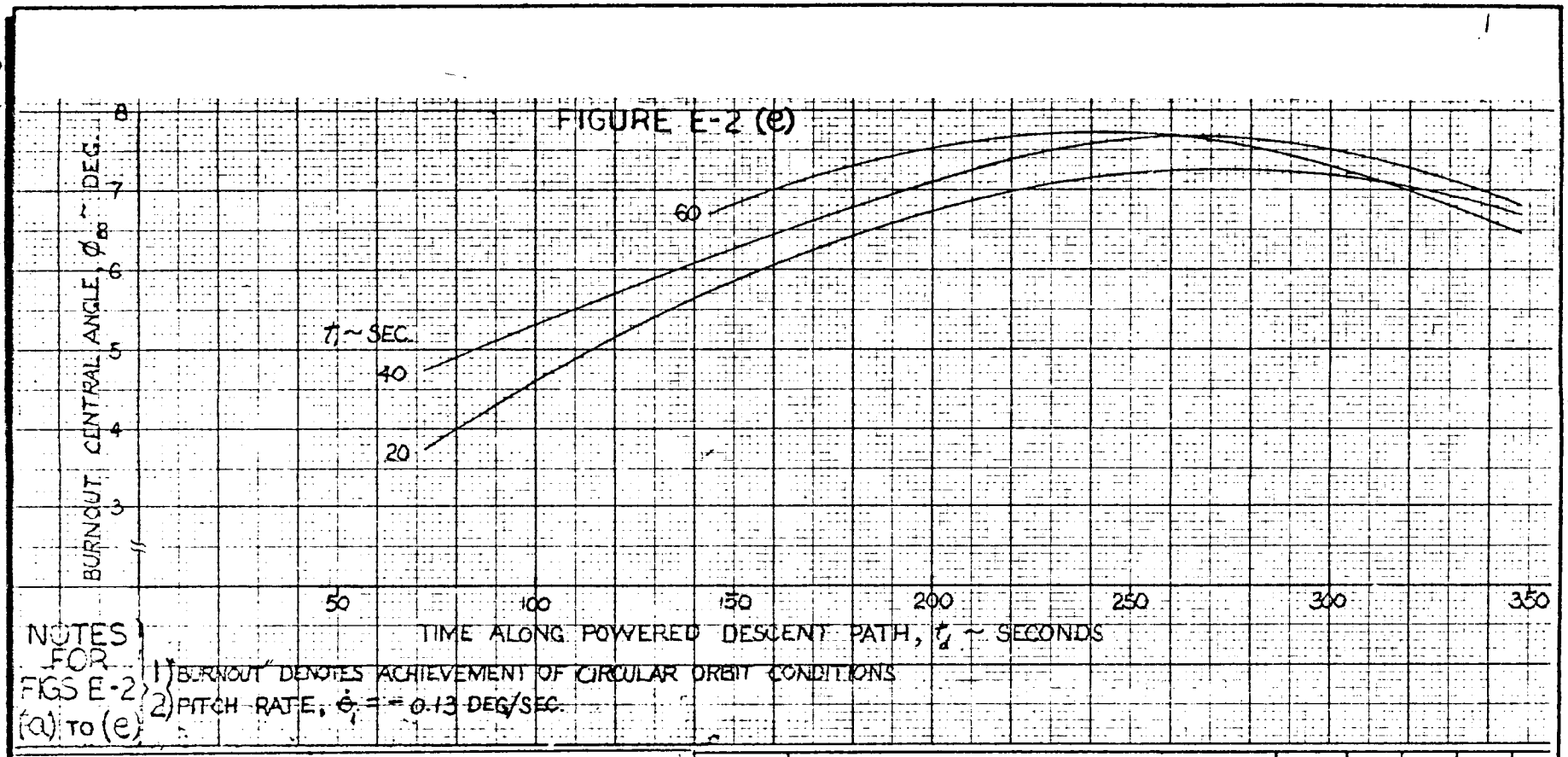


NOTES FOR FIGS E-2 (a) TO (e)
1) "BURNOUT" DENOTES ACHIEVEMENT OF CIRCULAR ORBIT CONDITIONS
2) PITCH RATE, $\dot{\theta}_1 = -0.13$ DEG/SEC.

SCALE	SERIES	RUN	CONFIGURATION
BURNOUT TRAJECTORY PARAMETERS FOR VARIOUS VERTICAL THRUST TIMES DURING ABORT TO CIRCULAR ORBIT. POWERED DESCENT PATH "C" (94° LEM CENTRAL ANGLE TO POWERED DESCENT)			ASCENT ENGINE $T/W_o = 0.48$ CSM ORBIT ALTITUDE = 80 N. Mi. SYNCHRONOUS ORBIT PERI- CYNTHION = 50,000 FEET DESCENT ENGINE $T/W_o = 0.396$

FIGURE E-2(a) TO (e) – CHARACTERISTICS OF A SET OF ABORTS.

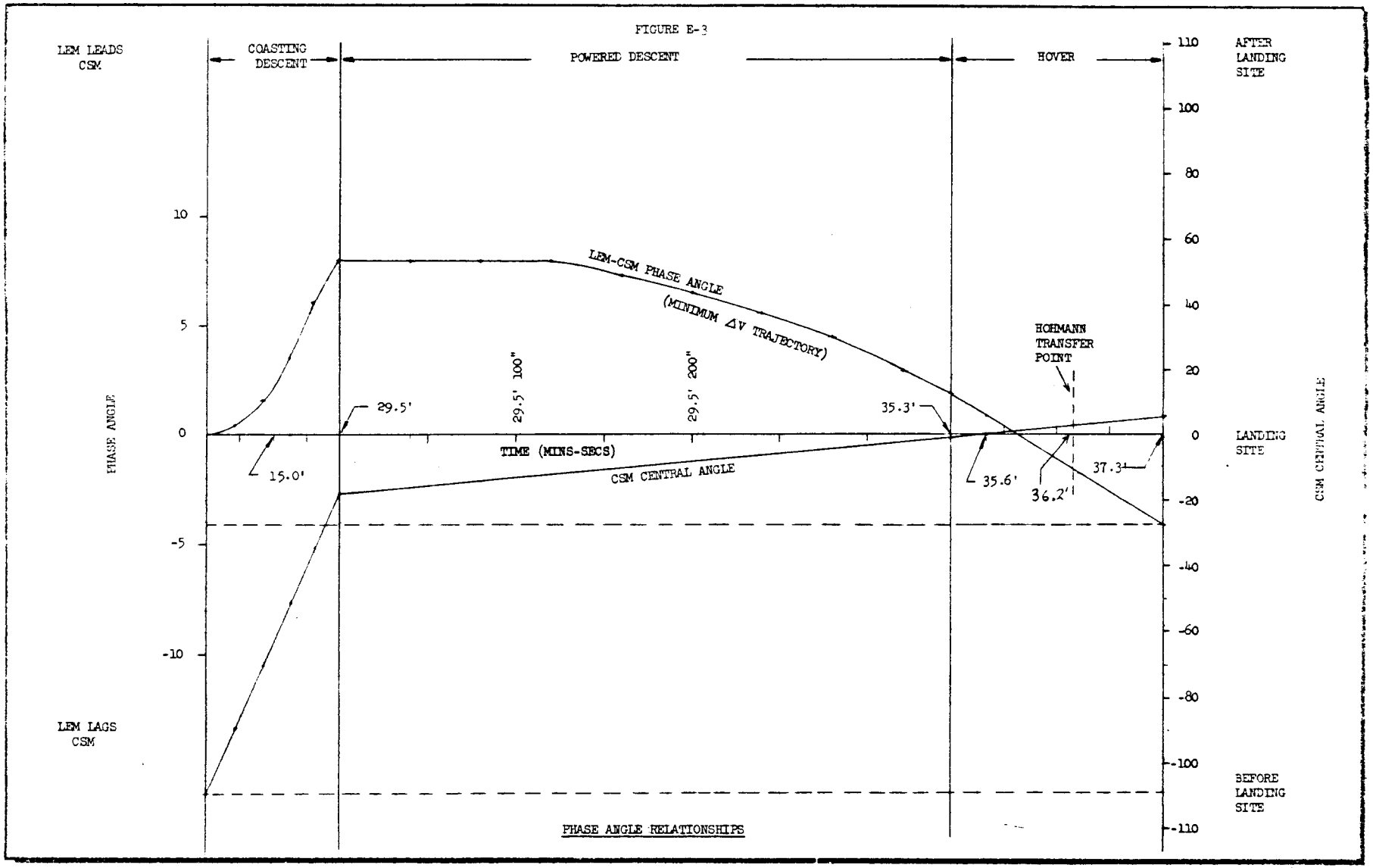
The remaining problem is to select the set of abort trajectories which can be handled most easily by an abort computer. Once a computational procedure is determined for selection of the abort paths from the nominal descent it is still necessary to establish correction procedures to handle cases where the descent deviates from the nominal. If the allowable deviations are restricted to small values (i.e. abort will be initiated if they are in danger of being exceeded) then it may be possible to work with approximate values of correction coefficients, (partial derivatives of required parameter values with respect to deviation of initial conditions of abort from the nominal) in order to correct the parameters to take into account an off-nominal descent. The complications that arise indicate that the use of a generalized abort computer, rather than one based solely on the expected descent might turn out to be no more complicated. Such a computer would be able to calculate the required abort trajectory parameters for any set of initial conditions. The procedure could be derived from curve fits of the precomputed information.

While the abort computation problem is eased somewhat by the use of trajectories which can be described in a simple parametric fashion, the difficulty involved in continuously generating these parameters as a function of time still remains. It is therefore desirable to simplify the functional relations as much as possible. Unfortunately there is a strict limit to the allowable degree of simplification. In the scheme just described, one of the four parameters, low pitch rate, is assumed to be constant throughout. A second parameter, vertical rise time, is shown to be somewhat arbitrary, at least to the extent that burn-out altitude does not need to be constrained as tightly as the other burn-out conditions. The remaining two parameters, however, must be defined with great accuracy, since they will define the burn-out velocity and path angle. In general, whatever parameters are used to describe the abort, all but two may be described in a sufficiently arbitrary fashion so that they can be computed from linear or step functions, while the remaining two should not be approximated.

Phasing Relations and Coasting Ascent to Rendezvous

The phasing problem, which places certain restrictions on the central angle relationship between the LEM and CSM at the start of coasting ascent, has often been thought to be more serious than it really is. In considering phase angle relationships (see Figure E-3), a common reference case is the phase angle required for a Hohmann transfer. When both the LEM and CSM orbits are circular at 50,000 ft. and 80 n.mi. respectively, the proper phasing has the LEM trailing the CSM by 9.4 degrees of central angle. At this phasing the LEM may start Hohmann transfer, otherwise the LEM may either continue in a parking orbit or take a non-optimum transfer. While in a parking orbit, the LEM, at the lower altitude, will be overtaking the CSM. Therefore, the critical situation comes about when the LEM is ahead of the required Hohmann point because waiting in a parking orbit will make matters worse rather than improve them. The ability to take a non-optimum transfer introduces a transfer window at the cost of additional ΔV. The synchronous orbit is one of these non-optimum transfer orbits. If the LEM were to start an ascent at 50,000 ft., leading the CSM by 8 degrees, the

FIGURE E-3



PHASE ANGLE RELATIONSHIPS

coasting ascent orbit would be identical to the synchronous orbit and would require 375 ft/sec more ΔV to inject and rendezvous than the Hohmann. (200 ft/sec at injection and 375 ft/sec to rendezvous as opposed to 100 and 100 for the Hohmann.)

The nominal mission starts with a synchronous orbit descent to 50,000 ft. followed by powered descent. The first limiting case for abort phasing occurs when the descent is aborted almost immediately. In this case the coasting ascent would be nearly identical to the synchronous orbit. As the descent proceeds, the CSM gains on the LEM so that the phasing becomes more favorable until, for an abort initiated 1.1 minutes after the hover point is reached, the Hohmann transfer conditions are attained. Hence, the least optimum transfers occur after early aborts, when ΔV is not critical and transfers become more optimum as the mission proceeds and ΔV considerations become more significant.

Orbit Insertions and Error Analysis

In the discussion of the abort from powered descent, it has been assumed that the desired burn-out conditions of the abort are circular orbit velocity at an altitude in the vicinity of 50,000 ft. This would be followed by injection into coasting ascent orbit requiring between 100 and 200 ft/sec ΔV, depending on the time of abort. An error analysis of the powered ascent has been performed to determine the ascent guidance accuracy required to guarantee a clear pericynthion altitude of 40,000 ft.

Figures E-4 and E-5 show some of the preliminary results of this error analysis. There are several types of error coefficients of interest. The first type gives the errors in burn-out conditions (altitude, velocity, path angle) resulting from certain inaccuracies which can be expected during the powered abort. Figure E-4 gives an example of these coefficients showing burn-out errors resulting from an error in the final pitch rate during an ascent from the lunar surface. This ascent has the same nominal pitch program as an abort from the hover point, and consequently, the same error sensitivity. (Similar plots show effects of thrust misalignment and thrust magnitude errors.) The second type of error coefficient relates the error in injection conditions to changes in the important parameters of the resulting orbit. These coefficients may be derived from the basic equations of orbital mechanics. Probably the most interesting of these coefficients is the variation in pericynthion altitude resulting from an error in injection flight path angle. Figure E-5 shows the allowable flight path angle error to maintain a pericynthion altitude of 40,000 ft., when the nominal injection conditions are 50,000 ft. altitude with a zero flight path angle. The allowable angle error is given as a function of injection velocity. The sensitivity to flight path angle decreases greatly as injection velocity is increased above circular orbit velocity. Thus, there is a distinct advantage in fashioning a nominal ascent trajectory in which a single firing terminates in the supercircular velocity required for the ascent coast to CSM altitude. Another procedure for increasing the tolerance to flight path angle error is to bias the nominal flight path angle at burn-out 0.5 degrees or more above horizontal. This would not reduce the effect of flight path angle

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FIGURE E-4

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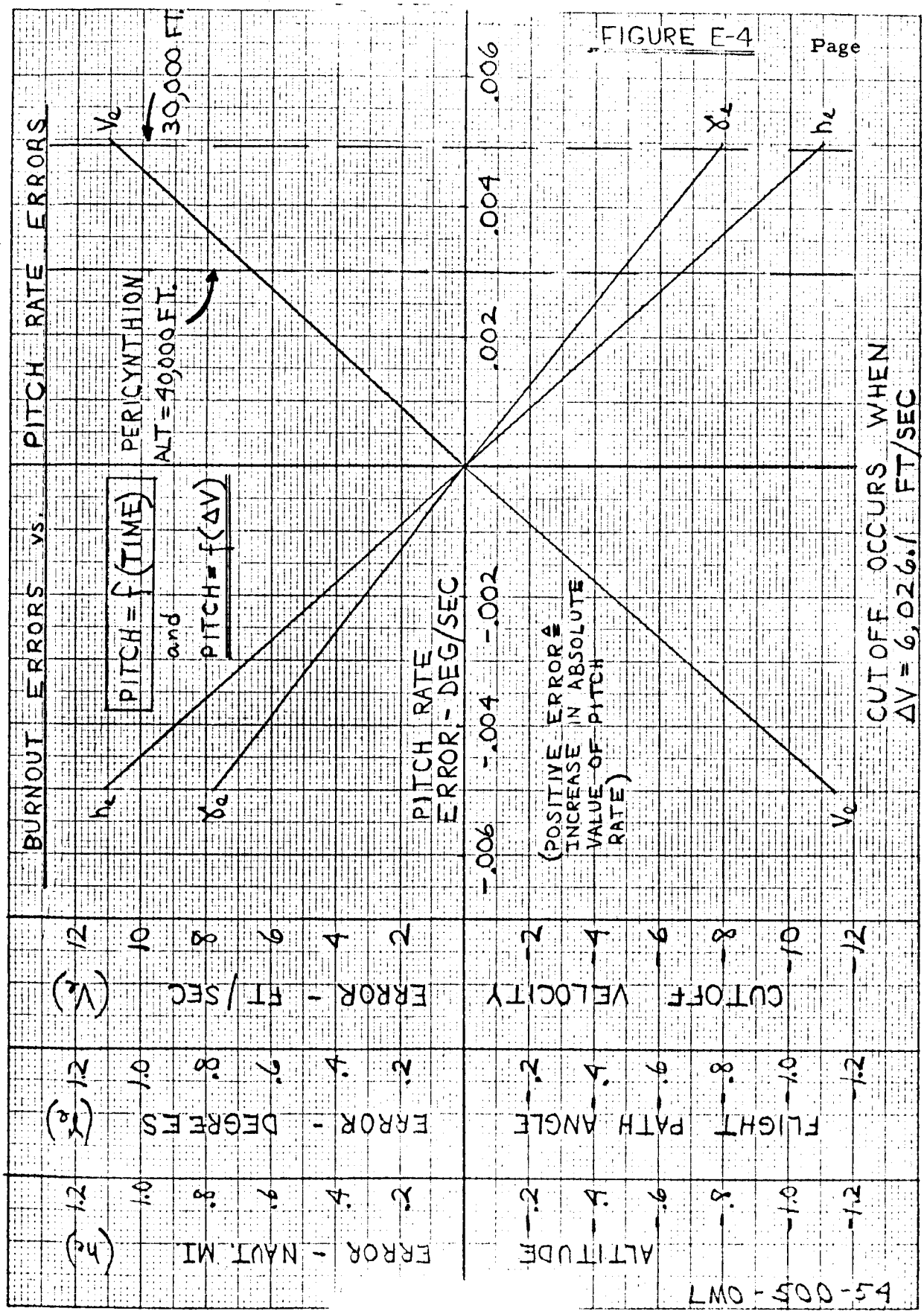

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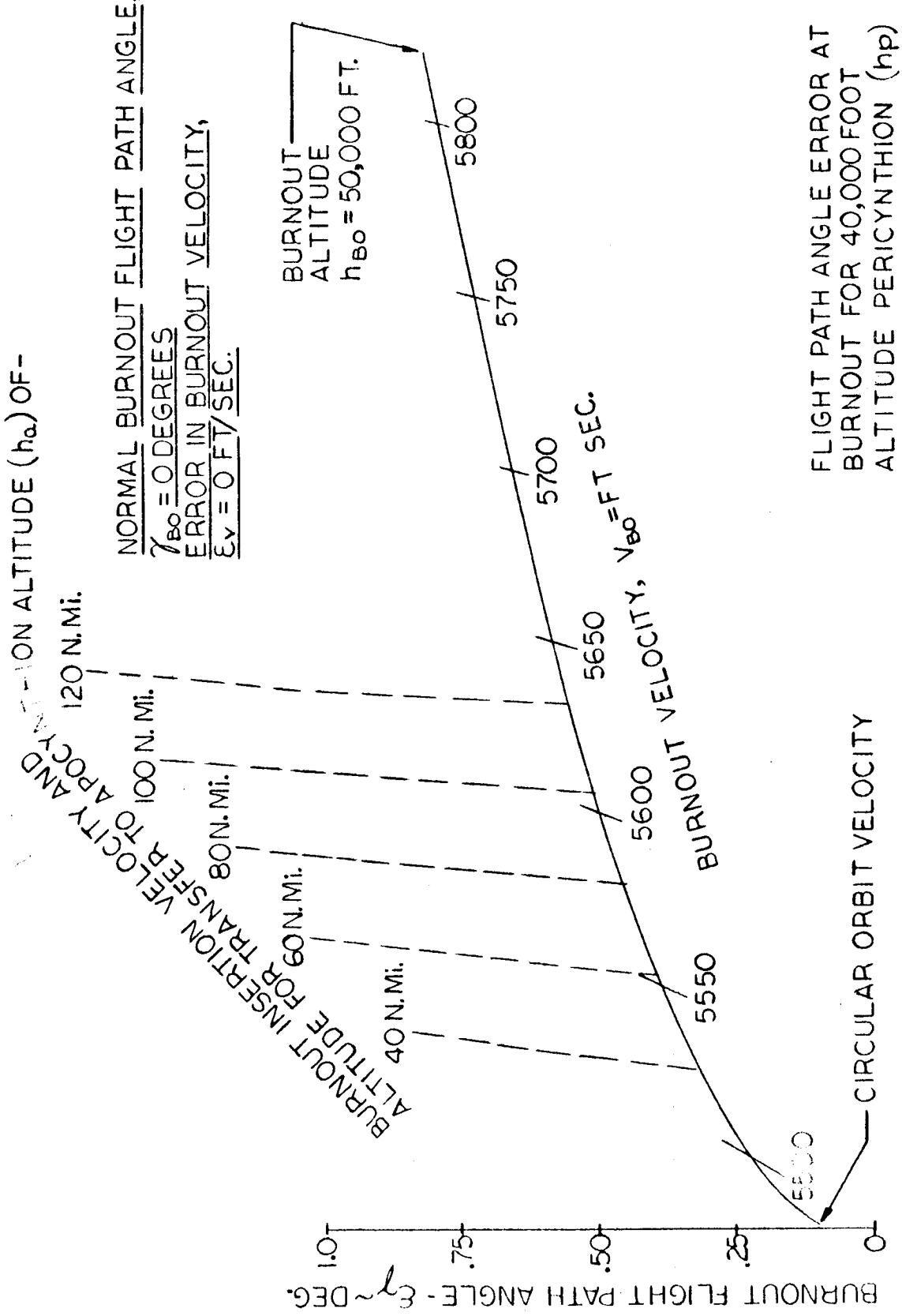
Contract No. NAS 9-1100
Primary No. 660



FLIGHT PATH ANGLE ERROR AT BURNOUT FOR 40,000 FOOT ALTITUDE PERICYNTHION (hp)

FIGURE E-5

Contract No. NAS 9-1100
Primary No. 660

REPORT LED-540-3
DATE 9 July 1963

errors on pericynthion altitude, but it would insure that pericynthion did not occur until more than half an orbit after injection. Once transfer orbit injection has taken place, the midcourse correction procedure would be used to effect the rendezvous.

The requirement for an ascent from the lunar surface at any time under back-up guidance presents the problem of achieving a safe pericynthion parking orbit in order to obtain proper phasing for ascent to the CSM within the ΔV capability of the LEM. Investigation into elliptical parking orbits shows that an increase in synodic time from the 17.7 hours of the 50,000 ft. circular orbit with respect to the 80 n.mi. CSM circular orbit occurs at the rate of about 1 hour for every 10 ft/sec increase in burn-out velocity. Therefore, in order to decrease the sensitivity of the clear pericynthion restriction to ascent guidance errors, additional burn-out velocity can be added at the cost of additional waiting time in orbit.

Abort Trajectories from Powered Descent

Although in general abort has been defined as any deviation from a planned mission, for the purposes of the present discussion only the trajectory aspects of aborts occurring during powered descent are considered. An emergency ascent from the lunar surface certainly represents an abort situation, but from the trajectory point of view there is not a significant distinction between emergency and nominal ascents.

The abort trajectories treated in this section fall basically into the following categories:

1. Descent initiation
   - from hover
   - prior to hover
2. Trajectory phase
   - powered phase
   - coasting phase
3. Propulsion system utilized
   - descent engine until burn-out
   - ascent engine only

Although all combinations of the two alternatives in each category are possible, the considerations pertaining to each case of a category are sufficiently independent of the other catagories to permit them to be treated generically in a separate discussion.

Generally, the nature of the coasting transfer orbit determines the requirement for burn-out conditions of the powered phase. However, the variation in burn-out conditions required to meet the injection conditions into various possible transfer orbits will not significantly affect the shaping of the major portion of the powered abort. Therefore, it will be assumed that the powered phase of abort ends with the LEM at a low altitude, (50,000 - 100,000 ft.) near circular parking orbit.

The development of abort trajectories and procedures does not, at this time, require an exact knowledge of weights, thrust levels or specific impulses; small changes in the vehicle performance or in the nominal trajectories may be countered by small changes in the parameters governing aborts.

Abort from Hover

As an example of the logical procedure for deriving an abort program, an abort from the hover point for which the nominal initial conditions are zero velocity at 1000 ft. altitude is being considered. Figure E-6 shows an inertial pitch angle program as a function of time for an optimum (minimum ΔV) ascent to a 50,000 ft. circular orbit. This particular trajectory applies for a T/W_o of 0.474. It is constrained by the requirement that the vehicle should rise vertically for at least 1500 ft. to allow rotation about the (vehicle) roll axis into the proper azimuth plane. From this point, the optimization process requires an instantaneous pitch to an attitude of 24 degrees above horizontal, followed by a period of low, constant pitch rate until burn-out time is reached. If the instantaneous attitude change of the optimum is replaced by a pitch maneuver performed at as rapid a rate as physical limitations allow, a simple, near optimum pitch program for a hover point abort can be generated. This pitch program is completely described by four parameters: vertical rise time, high rate pitch change angle, final pitch rate and burn-out time which are constants for a given set of initial and final conditions. This basic simplicity makes the procedure ideal for back-up guidance and manual operation.

Burn-out conditions may be described by four variables if out of plane situations are ignored. These are altitude, velocity magnitude, flight path angle and LEM/CSM phase angle; the latter may be considered least important from the point of view of successful ascent since it does not affect the clear pericynthion characteristics of the parking or transfer orbit. Flight path angle is the most critical in this respect.

The burn-out conditions may thus be considered as a point in the three-dimensional function space defined by altitude, velocity magnitude and path angle. If any individual characteristic of the ascent trajectory is altered, in general, all three coordinates of the end point will change. The relationships between these parameters, expressed in terms of the partial derivatives of the end point coordinates with respect to any trajectory parameter, can be determined from trajectory calculations. Since the end-point coordinate space is three-dimensional, burn-out conditions may be

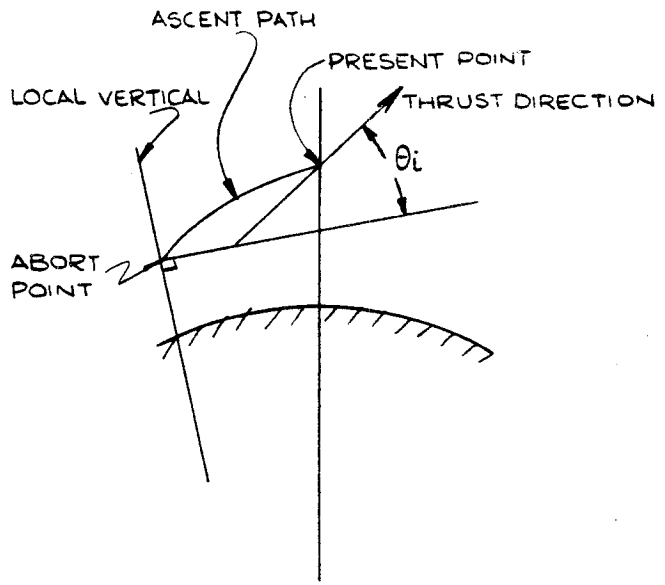
ASCENT ENG. T/W_o = 0.464

CSM ORB. ALT. = 80 N.MI.

SYNCH. ORB. PERICYN. = 50,000 FT.

DESCENT ENG T/W_o = 0.396

BURNOUT ALT. = 50,000 FT.



GRADIENT METHOD OPTIMUM INERTIAL PITCH ANGLE vs TIME DURING ABORT ASCENT P'W'RED DESCENT PATH "C" (94° LEM CENTRAL ANGLE TO P'W'RED DESC.)

FIG. E-6

completely controlled by the manipulation of three trajectory parameters. Hence, if during the vertical rise, the thrust magnitude is found to differ from the expected value the error which would result at burn-out may be nullified by adjusting any three of the four trajectory parameters. Furthermore, it may not be required that all burn-out conditions be fixed. If a circular parking orbit is required but there is some tolerance in the altitude, only a two-parameter control is necessary, since the deviating end-point is constrained to fall on a line in three-dimensional space rather than a point. In some cases, the effects of the parameters on the end-point are obvious, as in the case of burning time. For injection into a circular orbit, burning time has a large effect on final velocity, a small effect on path angle, (depending on attitude at burn-out) and almost no effect on altitude.

In addition to the inertial pitch program, there are other possibilities for referencing LEM attitude during an abort. One would be the use of the line-of-sight to the lunar horizon as an attitude reference. For the relatively short distance covered during the powered phase, line-of-sight to horizon is so nearly an inertial reference that there is very little change in the pitch procedure. Another suggestion involves the line-of-sight to the CSM as a reference. This is not at all similar to an inertial reference, so that, if the same trajectory shape is maintained, the simple "straight line" type of pitch profile may no longer be possible. Further, there is the complication caused by the variation in relative position of the CSM as the time of initiation of abort varies. To derive a simple pitch profile with respect to a LOS reference frame would require an appreciable deviation from the optimum procedure.

General criteria for evaluating abort pitch programming are:

1. reference should be easy to obtain;

2. pitch program should be simple (i.e. describable by as few parameters as possible) within this reference;

3. pitch program should be near optimum;

4. pitch profile should not have much "curvature" in an inertial frame (for ease of control).

An area which has not yet received much attention is the use of the remaining descent engine capability in performing the abort. This is particularly desirable in abort from hover since an addition of even 100 fps of velocity represents a valuable pad. At full throttle, the descent engine can be expected to burn for up to 40 seconds after hover. A reasonable procedure might be to plan to use up to 30 seconds of this capability and expend it all in vertical thrusting. Then, the descent engine can be jettisoned while the LEM is pitching to its new attitude, followed by ignition of the ascent stage. Treating vertical rise time as a variable, the three remaining parameters, rate of rapid pitch-over, final pitch rate and burn-out time may then be adjusted to give the correct final conditions. If the descent stage fails to burn long enough to give sufficient vertical rise prior to the rapid pitch maneuver, the ascent engine may be fired when the vertical velocity becomes zero, and abort continued as if from the original abort point.

Ascent Trajectories from Lunar Surface

In the nominal ascent from the lunar surface, the launching takes place 38 seconds after the CSM reaches the zenith of its orbit over the launch site. Powered ascent burn-out occurs at 50,000 ft. altitude, with the LEM injected into a Hohmann transfer orbit which will rendezvous with the CSM at 80 n.mi. The nominal ascent trajectory has been derived by slightly simplifying an optimum ascent trajectory. In the optimum, a short vertical rise time, required for terrain clearnace, is followed by an instantaneous pitch to 24 degrees above local horizontal, followed by a near constant pitch rate phase until burn-out. The nominal has been simplified by replacing the attitude discontinuity with a rapid pitch maneuver and by performing the final phase at a constant pitch rate. In this form, the pitch profile is compatible with the capabilities of the back-up guidance system.

Trajectory Characteristics

The LEM nominal ascent trajectory[3] is based on an intial thrust-to-weight ratio of 0.474, a specific impulse of 310 sec., and a constant engine thrust of 3500 lbs. The CSM is assumed to be in an 80 n.mi. circular orbit.

The LEM starts its ascent at an inertial pitch angle (referenced to the local horizontal at t=o) of 90 degrees. At t=17.4 sec., a constant pitch rate of -10 deg/sec is begun. At t=24 sec., when the pitch attitude is 24°, the pitch rate is decreased to -0.124 deg/sec. This low pitch rate is maintained until the end of powered ascent at t=296.2 sec. The burn-out velocity is 5580.5 ft/sec and the altitude 49,935 ft.; flight path angle is horizontal. These are the pericynthion conditions for a Hohmann orbit with apocynthion of 80 n.mi. The ΔV expended for this trajectory is 6026 ft/sec. Figure E-7 shows the nominal pitch profile vs. time, and Figure E-8 gives the resulting velocity and altitude time histories. Flight path angle, elevation angle and line-of-sight angle to the CSM are given in Figure E-9.

Ascent Launch Window

The nominal ascent procedure assumes that the launch occurs at the time when the phasing angle with the CSM makes direct insertion into the Hohmann transfer possible. Situations may develop where it is necessary or desirable to launch at some other time. A thorough analysis of this problem has been performed[4] and the results may be summarized as follows:

1. the ΔV penalty incurred by launching a few seconds late is prohibitive if direct insertion into a transfer orbit is required;

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[3] LEM Engineering Memo, LMO-500-48, "Trajectory Characteristics During LEM Mission, II", by P. Munter, dated 22 May 1963.

[4] LEM Engineering Memo, LMO-500-30, "A Generalized Study of ELliptic Transfer Orbits Between LEM and CSM Parking Orbits" by F. Murra, dated 17 April 1963.

FIGURE E-7

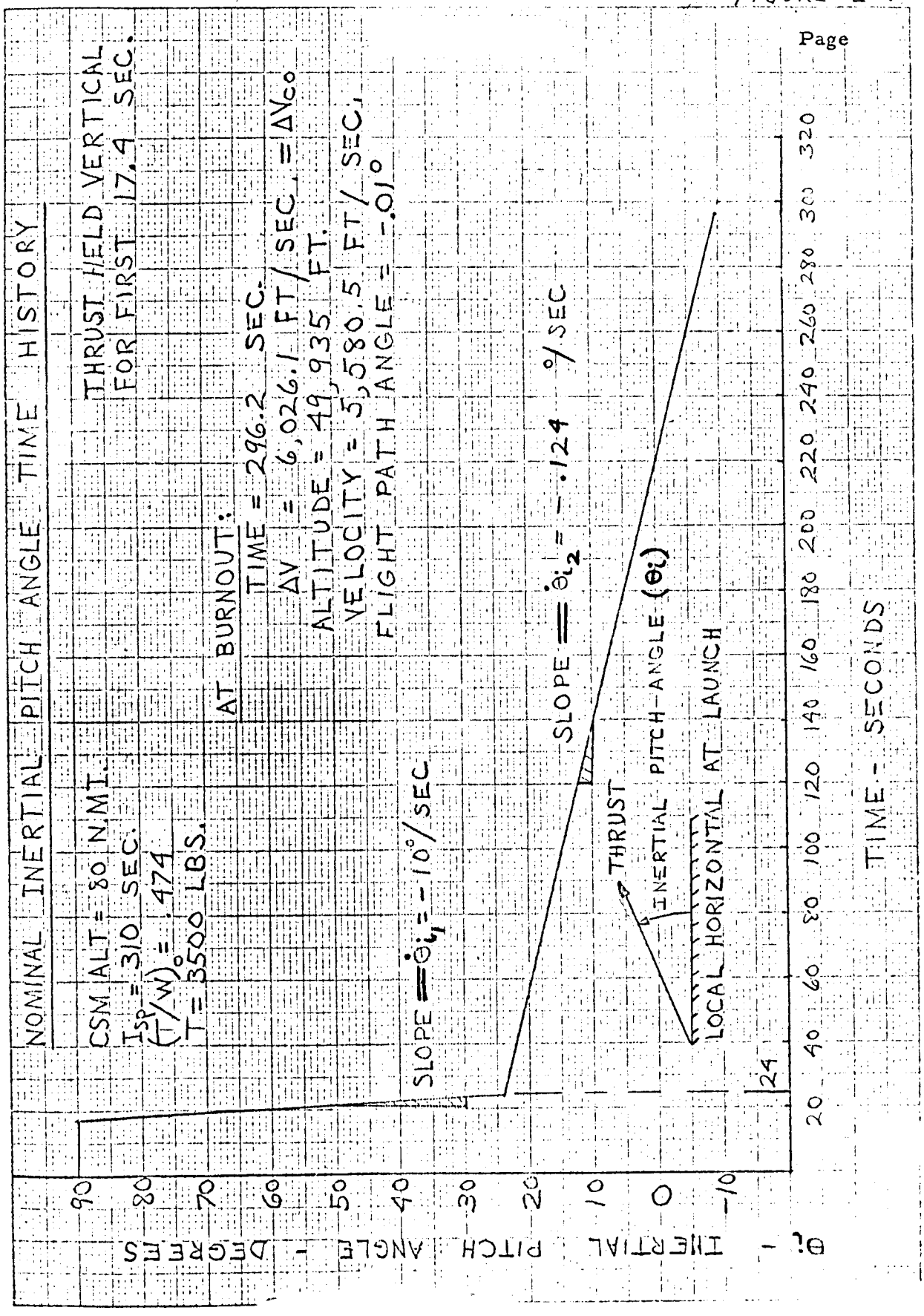

NOMINAL INERTIAL PITCH ANGLE TIME HISTORY
CSM ALT = 80 N.MI.
I_{sp} = 310 SEC.
(T/W)$_o$ = .474
T = 3500 LBS.
THRUST HELD VERTICAL FOR FIRST 17.4 SEC.
AT BURNOUT:
TIME = 296.2 SEC.
ΔV = 6,026.1 FT/SEC. = ΔV$_{co}$
ALTITUDE = 49,935 FT.
VELOCITY = 5,580.5 FT/SEC.
FLIGHT PATH ANGLE = -.01°
SLOPE = $\dot{\theta}_{i_1}$ = -10°/SEC
SLOPE = $\dot{\theta}_{i_2}$ = -.124 °/SEC
THRUST
INERTIAL PITCH-ANGLE (θ_i)
LOCAL HORIZONTAL AT LAUNCH
θ_i - INERTIAL PITCH ANGLE - DEGREES
90 80 70 60 50 40 30 20 10 0 -10
TIME - SECONDS
20 40 60 80 100 120 140 160 180 200 220 240 260 280 300 320
24

FIGURE E-8

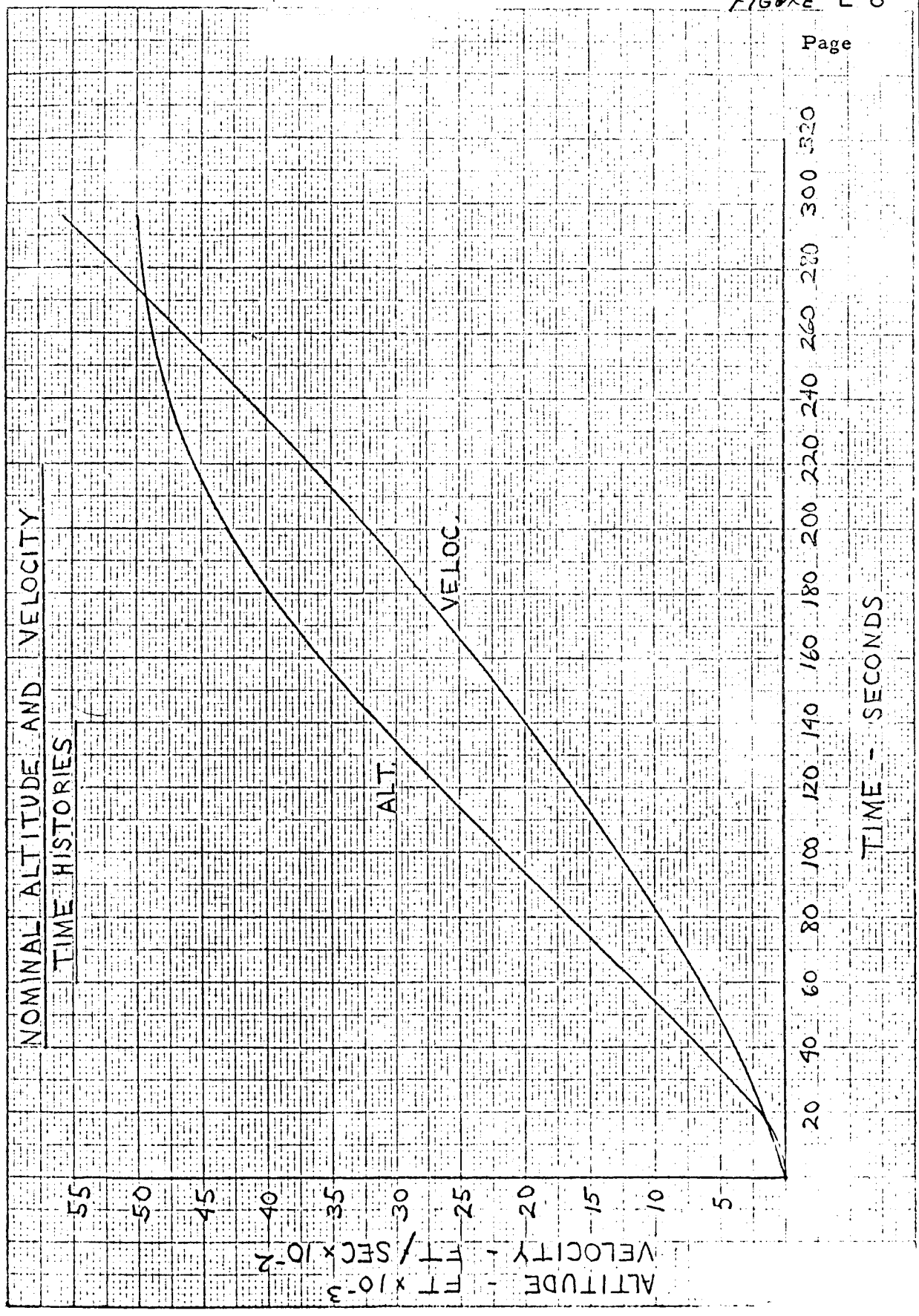
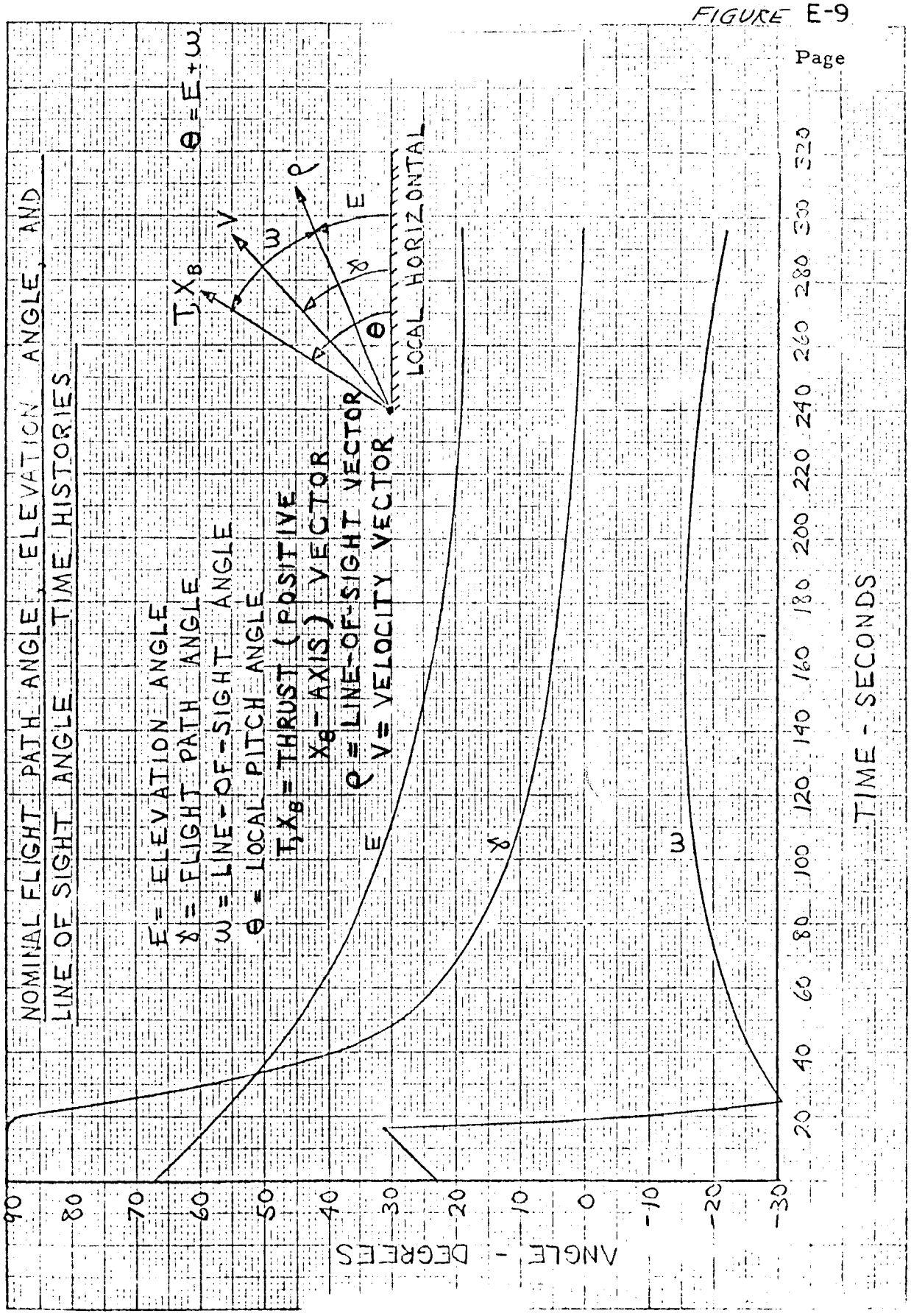

Page
NOMINAL ALTITUDE AND VELOCITY
TIME HISTORIES
VELOC.
ALT.
TIME - SECONDS
20 40 60 80 100 120 140 160 180 200 220 240 260 280 300 320
ALTITUDE - FT x 10⁻³
VELOCITY - FT/SEC x 10⁻²
55 50 45 40 35 30 25 20 15 10 5

FIGURE E-9


NOMINAL FLIGHT PATH ANGLE, ELEVATION ANGLE, AND LINE OF SIGHT ANGLE TIME HISTORIES
E = ELEVATION ANGLE
γ = FLIGHT PATH ANGLE
ω = LINE-OF-SIGHT ANGLE
θ = LOCAL PITCH ANGLE
T/X_B = THRUST (POSITIVE X_B - AXIS) VECTOR
ρ = LINE-OF-SIGHT VECTOR
V = VELOCITY VECTOR
θ = E + ω
T/X_B
V
ω
ρ
E
γ
θ
LOCAL HORIZONTAL
E
γ
ω
ANGLE - DEGREES
90
80
70
60
50
40
30
20
10
0
-10
-20
-30
20
40
60
80
100
120
140
160
180
200
220
240
260
280
300
320
TIME - SECONDS

2. a somewhat greater direct insertion early launch window is available, with a ΔV penalty of 100 ft/sec for an 80 second early launch;

3. if the launch occurs at a time not within the lunar launch window, or if the ΔV penalty for a small launch time error is too high, some time must be spent in a low altitude parking orbit;

4. every minute lateness of launch requires approximately 8.6 minutes in a 50,000 ft. parking orbit to regain proper phasing;

5. if a launch is made more than 12.8 minutes late, more time would be spent in a parking orbit than would be spent waiting on the moon for the next launch opportunity;

6. a maximum parking orbit time of 17.7 hours would result from launching a few moments too early.

Out-of-plane Launch

It is a requirement that it be possible to perform an ascent and rendezvous from a lunar launch site which is up to 2 degrees out of the CSM orbital plane. In general, plane change requirements do not affect the shape of the ascent trajectory, since these changes are either made during the coasting ascent, or combined with the rendezvous phase. However, whether or not the lunar launch site is in plane with the CSM orbit, it will be necessary to determine the azimuth heading which defines the plane in which the ascent trajectory lies. When the launch site is in the CSM plane, the CSM will pass directly overhead. The ascent plane is defined by the launch site local vertical and an azimuth heading parallel to the plane of the CSM. In this case, no plane change maneuvers are required. When the launch site is not in the CSM orbit plane, it is impossible to launch directly into the CSM plane. The LEM orbit plane having the least inclination to the CSM plane obtainable is defined by the launch site vertical and an azimuth heading parallel to the heading of the CSM at its zenith relative to the launch site. The inclination of this plane to the CSM plane is equal to the displacement of the launch site from the CSM plane in lunar central angle. The intersection (line of nodes) between this LEM "least inclination" plane and the CSM plane occurs 90 degrees from the launch site. The plane change must be made at the line of nodes and will require a ΔV of approximately 100 ft/sec per degree of plane change. There is a possiblity of reducing this ΔV penalty somewhat by combining the plane change maneuver with either the injection from a low altitude parking to the transfer orbit or with the rendezvous maneuver. With the first method, the launch is delayed about 3.5 minutes, and the LEM is inserted into a 50,000 ft. parking orbit; it then coasts in the parking orbit for 90 degrees of central angle, and performs the plane change and insertion into the transfer orbit. Since the ΔV's for the two maneuvers are perpendicular to each other their vector sum is less than the sum of their magnitudes, and a total ΔV saving will result.

For a 2 degree plane change, the plane change and transfer would require 300 ft/sec if done individually and 224 ft/sec if combined. If it is preferred to combine the plane change with rendezvous, the same procedure of adjusting launch time and coasting in a parking orbit so that rendezvous occurs on the line of nodes may be used. If the maneuvers are not combined it is not necessary to use a parking orbit, since the plane change may be made during coasting ascent (again, at or near the line of nodes).

Another method under consideration does not require a parking orbit, but has the plane change taking place at rendezvous. In order to make the rendezvous point coincide with the line of nodes, both the locations of the line of nodes and the point of rendezvous must be altered. The position of the line of nodes is changed by launching in a plane which is at a greater inclination to the CSM orbit than the "least inclination" orbit and the position of the rendezvous point is altered by taking a non-Hohmann transfer. It appears however that the ΔV penalties resulting from the increase in the angle between the orbital planes more than offset the saving due to combining the maneuvers.

In general, the method of plane change during coasting ascent to CSM orbit altitude is preferred for three reasons. First, it permits maintenance of the nominal ascent trajectory and supercircular burn-out velocity, with the attendant decrease in sensitivity of pericynthion altitude to burn-out errors. Second, it allows the use of a midcourse correction technique which combines both in-plane and out-of-plane adjustments in a single maneuver and thus results in considerable reduction in the complexity of the guidance computations. Third, delayed out-of-plane launch can be handled in the same manner as a delayed in-plane launch, that is by commanding burn-out at circular orbital velocity and then coasting in the parking orbit for a time interval the length of which is a simple function of the launch delay.

Further Trajectory Studies for Application to Back-up Guidance for Powered Descent

Present trajectory studies on powered descent have the following objectives:

1. to formulate guidance laws for the back-up guidance mode which should be compatible with the guidance scheme to be used in the primary system;

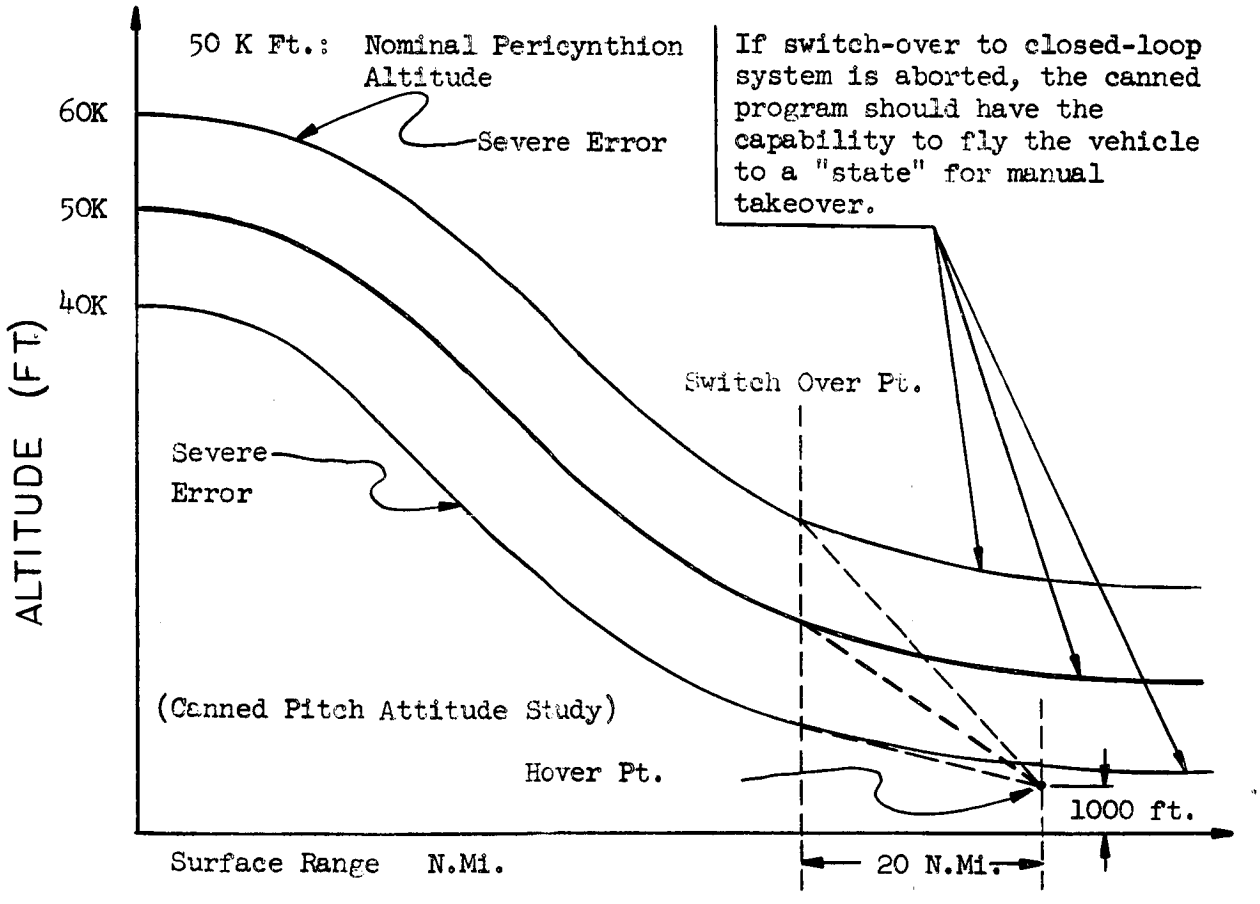
2. the guidance schemes developed for "possible" descent back-up guidance should require only the information on attitude, position and velocity which is available from the back-up guidance instrumentation and sensors;

3. the guidance laws formulated should not require an extensive amount of computation, i.e. a small computer, either digital or analog should suffice;

4. in the event of total loss of on-board computer facilities, it is desirable that a "canned" program be available to the crew. This "open-loop" system should be chosen to shape a trajectory, which in the case of severe error would fly the vehicle to safe altitudes at velocities which permit manual take-over.

The powered descent trajectory is broken down into four phases 1) initial powered descent, 2) final powered descent, 3) flare and hover, and 4) landing.

1. The initial phase of powered descent is tentatively planned to be an "open-loop" guidance law with the thrust held constant and pitch attitude programmed as a linear function of time. This phase of the trajectory is terminated at approximately 20 n.mi. from the landing site which is the best point for a switch-over to a closed-loop system for the final powered descent phase. The "canned" program chosen for this initial phase of the trajectory should, under severe error conditions, result in safe altitudes at the "switch-over" point and if "switch-over" to the closed loop system is aborted, the pitch program chosen for the initial phase should fly the vehicle to safe altitudes above the landing area at a speed amenable to pilot take over (see figure below).



BACK-UP GUIDANCE DESCENT TRAJECTORY

FIGURE E-10

The reference axis system for the initial phase of the trajectory, and in fact, the entire trajectory should be an inertial system fixed at the landing site (see figure below).

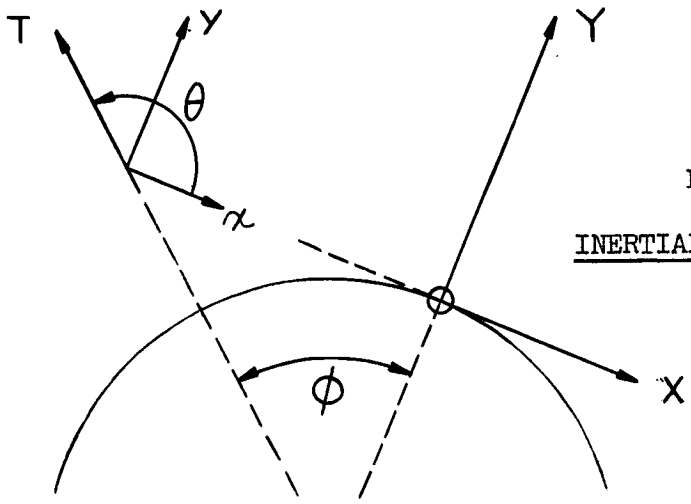


FIGURE E-11

INERTIAL REFERENCE SYSTEM

2. The "final powered descent" is tentatively planned to be a closed-loop system which utilizes radar information to yield altitude and velocity components in the reference system discussed above. This information is sufficient to yield a value for acceleration and pitch attitude which will fly the vehicle from initiation to hover altitude with a residual horizontal velocity of specified magnitude. In a linear coordinate system, the equations for final altitude and velocity, assuming constant acceleration are as follows:

$$h_f = h_o + \dot{h}_o t + \frac{1}{2}(a_h - gm)\, t^2 \qquad (1)$$

$$\dot{h}_f = \dot{h}_o + \frac{1}{2}(a_h - gm)\, t \qquad (2)$$

but if $h_f = 0$; then the time of flight remaining becomes

$$t = \frac{-\dot{h}o}{(a_h - gm)} \qquad (3)$$

Substituting (3) into (1) yields an expression for a_h, the required constant acceleration.

$$a_h = \frac{-(\dot{h}o)^2}{2\,(h_f - ho)} + gm \qquad (4)$$

The component of acceleration in the downrange direction is evaluated from the vehicle present downrange velocity, the

required final residual horizontal velocity and the time of flight calculated in equation (3)

$$\dot{x}_f = \dot{x}_o + a_x t$$

$$a_x = \frac{\dot{x}_f - \dot{x}_o}{t} = \frac{-(a_h - g_m)(\dot{x}_f - \dot{x}_o)}{\dot{h}_o} \qquad (5)$$

Substituting equation (4) into equation (5) yields

$$a_x = \frac{h_o\ (\dot{x}_f - \dot{x}_o)}{2\ (h_f - h_o)} \qquad (6)$$

From the calculated values of a_h and a_x, the magnitude and direction of the thrust vector can be determined

$$T = m\ (a_x^{\ 2} + a_h^{\ 2})^{\frac{1}{2}} \qquad (7)$$

$$\theta = \tan^{-1} \frac{a_h}{a_x} \qquad (8)$$

$$a_T = (a_x^{\ 2} + a_n^{\ 2})^{\frac{1}{2}} \qquad (9)$$

a_T = total acceleration

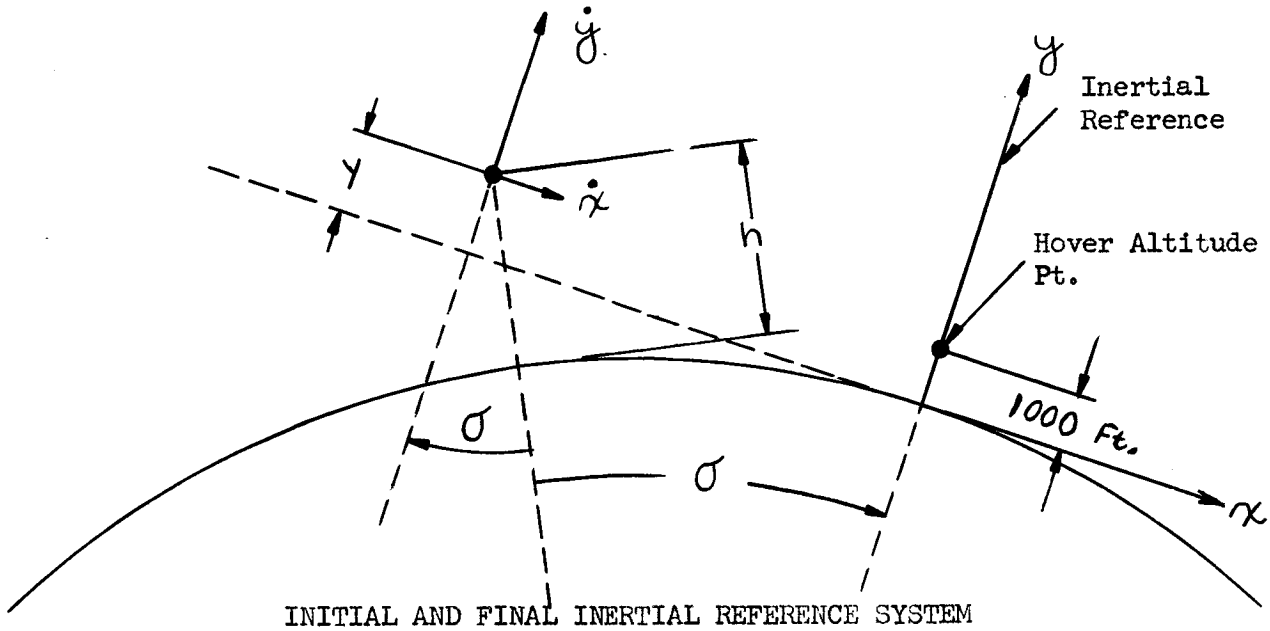
Equations (8) and (9) yield values for total acceleration and pitch attitude which will be constant throughout the trajectory under the following assumptions:

a. flat moon

b. constant gravity potential

If these assumptions are valid, a measurement of altitude and velocity components at the switch point is sufficient information to set up the values of total acceleration, a_{t1} and pitch attitude, θ, which holds constant throughout the remainder of the trajectory which flies the vehicle to the proper hover altitude and at residual horizontal velocity.

However, certain adjustments must be made when the actual environment is encountered. If all measured quantities (altitude and velocity) are transformed into the inertial reference system (see Figure E-12) and an average value of the gravity term gm is assumed, the single calculation is sufficient to set up the command signals for the remainder of the powered trajectory.



INITIAL AND FINAL INERTIAL REFERENCE SYSTEM
FIGURE E-12

From Figure E-12, the expression for the inertial altitude Y, can be derived

$$Y = (h + r_m) \cos(\sigma - r_m);$$ where

r_m is the radius of the moon or the predicted radial altitude of the landing spot; σ is the predicted central angle between LEM and the landing-site. The value σ can be determined by visual landmarks along the trajectory. The average value of gm can be calculated for the expression

$$g \text{ average} = \frac{gm}{2}\left(\frac{r^2}{r_m^2} + 1\right);$$

where r is the present LEM radial position which in terms of available information becomes

$$r = (r_m + h)$$

The calculations in this procedure are relatively simple, but possibly too large for speedy manual calculation.

Another approach is to calculate continuously the commanded acceleration, a_t, and pitch attitude, θ, using equations (4), (6), (8) and (9). As a result, direct readings of altitude, h, rather than the transformed quantity, Y, is used. This approach relies on the fact that $h \to Y$ as $\sigma \to o$. The arguments are strong for this approach. First, the uncertainty in both the known measurement of the radius of the moon r_m, and the gravity term, g_m, is large enough to negate the calculation law in favor of continuous technique. Secondly, the unpredictable variation in the lunar terrain might require the use of h rather than Y for safety reasons alone.

F. SYSTEMS STUDIES OF FEASIBLE GUIDANCE TECHNIQUES

1. Guidance Concepts for Ascent and Abort

General ascent or abort trajectories of the type considered here are characterized by a relatively short thrusting period during which approximately 95% of the total integrated specific force is applied, a coasting period of a duration that depends upon the specific requirements of the trajectory, and a smaller, final thrusting period to inject the vehicle into the desired orbit. The initial portion of this profile has been defined as the powered ascent or abort from powered descent, resulting in orbital conditions being achieved at burn-out. The coast phase duration is a function of the particular situation as discussed previously in Section E , and the final thrusting period is required only if the orbit has to be changed. (An example of this is the injection of the LEM from a low altitude parking orbit into a transfer orbit to rendezvous with the CSM). Corrections to the trajectory, when required for rendezvous, are classified as midcourse corrections and are considered separately from the ascent (or abort) phase. The rendezvous phase includes correction to the trajectory of the LEM in order to intercept the CSM as well as the ΔV required to adapt the LEM transfer orbit to that of the CSM.

2. Open-loop Attitude - Thrust Program

The nominal powered ascent or abort trajectories at any specific set of initial conditions can be described by attitude and thrust time profiles. This prescribes the direction and magnitude of the thrust vector at every instant and therefore establishes the value of the total integrated specific force. The nominal profile is one that has been developed with respect to desired burn-out conditions, minimum ΔV and minimum error sensitivity.

For such a nominal trajectory and for a constant thrust engine, it is possible to store, with relatively simple mechanization, the desired attitude vs. time and thrust cut-off time. A stored program to command the vehicle attitude and cut-off represents a simple guidance scheme, but is limited in performance since it cannot compensate for the effects of deviations of the actual from the desired trajectory. The actual trajectory will deviate from the nominal one because of errors induced by these conditions:

- Variations in thrust magnitude
- Engine misalignment
- Attitude reference inaccuracies
- Variations in vehicle initial weight
- Timing errors in cut-off
- Deviations in attitude control system fuel expenditure as a result of variation in LEM c.g. location.

Expected values of these errors produce burnout errors larger than can be tolerated in either ascent from the lunar surface or abort from powered descent.[1] The criterion used to establish the acceptable burn-out error limits is derived from the more stringent of either the requirement that the resultant orbit have a clear pericynthion and also be one with which the CSM can rendezvous and from which rescue of the LEM is possible, or that, with midcourse correction, the LEM can rendezvous with the CSM.

3. Simple Compensation Schemes

ΔV Cut-off

The Δ V cut-off scheme is designed to provide compensation for errors due to off-nominal engine thrust magnitude when following a stored attitude-thrust time profile as described above. With this scheme, the programmed attitude-time profile is followed as before, but instead of terminating the thrust phase at the nominal time, the accumulated ΔV is measured by an integrating accelerometer oriented along the vehicle thrust axis is used to determine the proper instant for cut-off.

Since there is not a one-to-one relationship between velocity and ΔV in this situation, the ΔV cut-off technique provides only partial correction for thrust magnitude errors. The magnitude of velocity error at burn-out is reduced; however, the error in direction of the velocity vector (which determines the flight path angle) remains uncorrected.

Attitude Profile vs. Measured ΔV

The next logical step in providing a simple compensation scheme for the errors resulting from an open-loop, stored-attitude profile guidance technique is to pre-program the attitude commands as function of the ΔV achieved. As before, the integrating accelerometer oriented along the vehicle's thrust axis measures the value of ΔV at any instant. The stored guidance program must then either command variable body rates or a continually changing pitch angle as a function of ΔV. This technique attempts to limit the two in-plane components of velocity simultaneously. Two additional accelerometers, oriented along the body axes orthogonal to the nominal thrust axis are used together with the vehicle attitude control system to reduce errors due to misalignment of the engine thrust vector.

In a gravity-free environment, this technique can accurately compensate for thrust errors. However, in a gravity field, a significant source of error remains uncompensated by this simple ΔV guidance scheme. The ΔV program, when used to compensate for an off-nominal thrust magnitude, results in a correspondingly off-nominal burn duration. As a consequence, gravity will act on the LEM for varying time intervals and introduce an error in vertical velocity and hence in-flight path angle which a simple ΔV measurement technique cannot detect or correct. For example, a thrust deviation of 100 lbs. off-nominal would result in a deviation of burn time of approximately 8 seconds. This produces an error in vertical velocity of about 40 fps or a flight path angle error of 0.3 degrees which is still considerably larger than desired.

[1] Reference. LEM Engineering Memo L540-M03-26 "LEM Guidance Burn-out Errors for Ascent from the Lunar Surface".

The "t_1 Compensation" Technique

The "t_1 compensation" technique can compensate for the major errors remaining in the basic attitude profile vs. measured ΔV guidance scheme. In this technique, an approximate determination of the thrust deviation from the nominal value is obtained and used to compute the compensation required for the vertical velocity error resulting from the deviation from the nominal burn time. This computation is performed by a simple linear combination of measured data obtained by comparing the actual time required to reach a prescribed ΔV with the nominal time at which the ΔV value has been expected. The effects of thrust misalignment are again compensated for by the use of two cross-axis oriented accelerometers.

The basic principle of the "t_1 compensation" scheme is that for a constant thrust engine as a first order approximation the vehicle can be considered to have a linearly increasing acceleration with time over a short thrusting interval.

Errors in vehicle weight and in thrust magnitude can be related to the deviation of the measured value of acceleration at some time T_1 from the nominal value expected. Error in rate of change of thrust (caused by either off-nominal Isp or mass flow rate) can similarly be related to deviations in rate of change of acceleration. It takes two measurements (one of time and one of ΔV achieved) to establish the required error function. Having this function, the duration of powered flight can now be predicted and the vertical and horizontal velocity errors expressed as linear combinations of these two measurements.

The vertical velocity error is compensated by computing and making a small pitch angle adjustment to the nominal pitch profile. The horizontal velocity error is compensated by adjusting the cut-off ΔV appropriately. The values of these compensation terms can also be derived as linear combinations of the two measurements mentioned above.

The problem of engine misalignment is handled by measuring simultaneously the accumulated ΔV on the three body-mounted accelerometers as early in the powered flight as possible to permit adequate resolution of measured quantities. Comparison of the ΔV_x, ΔV_y, ΔV_z terms defines the misalignment of the thrust vector from the thrust axis (x body axis). The ratios $\frac{\Delta V_y}{\Delta V_x}$ and $\frac{\Delta V_z}{\Delta V_x}$ are proportional to the misalignment. The ΔV_x term can be approximated by the expected nominal ΔV to simplify the mechanization of the required computation.

These misalignment terms then provide the proper error signals to the vehicle's attitude control system to correct the thrust direction and prevent any further buildup in misalignment velocity error.

Delta Guidance for Midcourse Corrections

The Delta Guidance technique is applicable to the determination of corrections to the midcourse coast phase of the LEM ascent. Trajectory deviation due to errors in injection into the transfer orbit must be

corrected in order to ensure that rendezvous can be accomplished within the ΔV limitations. In order to reduce the computational requirements associated with an explicit solution to the guidance equations, the Delta technique makes use of a Taylor series expansion about a nominal trajectory. The expansion points on the nominal trajectory correspond to observation times when appropriate radar measurements of actual LEM/CSM relative position and velocity in a convenient coordinate system are made. Comparing these values with the expected nominal ones produce the Delta terms or residuals which then can be used to determine the corrective action to be taken at the proper time. The six different radar measurements that may be made at any observation time are:

- Range
- Range Rate
- In-Plane LOS Angle
- Out-of-Plane LOS Angle
- In-Plane LOS Angle Rate
- Out-of-Plane LOS Angle Range

While these six measurements are sufficient to define the corrections at any point, some types of radar data may be more accurate than others and therefore three measurements, made at two observation points result in a more accurate determination of the necessary corrections. The analytical studies of this technique, described in the following section, investigate various combinations of these measurements. The procedure for application of the Delta technique is to precompute the necessary coefficients with respect to the nominal trajectory. One set of coefficients then applies to the particular nominal trajectory and set of measurement times chosen. Once it has been determined that these coefficients are adequate for the accuracies required, it is necessary to investigate the range of off-nominal conditions for which allowance must be made. Thus the number of sets of coefficients which must be computed to cover the field of possible ascent trajectories can be determined. These off-nominal ascent trajectories can arise from an abort at different times from the powered descent or the lunar surface.

Explicit Guidance for Midcourse Corrections

An explicit guidance technique uses directly the free flight equations of motion to determine the guidance commands necessary to intercept the CSM. Given the present positions of the LEM and CSM the trajectories that the two vehicles will traverse in the presence of the gravitational field of the moon can be computed and initial conditions for the LEM can be specified so that its trajectory will intercept that of the CSM. The explicit technique can compute all corrections in flight with any initial conditions and is therefore not limited to a nominal trajectory or a particular correction time. Thus the explicit technique can be used not only for midcourse corrections from any off-nominal ascent, but also to provide the ascent guidance itself from any abort situation. However, the exact mechanization of this technique requires considerably more flight equipment than is feasible

for a back-up guidance system. For this reason, various simplifications have been examined to determine if a minimal configuration could be found that would still provide the necessary accuracy.

The simplifying assumptions made for the purposes of this study were:

- The CSM is in a circular orbit.
- The gravitational field in the vicinity of the CSM is linear with altitude.

These two approximations yield results that become increasingly accurate as the LEM approaches the CSM. As with the Delta guidance scheme, radar measurements are made of CSM to LEM range, range rate, line of sight, and line of sight rate.

The concept of the simplified technique then, results in the following:

- The reference coordinate system is CSM centered, and is aligned with respect to the CSM local vertical.
- The navigational data inputs of position and velocity are provided by the LEM rendezvous radar and the attitude reference system.
- The time of flight is a variable which may be selected by the operator.
- The impulsive velocity to be gained is computed and resolved into a command attitude angle and a thrust duration and magnitude.

The performance and accuracy requirements as well as a proposed hardware configuration for this guidance technique are described in Section F-4.

Homing Guidance for Rendezvous

The homing guidance technique proposed for the back-up guidance system in the rendezvous phase makes use of either the LEM or CSM rendezvous radar to provide LEM/CSM relative information. The basic feature of this technique is that the LEM essentially flies a collision course to the CSM. This characteristic is achieved by keeping the inertial rate of the line-of-sight (LOS) to the CSM below a given threshold value, while at the same time reducing the range rate step by step as a function of range-to-go. The basic procedure consists of a sequence of operational steps:

1. Range to the CSM is measured continually during the coasting ascent flight. At a given range from the CSM, thrust is applied along the LOS in a direction that will result in a range rate within predetermined bounds appropriate to the range at which thrust is initiated.

2. Following range rate reduction, LOS rate to the CSM is measured and thrust is applied normal to the LOS to reduce this rate to the threshold level.

3. The LEM is allowed to coast until the next range check point is reached. Steps 1 and 2 are then repeated.

4. This procedure is continued through a number of range check points until final docking range and near zero relative velocity are attained.

The attitude maneuvers required of the LEM as it proceeds through the above steps are as follows:

1. With the rendezvous radar locked on and tracking the CSM, the LEM attitude is adjusted to null the radar antenna gimbal angles. This results in the LEM Z-axis being directed along the LOS to the CSM and gives the crew direct CSM visibility through the forward cabin windows.

2. Range rate reduction to the value, commensurate with the range at which the correction is made, is performed by the Z-axis RCS engines, with the Z-axis aligned to the LOS to the CSM.

3. The direction of the normal component of the relative velocity vector is established from measurement of the inertial rates of the antenna gimbals. If the gimbal angles are aligned parallel to the LEM X and Y axes, then inertial gimbal rates are directly proportional to components of the relative inertial velocity vector along the body X and Y axes. The LEM is rotated about the Z-axis until one of the gimbal rates reaches the measurement threshold value. As a result, the body axis corresponding to the gimbal axis is now aligned with the net normal component of relative inertial velocity. Thrust along that axis is now applied to null the indicated LOS rate, and thus eliminate the normal velocity component. The RCS engines are used in this phase.

The concept of multiple thrust phases rather than continuous control has been adapted for several reasons. For one, the range versus range rate regime for multiple thrusts is a simpler one than for continuous thrust, and lends itself more readily toward a display presentation that the crew can follow in either a minitoring role or a manual mode. Furthermore, during some back-up modes involving LEM radar failure, several of the operations required for successful rendezvous must be performed manually by the crew and the coasting time between thrust application allows this to be done.

4. Results of Guidance Method Analyses

a. Delta Midcourse Guidance

A differential guidance technique has been studied to determine the feasibility of performing the midcourse correction during coasting ascent; the following factors have been considered:

1. Range of ascent burn-out errors which can be satisfactorily compensated by a single corrective impulse.

2. Increase in these allowable burnout errors when two corrections are permitted.

3. Optimum time of application of the single or double correction when considering miss distance and ΔV requirements

4. Selection of optimum radar parameters to be used for measurements when trajectory characteristics and radar errors are considered.

Principle of Differential Correction Technique

The differential midcourse correction (DMC) technique establishes an actual orbit by measuring differences between actual and nominal values of observed data. These differences result in terms called observational residuals. They can be expressed in terms of a first order Taylor's expansion of the six orbital parameters. Since this is a point mass, three-dimensional analysis, six orbital parameters are involved. This relationship[2] is expressed as follows:

$$\Delta P_1 = \frac{\delta P_1}{\delta \rho_1} \Delta \rho_1 + \text{----} + \frac{\delta P_1}{\delta \rho_6} \Delta \rho_6$$

$$\vdots$$

$$\Delta P_6 = \frac{\delta P_6}{\delta \rho_6} \Delta \rho_1 + \text{----} + \frac{\delta P_6}{\delta \rho_6} \Delta \rho_6$$

where

ΔP = observational residuals

$\frac{\delta P}{\delta \rho}$ = partials relating the observational residuals to the orbital parameter

ρ = orbital parameter

$\Delta \rho$ = differences in the orbital parameters of the actual and nominal orbits.

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[2] Ref. S. Herrick, "Astrodynamics," D. Van Nostrand Co. Princeton, N.J., 1961. R.M. Baker & M.W. Makemson, "An Introduction to Astrodynamics", Academic Press, N.Y. 1960, pp. 142-152

Through inversion of these equations, the observational residuals can determine the orbital parameters, and with them off-nominal position and velocity may be determined at the time of midcourse correction. The data will be used to determine the midcourse correction velocity impulse required for an intercept orbit. The steps previously described can be combined so that the required midcourse correction can be computed directly with the observational residuals.

Procedures and Assumptions

1. In the differential midcourse correction method, the deviation from a precomputed reference trajectory is determined by radar measurements. Since the measurement is used to determine the trajectory deviation, rather than the trajectory itself, and since the deviations are expected to be small, linearization of the dynamical equations of motion is permissable. This linearization greatly simplifies the guidance law, but restricts operation to trajectories which do not deviate excessively from the reference trajectory. As stated previously, one of the objectives of this study is to determine the range of validity of the linearization assumption.

2. It has been established (Ref. LMO-500-22) that the rendezvous maneuver can be started at ranges up to 20 n.m. with small ΔV penalty, for trajectories, the uncompensated miss distance of which is less than 20 n.m. In order to keep the rendezvous fuel requirements within the ΔV budget, "acceptable" midcourse corrections were limited to those providing a miss distance of 10 n.m. or less.

3. The method of analysis assumes that the observational residuals, which are the differences between the precomputed and observed radar data, determine the injection errors which have occurred. These residuals are multiplied by computed constants in order to determine the velocity corrections required to reduce the miss distance. The trajectory which results from application of the velocity corrections is referred to as a mean trajectory. It is not a nominal trajectory because mechanization of the Delta guidance law neglects higher order terms and cross coupling effects as part of the linearization process. The resulting miss distance can be considered a bias error.

4. In addition to this geometrical bias error, errors in the radar measurements will affect the observational residuals and thereby affect the corrective velocity impulses. For random measurement errors, the trajectories which result from the erroneous corrections are treated in a statistical sense and represent a dispersion of trajectories about the mean trajectory. To determine the range of this dispersion, the covariance error matrix of the midcourse velocity correction due to errors in radar measurements must be determined. The diagonal elements represent the variance of each applied velocity component, and the off-diagonal elements indicate the statistical covariance. The

RSS of the component one sigma velocity errors is used to indicate the magnitude of the velocity correction which might be due to the random radar errors. A velocity correction has only been applied if the correction is greater than that corresponding to the random radar errrors.

For those trajectories in which a midcourse correction has been applied, the ΔV required has been computed by taking the RSS of the mean correction (corresponding to actual injection errors and linearization effects) and the correction caused by the random radar errors.

5. The covariance error matrix of the midcourse velocity correction is also used to determine the covariance miss distance matrix at the nominal time of intercept. The three diagonal elements represent the variance of each component miss distance. From the three components of variance, an RSS miss distance is computed which indicates the magnitude of the random effects. This in effect neglects the covariance terms of the distribution. Strict interpretation of the miss distance in terms of probability requires integration of the trivariant distribution as defined by the covariance error matrix. For this preliminary investigation, however, the RSS is adequate for comparative purposes, and for determination of order of magnitude effects of the random errors.

6. It has been assumed that larger insertion errors can be tolerated if two midcourse corrections, rather than one, are permitted. The intent of the first correction is to maintain the LEM near the reference trajectory when large insertion errors occur. The second correction is then used to achieve the desired intercept. It is assumed that trajectory errors due to the random errors of the first correction are corrected by the second. The random miss distance at intercept is thus a result of the radar measurements used to determine the second correction only. The total RSS ΔV for the second correction, however, includes the random errors of the first correction propagated to a velocity correction at the time of the second midcourse (in addition to random radar effects and geometrical effects of the second correction measurements). The total ΔV required for the midcourse corrective maneuver is the arithmetic sum of the ΔV required for each correction.

7. Since there are six orbital parameter differentials to be determined, a minimum of six observational residuals are required to solve for the unknowns. The necessary information was obtained by making three radar measurements at two separate times. For the purposes of this study, the radar parameters used for the "in-plane" measurements are combinations of range, ρ, range rate, $\dot{\rho}$, and line of sight angle, θ_ρ. The "out-of-plane" conditions are determined by "out-of-plane" angle θ_ψ measurements. Angular rates of "in-plane" and "out-of-plane" angles have not been included because preliminary investigation indicates that the

orbital parameter determination is highly sensitive to angular rate measurement errors.

For each off-nominal trajectory, the characteristics of the three combinations of radar parameters $(\rho, \theta\rho, \theta\Psi)(\dot{\rho}, \theta\dot{\rho}, \theta\Psi)(\rho, \dot{\rho}, \theta\Psi)$ are computed to determine which set gives the best results.

8. The effect of radar measurement errors lessens as the time of final observation is delayed. Previous investigations (Ref. L500-M03-9), incidentally, have indicated that the midcourse correction of ΔV required to eliminate insertion errors becomes excessive when the applied corrections are delayed much beyond the mid-time of flight. Therefore, for this preliminary investigation, the time of making the second midcourse correction is set to equal the mid-time. To obtain comparison between one and two midcourse corrections, the time of the single correction has also been set at mid-time of flight.

   For the two correction case, a reasonable time for the first correction cannot be predetermined and it therefore becomes a parameter of study. For each set of insertion errors, the time of the first correction is increased in four steps to a maximum time of one quarter of the total flight time. The first radar measurement (t_1) is assumed to occur 100 seconds after burn-out and the time of the second measurement (t_2) has been allowed to vary. The first correction occurs at T_1, 50 seconds after t_2. The third measurement time (t_3) occurs 50 seconds after the first DMC (T_1). The fourth measurement is fixed and occurs 50 seconds prior to the final DMC, which has been set at mid-time of flight. The observation times for the single DMC case are (t_1) and (t_4), or 100 seconds after burn-out and at mid-time.

9. The results of the study are dependent upon the reference ascent trajectory (assumed nominal) chosen. The bounds of ascent trajectories being considered are 140° and 240° central angle intercepts by the LEM vehicle. Therefore these two reference trajectories are chosen for the initial investigations. The 140° trajectory might simulate an ascent from the lunar surface, while the 240° trajectory might simulate an abort reference trajectory. Should the differential midcourse correction technique prove to be acceptable for the 140° and 240° trajectories, adequate performance would probably be achieved with other trajectories within this band. Future analyses will, of course, be performed to confirm this.

10. Although the radar errors previously discussed are considered to be random, it is also necessary to consider the effect of bias errors and drift errors in the measurements. This type of error will appear as a bias error at the time of intercept, in addition to the previously discussed geometric error caused by the off-nominal conditions. The effects of attitude reference drift and alignment errors must be determined to specify

the attitude reference performance required for the midcourse correction.

The attitude reference will introduce bias and drift errors in the angular observational residuals. In order to determine these effects independently of the geometric effects, the insertion errors are assumed to be zero and the bias errors have been added to the observations.

11. The equations used to generate trajectories, the partials relating the observational residuals to orbital parameters, and the partials used to obtain the error propagation coefficients, were based on an assumed perfectly circular orbit of the CSM, and a linear expansion of gravity in the vicinity of the CSM. The coordinate system used to define the trajectory characteristics is a CSM centered local vertical coordinate system defined in Figure F-1 below.

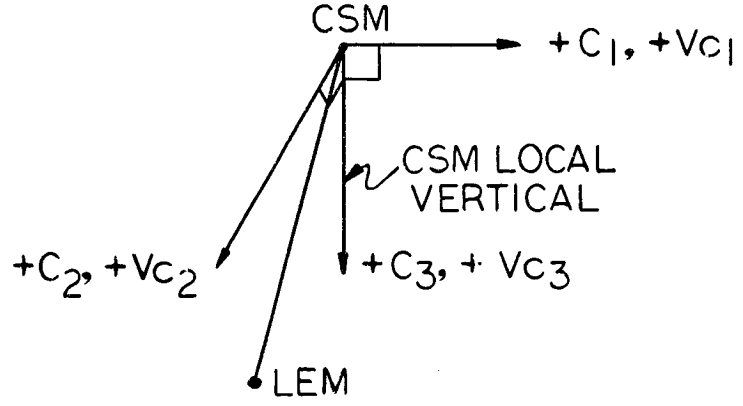

FIGURE F-1

CSM CENTERED LOCAL VERTICAL COORDINATE SYSTEM

Reference Data

1. The one sigma random errors assumed are:

Range - 0.3%

Range rate - 0.3%

Angle error - 0.3 degrees

A parametric study of radar errors and their effects on fuel and miss distances has not been performed since the rendezvous radar performance is specified in an agreement with MIT.

2. The 240° reference trajectory, which is representative of an abort from descent, is defined by the following nominal relative position and velocity at insertion in CSM coordinates.

$C_1 = 0$ $C_2 = 0$ $C_3 = 436{,}110$ ft.

$Vc_1 = 696.05$ ft/sec $Vc_2 = 0$ $Vc_3 = -45.87$ ft/sec

Nominal time of intercept - 4905 sec.

Radar measurement times

$t_1 = 100$

$t_2 = t_1 + (275)\,u \quad u = 1, 2, 3, 4$

$t_3 = t_2 + 100$

$t_4 = 2400$

3. The 140° reference trajectory selected to simulate a lunar launch is defined by the following nominal conditions at insertion.

$C_1 = -98{,}310$ ft. $C_2 = 60{,}760$ ft. $C_3 = 436{,}110$ ft.

$Vc_1 = 671.83$ ft/sec $Vc_2 = 43.316$ ft/sec $Vc_3 = 0.0446$ ft/sec

Nominal time of intercept - 2654 sec.

Radar measurement times

$t_1 = 100$

$t_2 = t_1 + (150)\,u \quad u = 1, 2, 3, 4$

$t_3 = t_2 + 100$

$t_4 = 1300$

Results

The results of the insertion error investigation are summarized in Tables F-I and F-II for the 240° and 140° reference trajectories respectively. These tables present the pertinent data for the case in which the range-angle (ρ, θ_ρ) combination of radar parameters is used for "in-plane" observations. This combination in general gives the best results in terms of miss distance performance and the ability to correct for off-nominal conditions. Alternate combinations of radar parameters which give acceptable results are also indicated. In a few instances, the alternate combination actually performs like the range-angle combination. However, there are insertion errors so that no alternate is acceptable. In general the $\rho - \theta_\rho$

TABLE F-I

140° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	Alt.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
1	100	1 2	337 850	Unacceptable Unacceptable			ΔC_1 = 607,600
2	10	1 2	3.5 .9	1.83 2.4	63.6 76.8		ΔC_1 = 60,760
3	12.8	1 2	.88 .33	1.9 2.1	105.1 74.5	$\underline{\rho}$ $\underline{\dot{\rho}}$	ΔC_2 = 121,520
4	1.3	1 2	Corr. not applied 1st. Corr. not applied				ΔC_2 = 12,152
5	126	1 2	25 1.0	Unacceptable 2.1	 264.9		ΔC_3 = -60,760
6	11	1 2	1.1 0.19	1.9 2.8	35.6 50.3		ΔC_3 = -6,076
7	108	1 2	4.7 0.14	1.8 2.1	144 250.5	$\overline{\dot{\rho}}$ $\overline{\theta}$	ΔV_{c1} = 100
8	11	1 2	.24 1st Corr. not applied	1.9	34.2		ΔV_{c1} = 10
9	7.5	1 2	2.5 0.5	1.9 2.2	63 106.1	$\underline{\dot{\rho}}$ $\underline{\rho}$	ΔV_{c2} = -50

TABLE F-I (Continued)

140° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	Alt.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
10	.76	1 2	Corr. not applied 1st. Corr. not applied				$\Delta V_{c2} = -5$
11	64.7	1 2	26.4 0.34	Corr. not applied 2.1	 268		$\Delta V_{c3} = -100$
12	6.5	1 2	.68 1st. Corr. not applied	1.9	29.3		$\Delta V_{c3} = -10$
13	18.7	1 2	2.6 0.41	1.8 2.8	76.5 60.3	$\frac{\rho}{-}$ $\frac{\theta}{-}$	$\Delta C_1 = 60,760$ $\Delta V_{c1} = 10$ $\Delta C_2 = 12,152$ $\Delta V_{c2} = -5$ $\Delta C_3 = 6,076$ $\Delta V_{c3} = -10$
14	60	1 2	32 1.34	Unacceptable 2.1	 280.9		+3%, $\theta_P = f(t)$ $\Delta C_1 = +19,300$ $\Delta V_{c1} = -19$ $\Delta C_2 = 12,700$ $\Delta V_{c2} = 97.4$ $\Delta C_3 = -10,850$ $\Delta V_{c3} = -91.5$
15	16	1 2	2.0 0.14	1.9 2.4	55 80.3	$\frac{\dot{\rho}}{-}$ $\frac{\rho}{-}$	+3%, $\theta_P = f(\Delta V)$ $\Delta C_1 = 23,200$ $\Delta V_{c1} = -2.05$ $\Delta C_2 = -3,175$ $\Delta V_{c2} = -24.3$ $\Delta C_3 = -3,500$ $\Delta V_{c3} = -24.9$

TABLE F-I (Continued)

140° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	Alt.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
16	64	1 2	26 2.0	Unacceptable 2.1	 288.7		-3%, $\theta_P = f(t)$ $\Delta C_1 = -20,300$ $\Delta V_{c1} = 21$ $\Delta C_2 = 12,700$ $\Delta V_{c2} = 97.4$ $\Delta C_3 = 12,160$ $\Delta V_{c3} = 96.9$
17	22	1 2	3.0 0.43	1.9 2.3	160 118.3		-3%, $\theta_P = f(\Delta V)$ $\Delta C_1 = -24,600$ $\Delta V_{c1} = 2.7$ $\Delta C_2 = 6,350$ $\Delta V_{c2} = 48.7$ $\Delta C_3 = 4,900$ $\Delta V_{c3} = 29.1$

combination always gives acceptable performance for all of the insertion errors investigated, where the other would not. The one exception to this statement is an unreasonably large 100 n. mi. position error for which none of the radar combinations could make satisfactory correction. This results from the linearization process at extremely large error distances.

To determine the effect of different types of insertion error (position and velocity), each error has been varied independently (runs 1-12). Trajectory results have been obtained for what is considered large and small insertion errors. The effects of simultaneous errors have been investigated for the special cases where the errors are equivalent to a lunar launch with +3%, -3% variation in engine thrust and the pitch program is a function of time with a ΔV engine cutoff criteria. These are runs 14 and 16. Runs 15 and 17 are equivalent to insertion errors resulting from a +3% and -3% variation in engine thrust and a pitch program which is a function of ΔV. Runs 13 and 18 for the 240° trajectory represent small and large combinations of insertion errors. The data presented in Tables F-I and II represent the results of performing 180° trajectory runs on the computer when the three radar combinations and the four variations of the time of the first correction are included.

Tables F-I and II further present the data summarized in terms of miss distance at the nominal time of intercept and the total ΔV required for the midcourse correction, including the random radar effects. The "Bias" column shows the geometric effects of the insertion errors and the "Random" column the effects of radar measurements errors. The data presented for the double correction are based on the time of the first correction which gives the minimum miss distance. For some small insertion errors, the first correction for the double correction case has not been applied because the random radar errors exceed the necessary correction. These cases are not used for comparison purposes (single vs. double correction). When the miss distance exceeds 10 n. mi., the results are noted as "unacceptable." In the CSM centered coordinate system, the Insertion Errors can be interpreted as follows:

ΔC_1 = horizontal in-plane position error

ΔC_2 = horizontal out-of-plane position error

ΔC_3 = vertical position error

ΔV_{c1} = horizontal in-plane velocity error

ΔV_{c2} = horizontal out-of-plane velocity error

ΔV_{c3} = vertical velocity error

The single correction for the 140° reference trajectory is generally effective, except for large errors in vertical velocity and altitude which occur when the lunar launch pitch program is a function of time. The data indicate that a single correction is effective when the pitch program is a function of ΔV for a 3% engine thrust variation, but is not effective in reducing the insertion errors when the pitch program is a function of time. As previously noted, no attempt has been made to determine the best time for the single correction. There is a tradeoff between an early midcourse correction, when

TABLE F-II

240° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	* Radar Comb.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
1	100	1 2	Unacceptable Unacceptable				ΔC_1 = 607,600
2	10	1 2	1.7 1st. Corr. not applied	4.2	25	$\dot{\rho}$ θ — —	ΔC_1 = 60,760
3	10	1 2	7.5 1.24	4.2 5.1	59 157.5	$\dot{\rho}$ θ $\dot{\rho}$ θ	ΔC_2 = 121,520
4	1.0	1 2	Corr. not applied .51	6.2	53.4	$\overline{\dot{\rho}}$ $\overline{\theta}$	ΔC_2 = 12,152
5	309	1 2	Unacceptable 6.1	5.1	253.3	— — Non	ΔC_3 = 60,760
6	31	1 2	2.3 0.24	4.1 5.6	64.0 54.4	$\dot{\rho}$ θ $\dot{\rho}$ θ	ΔC_3 = 6,076
7	311	1 2	1.22 0.57	Unacceptable 5.1	189.7	$\overline{\dot{\rho}}$ $\overline{\theta}$	ΔV_{c1} = 100
8	30.5	1 2	2.4 1st. Corr. not applied	4.1	63.3	$\dot{\rho}$ θ — —	ΔV_{c1} = 10
9	8.3	1 2	0.9 .007	4.2 5.1	50.0 141.3	$\dot{\rho}$ θ $\dot{\rho}$ θ	ΔV_{c2} = -50
10	0.84	1 2	Corr. not applied 0.14	6.2	49.3	$\overline{\dot{\rho}}$ $\overline{\theta}$	ΔV_{c2} = -5

TABLE F-II (Continued)

240° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	* Radar Comb.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
11	59.5	1 2	104 0.99	Unacceptable 5.1	 183.7	$\dot\rho\ \rho$ $\dot\rho\ \theta$	$\Delta V_{c3} = -100$
12	6	1 2	1.3 1st. Corr. not applied	4.1	22	$\dot\rho\ \rho$	$\Delta V_{c3} = -10$
13	54	1 2	4.0 1.0	4.14 6.3	122 69.9	$\dot\rho\ \theta$ $\dot\rho\ \theta$	$\Delta C_1 = 60{,}760$ $\Delta V_{c1} = 10$ $\Delta C_2 = 12{,}152$ $\Delta V_{c2} = -5$ $\Delta C_3 = 6{,}076$ $\Delta V_{c3} = -10$
14	51	1 2	Unacceptable 1.35	 5.1	 262.8	$\dot\rho\ \theta$	+3%, $\theta_P = f(t)$ $\Delta C_1 = 19{,}300$ $\Delta V_{c1} = -19$ $\Delta C_2 = 12{,}700$ $\Delta V_{c2} = 97.4$ $\Delta C_3 = -10{,}850$ $\Delta V_{c3} = -91.5$
15	22	1 2	5.8 0.2	4.14 5.6	78 98.5	$\dot\rho\ \rho$ $\dot\rho\ \theta$	+3%, $\theta_P = f(\Delta V)$ $\Delta C_1 = 23{,}200$ $\Delta V_{c1} = -2$ $\Delta C_2 = 3{,}175$ $\Delta V_{c2} = 24.3$ $\Delta C_3 = -3500$ $\Delta V_{c3} = -24.9$

TABLE F-II (Continued)

240° Reference Trajectory ($\rho - \theta_P$)

Run No.	Miss Dist. No Corr. N.Mi.	No. of Corr.	Miss Dist. (corr.) Bias N.Mi.	Miss Dist. (corr.) Random N.Mi.	Total ΔV Reqd. for DMC ft/sec	* Radar Comb.	Insertion Error (ΔC in ft.) (ΔV_c in ft/sec)
16	52	1 2	Unacceptable 1.3	5.2	241.2	$\overline{\dot{\rho}}$ $\overline{\theta}$	-3%, $\theta_P = f(t)$ $\Delta C_1 = -20,300$ $\Delta V_{c1} = 21$ $\Delta C_2 = 12,700$ $\Delta V_{c2} = 97.4$ $\Delta C_3 = 12,160$ $\Delta V_{c3} = 97$
17	46.5	1 2	12.0 0.5	Unacceptable 5.1	152.6	$\overline{\dot{\rho}}$ $\overline{\theta}$	-3%, $\theta_P = f(\Delta V)$ $\Delta C_1 = -24,600$ $\Delta V_{c1} = 2.7$ $\Delta C_2 = 6,350$ $\Delta V_{c2} = 48.7$ $\Delta C_3 = 4,900$ $\Delta V_{c3} = 29.2$
18	266	1 2	170 2.63	Unacceptable 5.6	234.3	$\overline{\dot{\rho}}$ $\overline{\theta}$	$\Delta C_1 = 60,760$ $\Delta V_{c1} = 100$ $\Delta C_2 = 60,760$ $\Delta V_{c2} = -50$ $\Delta C_3 = 18,228$ $\Delta V_{c3} = -100$

* Other combinations of radar measurements might yield acceptable miss distances for the particular burn-out condition.

radar errors are large but linearization errors are small, and a late midcourse correction. It may be possible to correct for a timed pitch program by optimizing the time of the correction.

The double correction for the 140° reference trajectory is effective for all insertion errors considered (except when ΔC_1 is 100 n. mi.). A first correction occurring 550 seconds after burn-out generally gives reasonable performance in terms of accuracy and ΔV required.

The single correction for the 240° reference trajectory is not effective for large insertion errors. Since the 240° trajectory may be required for aborts, the insertion errors could be substantial. Therefore, it is preliminarily concluded that a double correction will be necessary for abort situations. In general, double correction will be effective for almost all of the insertion errors considered.

The 240° trajectory study shows an extreme sensitivity to out-of-plane conditions when the time of the first correction occurs at 1250 seconds which is the latest time for the first correction considered. Small out-of-plane conditions require velocity corrections on the order of thousands of feet per second. Further investigation has revealed that at this condition the remaining time of flight is equivalent to a 179° central angle. It can be shown that the 180 degree central angle trajectory becomes a singular point when correcting for "out-of-plane" conditions. Similar to the Hohmann transfer trajectory, there is no velocity correction which will correct for out-of-plane position and maintain the nominal 180 degree time of flight. Since the differential correction technique is based on maintaining a constant time of flight, the correction coefficients become indeterminate in this situation. Therefore, trajectories approaching this singular condition become sensitive to out-of-plane conditions. For trajectories with intercept angles greater than 180 degrees it appears to be more efficient in terms of ΔV to make early midcourse correction for "in-plane" errors only, and make "out-of-plane" corrections at the time of the second midcourse correction. This is evident from the fact that the ΔV required for single and double DMC is 50 ft/sec and 141 ft/sec respectively for 50 ft/sec ΔV_{c_2} insertion errors. Another benefit would occur in reducing the ΔV due to not using the early "out-of-plane" radar measurements, since random errors in these measurements would cause unwarranted ΔV corrections.

At the time of this preliminary report, the computer program has not been modified to investigate the trajectory characteristics when the "out-of-plane" corrections are not applied with the first correction. It is therefore not possible to estimate the best time to make an early DMC. However, based on the data presently available, the first correction occurring approximately 700 seconds after burn-out gives reasonable results. Continuing work should serve to define this time more accurately.

The effect of bias and drift errors in the angular measurements are shown in Table F-III for the 140° and 240° reference trajectories. The data presented show the miss distance and the ΔV applied for the single DMC due to each error.

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EFFECT OF BIAS AND DRIFT ERRORS

TABLE F-III

Angular Error	140° Reference Trajectory		240° Reference Trajectory	
	Miss Dist. N. M.	ΔV Applied ft/sec	Miss Dist. N. M.	ΔV Applied ft/sec
0.1°/hr drift	0.3	0.5	1.0	1.8
1.0°/hr drift	1.3	5.0	8.7	18.0
10°/hr drift	11.4	50.7	86.5	180.0
0.1 deg. bias	0.7	1.8	1.5	2.8
0.3 deg. bias	1.6	5.5	3.5	8.5
1.0 deg. bias	5.3	21.4	11.8	---

The 240° trajectory is more sensitive to bias and drift errors, probably due to the longer time of flight. These data indicate that the back-up attitude reference alignment and drift performance should be on the order of 0.3 degrees and 1.0 degrees/hour, respectively.

Conclusions

Preliminary conclusions based on the presently available data are:

1. The differential midcourse correction can compensate for a large range of ascent burn-out errors.

2. Two midcourse corrections provide compensation for a larger range of burn-out errors than a single correction.

3. The range-angle radar measurements combination gives the best results for large insertion errors.

4. Two midcourse corrections are necessary for abort situations if insertion errors on the order of 100 ft/sec velocity and 10 n.mi. position are assumed.

5. "Out-of-plane" corrections should not be made when the remaining time of flight corresponds to approximately 180 degrees central angle.

6. Better results in terms of ΔV requirements will probably be obtained when only "in-plane" corrections are made early and both "in-plane" and "out-of-plane" corrections are made with the final correction.

7. The attitude reference alignment requirements are on the order of 0.3 degrees and the drift rate specification should be 1.0 deg/hr or better.

b. Back-up Explicit Guidance Analysis

A back-up explicit guidance midcourse correction technique incorporating linearized equations has been investigated for use during the coasting ascent phase of the LEM mission. Linearization of the equations would permit significant simplification of the guidance system.

The results of this analysis indicate that the simplified equations cause large ΔV expenditures for midcourse corrections and rendezvous. In addition, the midcourse corrections do not significantly improve off-nominal trajectories.

Analysis of a Simplified Back-up Guidance System Using an Explicit Guidance Technique

This guidance scheme has been developed using linearized equations of motion which describe the motion of the LEM with respect to a CSM centered Cartesian coordiante system. The analysis has been performed with the aid of

an analog computer and an existing three degree-of-translational freedom digital program. The digital program has generated nominal and off-nominal reference trajectories. Range, LOS angle, range rate and LOS rate taken from several points along these trajectories provides inputs to the analog navigation computer which determines the LEM position and velocity relative to the CSM, and supplies this information to the analog guidance computer which in turn calculates the distance by which the LEM would miss the CSM. These miss distances determine the velocity increments required to give an intercept trajectory. The equations programmed in the navigation and guidance computer are presented in Table F-IV.

Adding these velocity increments to the existing velocity, a rendezvous ΔV and trajectory analysis has been performed digitally for these new trajectories. The rendezvous guidance law is described in LEM Memorandum IMO-500-22, 1 April 1963.

Analysis - Phase I

One nominal and two off-nominal reference trajectories listed below have been used for this analysis.

1. Nominal Hohmann transfer to the CSM in an 80 n. mi. circular orbit.

2. In-plane transfer resulting from a +2% thrust variation during powered ascent. (The powered ascent profile calls for a 17.4 second vertical rise, 10 deg/sec pitch rate for 6.6 seconds, 0.12391 deg/sec pitch rate until a ΔV cutoff of 6026.1 fps is reached.)

3. 2° out-of-plane transfer with nominal Hohmann insertion velocity at pericynthion.

The nominal Hohmann transfer trajectory gives an indication of the validity and accuracy of the explicit guidance method mechanized with the linearized equations. Velocity corrections computed at the start of the nominal coasting ascent should have been zero. However, the corrections calculated at these points by the guidance system are prohibitively large (i.e. 120 fps at a point near the start of the coasting ascent), and so are the velocity corrections for the off-nominal trajectories.

In an attempt to improve the calculated velocity corrections, the correction term $\left(\frac{h}{R} V_{cl}\right)$ has been added to the navigation equations for V_{cl} in order to be consistent in the linearization of the guidance equations. Although the addition of the correction term improves the quality of the midcourse corrections they were still excessive in magnitude in their effect on coasting ascent trajectories. The rendezvous analysis of the corrected trajectories reveals that when the rendezvous maneuver can be completed, the ΔV expenditure is larger than the ΔV budget allotment.

TABLE F-IV

EQUATIONS FOR THE N & G COMPUTER

Navigation Equations

$$C_1 = \rho \cos\Psi \cos(\sigma_A - \phi)$$

$$C_2 = \rho \sin\Psi$$

$$C_3 = -\rho \cos\Psi \sin(\sigma_A - \phi)$$

$$Vc_1 = \dot{\rho} \sin\Psi \cos(\sigma_A - \phi) - \rho\dot{\Psi} \sin\Psi \cos(\sigma_A - \phi) - \rho(\dot{\sigma}_A - \dot{\phi}) \cos\Psi \sin(\sigma_A - \phi)$$

$$Vc_2 = \dot{\rho} \sin + \rho\dot{\Psi} \cos\Psi$$

$$Vc_3 = -\dot{\rho} \cos\Psi \sin(\sigma_A - \phi) + \rho\dot{\Psi} \sin\Psi (\sigma_A - \phi) - \rho(\dot{\sigma}_A - \dot{\phi}) \cos\Psi \cos(\sigma_A - \phi)$$

Guidance Equations

$$0 = C_1 + 6(\dot{\sigma}_A t - \sin\dot{\sigma}_A t) C_3 + (-3t + \frac{4}{\dot{\sigma}_A} \sin\dot{\sigma}_A t)(Vc_1 + \Delta Vc_1) + \frac{2}{\dot{\sigma}_A}(1 - \cos\sigma_A t)(Vc_3 + \Delta Vc_3)$$

$$0 = (4 - 3\cos\dot{\sigma}_A t) C_3 - \frac{2}{\dot{\sigma}_A}(1 - \cos\dot{\sigma}_A t)(Vc_1 + \Delta Vc_1) + (\frac{1}{\dot{\delta}_A} \sin\dot{\sigma}_A t)(Vc_3 + \Delta Vc_3)$$

$$0 = (\cos\dot{\sigma}_A t) C_2 + (\frac{1}{\dot{\sigma}_A} \sin\sigma_A t)(Vc_2 + \Delta Vc_2)$$

Symbols

$C_{1,2,3}$ = LEM position in CSM centered, CSM local vertical coordinate system as defined in L500-M03-9

$Vc_{1,2,3}$ = LEM velocity in CSM coordinate system defined above

ρ = Range to CSM

Ψ = Angle between the LOS and its projection in the CSM orbital plane

$\sigma_A - \phi$ = Angle between the LOS projection in the CSM orbital plane and the CSM local horizontal

$\dot{\phi}$ = Component of the inertial LOS rate in the CSM orbital plane

$\dot{\sigma}_A$ = CSM orbital rate

t = Time of flight remaining

$\dot{\rho}$ = Range rate

The midcourse corrections with and without correction terms are presented in Table F-V

TABLE F-V

MIDCOURSE CORRECTION ΔV (fps)

TRAJECTORY		Time From Insertion (seconds)				
		0	500	1000	1500	2000
NOMINAL HOHMANN	NO CORRECTION TERM	120	58.7	23.3	9.35	1.71
	WITH CORRECTION TERM	56.6	36.3	21.8	7.27	4.4
IN-PLANE +2% THRUST VARIATION	NO CORRECTION TERM	168	129	117	128	151
	WITH CORRECTION TERM	132	110	108	123	155
2° OUT-OF-PLANE, HOHMANN INSERTION VEL.	NO CORRECTION TERM	228	85.3	37	20.4	21.8
	WITH CORRECTION TERM	207	66.4	37.6	22.1	22.2

Table F-V shown that the computed midcourse ΔV decreases with time for the nominal Hohmann because the equations describe the situation more accurate as the range decreases, and the vehicle is already in an intercept trajectory. For the off-nominal trajectories, the midcourse correction ΔV becomes minimum when the accuracy of the situation described by the equations is improved while the ΔV required to correct the trajectory is still relatively small. Later the accuracy improves still further but the LEM is so far off the nominal that a large midcourse correction ΔV is required.

Analysis - Phase II

The analysis performed during Phase I indicates that the explicit guidance technique as originally envisioned is unacceptable for back-up guidance purposes. The navigation and guidance equations have been modified to describe the motion of the LEM in a more accurate but complex CSM centered curvilinear coordiante system (see "A Study of Certain Aspects of the Ascent of a Lunar Excursion Module from the Moon's Surface Toward Rendezvous with a Command Module in Lunar Orbit" by H. U. Burri).

The procedure used in the Phase I analysis has been repeated for Phase II. The velocity corrections calculated with the new equations are improved but still require large midcourse ΔV expenditures (an average of about 70 fps). The rendezvous ΔV analysis shows that some velocity corrections to an off-nominal trajectory force the LEM further away from the nominal and cause large miss distances (greater than 30 n. mi. for some corrections).

The original concept of an explicit guidance system includes the possibility of two midcourse corrections. The first would be made immediately following powered ascent, and could thus correct a severely off-nominal trajectory. A second, vernier type, midcourse correction would be applied later to yield an accurate intercept trajectory. However, the analysis shows that early ΔV corrections computed with the simplified explicit guidance system are excessive, and, in many cases, the resulting trajectories are further from the nominal. It is, therefore, not advantageous to consider a two-midcourse correction technique with the simplified explicit guidance system.

Previous analyses have shown that coasting ascent trajectories are extremely sensitive to velocity errors at or close to insertion (see LEM Memorandum LMO-500-54, 10 June 1963). Therefore, mission safety and midcourse ΔV considerations require off-nominal trajectories to be detected and corrected early during the coasting ascent.

The results of the analysis show that the simplified explicit guidance system computes grossly exaggerated midcourse corrections at distances far from the CSM. Therefore, the necessity of making a midcourse correction early, and the inability of the simplified guidance system to compute satisfactory midcourse corrections at these times, prohibits the use of a simplified explicit guidance system for midcourse corrections.

If an explicit guidance system, with its previously stated advantages, is to become a reality, the concept of a simple guidance system must be discarded. It can be shown that the next step above the system concept used for this analysis is an exact explicit guidance law with all terms included, and with a corresponding increase in system complexity.

c. Analysis of "t_1 Compensation" Back-up Guidance Law for Powered Ascent

The "t_1 Compensation" guidance law as applied to the phase of powered ascent from the lunar surface is effective in reducing the miss distance at rendezvous with the CSM.

Procedures and Assumptions

It has been assumed that the following ascent profile is being followed:

1. Vertical ascent at an inertial pitch angle of 90° until a nominal ΔV is reached.

2. Pitch-over at a rate of 10°/sec for 6.6 seconds.

3. Pitch-over at a rate of 0.12°/sec until a nominal ΔV is reached and thrust is cut off.

4. Coast to intercept with the CSM (which is in an 80 n. mi. circular orbit around the moon).

Using this profile, the nominal thrust termination conditions required for a Hohmann transfer to intercept with the CSM were obtained from an IBM 7094 computer run. They are: altitude = 49,974 ft., velocity = 5,580 ft/sec and flight path angle = -0.013°.

The "t_1 Compensation" guidance law limits the powered ascent profile so that near-nominal cutoff conditions are attained despite off-nominal engine conditions. The method of constraint is based on determining the acceleration and acceleration rate errors and then correcting for the predicted horizontal and vertical components of the resultant velocity error. The correction takes the form of a "kick angle (θ_k)" that adjusts the pitch angle to correct for vertical velocity error, and an adjustment of the ΔV required for thrust cutoff to correct for horizontal velocity error. The combination of corrected horizontal and vertical velocities constrains both flight path angle and resultant velocity at thrust cutoff to near-nominal conditions. As previously established, velocity errors are more significant than position errors, and no attempt has been made to correct altitude errors at thrust cutoff.

In addition to these velocity corrections, a further set of corrections is made to compensate for initial thrust vector misalignment by adjusting the pitch angle command as a function of body fixed accelerometer readings. This is discussed fully in LMO-540-49 which also contains a general description of the guidance law.

The corrections for horizontal and vertical velocities are obtained by solving the following equations:

$$\theta_k = A_0 + A_1 t_1 + A_2 t_{13} + A_3 \Delta V_2 \qquad 1$$

$$\Delta V_5 = B_0 + B_1 t_{13} + B_2 \Delta V_2 \qquad 2$$

Where θ_k = the "kick angle" correction.

ΔV_5 = the value of integrated specific force at which the thrust is cut off.

t_1 = the time at which the fast pitch-over rate is started.

t_{13} = the time between the start of the fast pitch rate and the time at which θ_k and ΔV_5 are computed.

ΔV_2 = the ΔV at the end of the fast pitch rate.

A_0, A_1, A_2, A_3, B_0, B_1, B_2 = empirically determined constants.

The original analytical derivation of "t_1 Compensation" (Reference LEM Memorandum LMO-540-49) was performed with a powered ascent profile different from the one described above. To expedite the analysis of "t_1 Compensation" for the ascent profile presently under consideration, it was decided that an empirical determination of the constants (A_0 - A_3, B_0 - B_2) be made in lieu of rederiving the analytic expression.

The analysis was performed on an IBM 7094 computer using a generalized three degree of translational freedom program (Reference LEM Memorandum LMO-500-44). Results were obtained by simulating thrust magnitude errors, Isp errors, initial mass errors, initial thrust misalignment errors, gyro drift errors, and pitch rate uncertainties, using both the "t_1 Compensation" guidance law and no compensation. These results were then compared with the nominal trajectory parameters and presented with the conclusions, as well as the range of parameters considered, in the following sections.

Range of Parameters

The nominal trajectory considered used an engine with an initial thrust of 3500 lbs, an initial Isp of 310 seconds, and a thrust-to-weight ratio of 0.474. The LEM considered had an initial mass of 229.3 slugs. In addition the CSM was assumed to be in a circular lunar orbit at an altitude of 80 n. mi. The effects of the following off-nominal conditions were then studied considering each error separately:

1. Thrust magnitude errors: $\pm$1.5%; $\pm$3%; $\pm$6%.
2. Isp errors: $\pm$3 seconds; $\pm$5 seconds; $\pm$10 seconds.
3. Initial mass errors: $\pm$0.75%; $\pm$1.5%; $\pm$3%.
4. Pitch axis thrust misalignment: $\pm$0.3 degrees.
5. Gyro drift: $\pm$1 degree/hr = 0.00028 degrees/sec.
6. Pitch rate uncertainty: $\pm$0.001 degrees/sec.

NOTE: The pitch rate uncertainty parameter determined the effect of a bias error in the commanded pitch rate, and acted only during the pitch maneuver. The gyro drift parameter simulated drift of the attitude reference which acts throughout the entire flight.

Results

The results are presented in the form of tables and graphs as follows:

Table F-VI is a comparison of the minimum range to the CSM, flight path angle error at cutoff, velocity error at cutoff, and altitude error at cutoff both with and without compensation for pitch axis thrust misalignment of $\pm$0.3 degrees. The graphs are grouped in sets of 5 according to their abscissas.

Figures F-2-5 are plotted as a function of thrust magnitude error, Figures F-6-10 are functions of specific impulse error and Figures F-11-15 are functions of initial mass errors. Within each group the graphs follow the following order:

Figures F-2, 6 and 11 are the minimum range between the LEM and the CSM near intercept, both with and without "t_1 Compensation". Figures F-2, 7 and 12 depict the flight path angle error at thrust cutoff, both with and without the use of compensation. Figures F-3, 8 and 13 are the velocity errors at cut-off both with and without compensation. Figures F-4, 9, and 14 are plots of altitude errors at cut-off, both with and without "t_1 compensation. Figures 5, 10 and 15 depict the change in cut-off time between the compensated and uncompensated.

Conclusions

Figures F-2, 5 and 10 clearly indicate that the "t_1 Compensation" method is very effective for limiting the intercept range when off-nominal accelerations are encountered due to thrust magnitude errors, Isp errors and initial mass errors. In addition, for the same off-nominal conditions, the plots of flight path angle, velocity and altitude errors at thrust cut-off (Figures F-2-4-6-8 and 12-14) show that the compensation is very effective in constraining the ascent profile to follow the nominal.

Figures F-6-10, when compared with Figures F-2-5 and 11-15, indicate that "t_1 Compensation" corrects for thrust magnitude and initial mass errors more effectively than it corrects for Isp errors. The reason for the difference in effectiveness is that the constants (A_0 - A_3, B_0 - B_2) have been selected to compensate for off-nominal thrust magnitude only. Initial mass errors have exactly the same effect as thrust magnitude errors, since thrust-to-weight ratio is the significant parameter, and so mass error effects are also eliminated. The effect of Isp errors, however, is not equivalent to either the effect of thrust magnitude or that of initial mass errors. An improved method of empirically determining the constants is presently under investigation in an attempt to minimize the effect of Isp errors as well.

Table F-I indicates that the "t_1 Compensation" method corrects satisfactorily for ascent engine misalignments. This correction, made during the vertical rise, assumes that the ascent misalignment remains constant, and that therefore only one pitch command correction is required.

Altitude at thrust cut-off is not explicity constrained by "t_1 Compensation", but an examination of Figures F-4, 9 and 14 reveals that this quantity has been fully controlled by explicitly constraining both the flight path angle and velocity at thrust cut-off. Since the coasting trajectory is relatively insensitive to initial altitude errors, this secondary compensation is sufficient.

According to LEM Memorandum LMO-500-15, a 1 ft./sec. error in cut-off velocity will cause about a 0.8 nautical mile miss at the CSM. Immediately prior to cut-off, the LEM acceleration is 28 ft./sec.^{2}. It can therefore be concluded that when instrumenting the "t_1 Compensation" method, cut-off

FIGURE F-2

MINIMUM RANGE TO C.S.M. VERSUS THRUST MAGNITUDE ERROR FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENS.

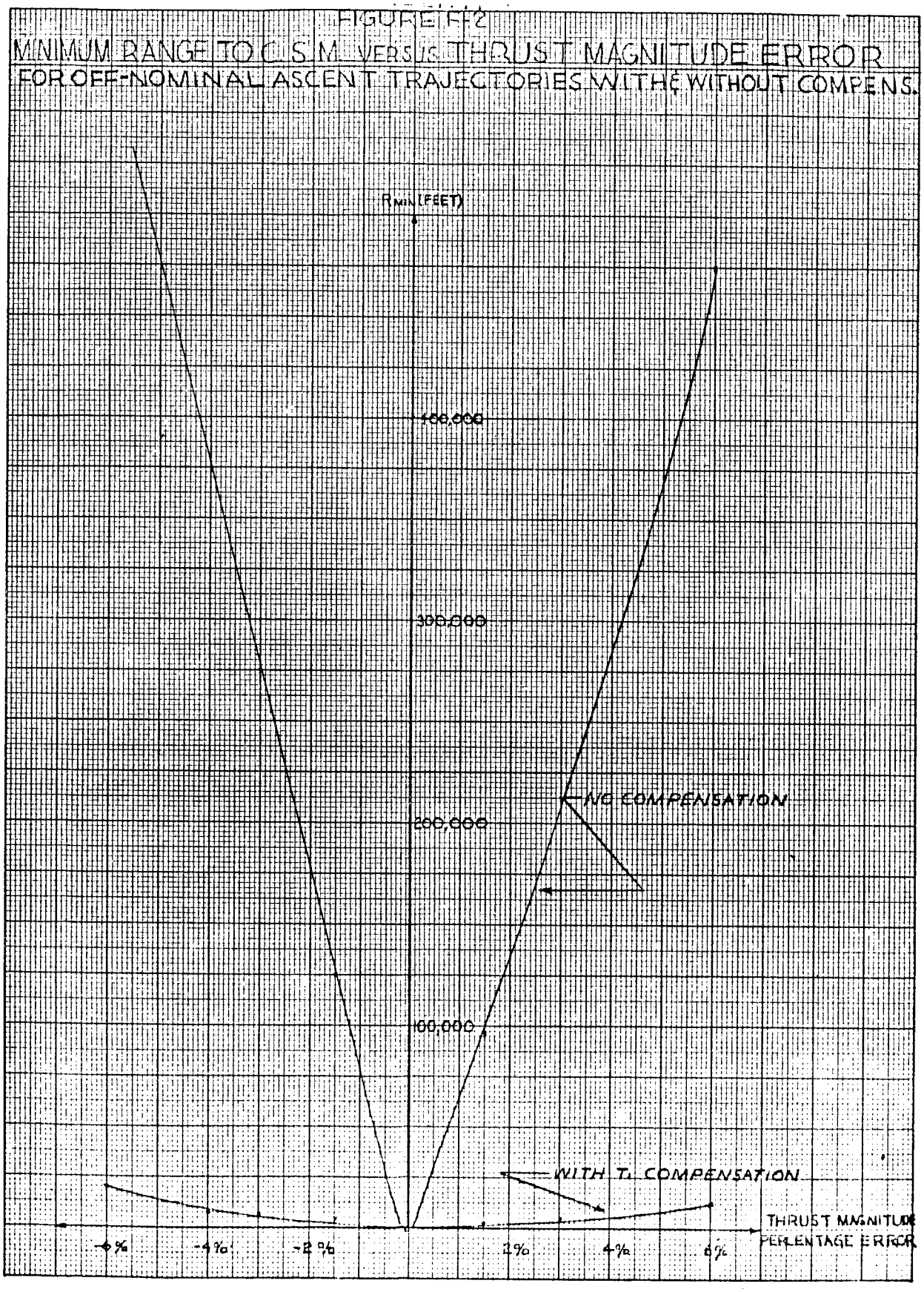

FIGURE F-3

FLIGHT PATH ANGLE ERROR vs. THRUST MAGNITUDE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENS

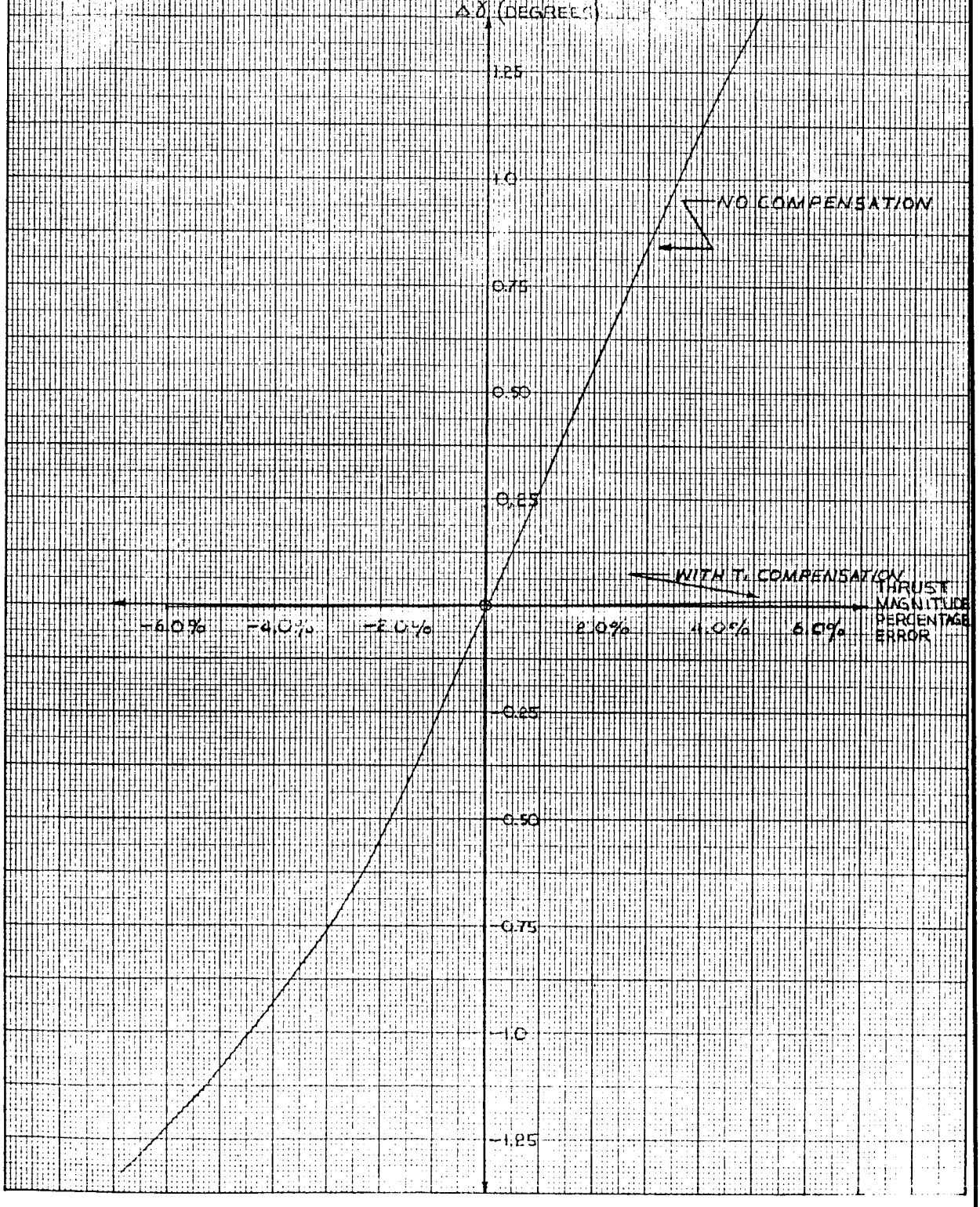

FIGURE F-4

VELOCITY ERROR AT CUT-OFF vs. THRUST MAGNITUDE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

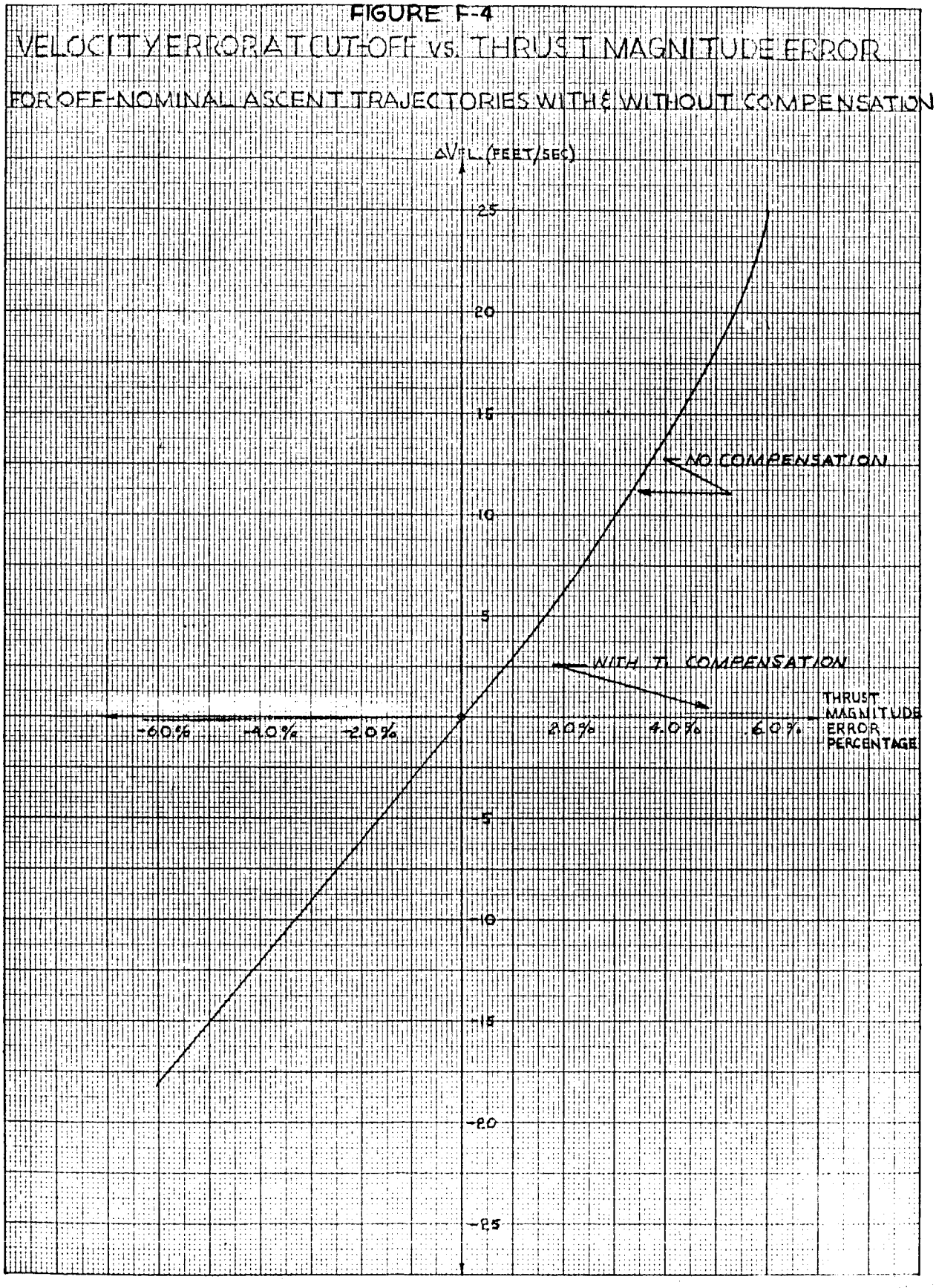

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FIGURE F-5

ALTITUDE ERROR vs. THRUST MAGNITUDE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

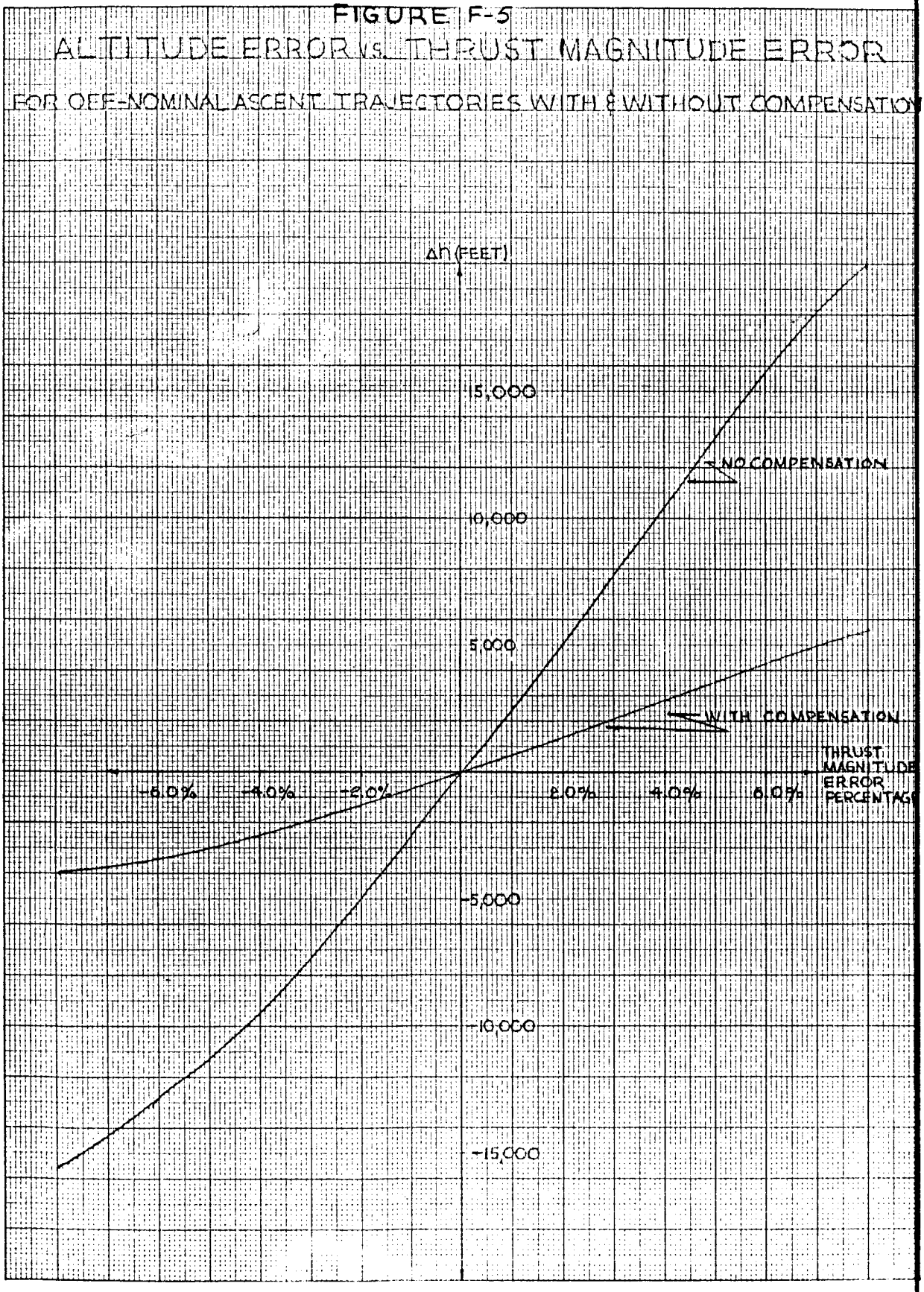

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FIGURE F-6

CHANGE IN CUT-OFF TIMES OF COMP. MINUS UNCOMP. SYSTEMS VS. THRUST MAGNITUDE ERROR FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

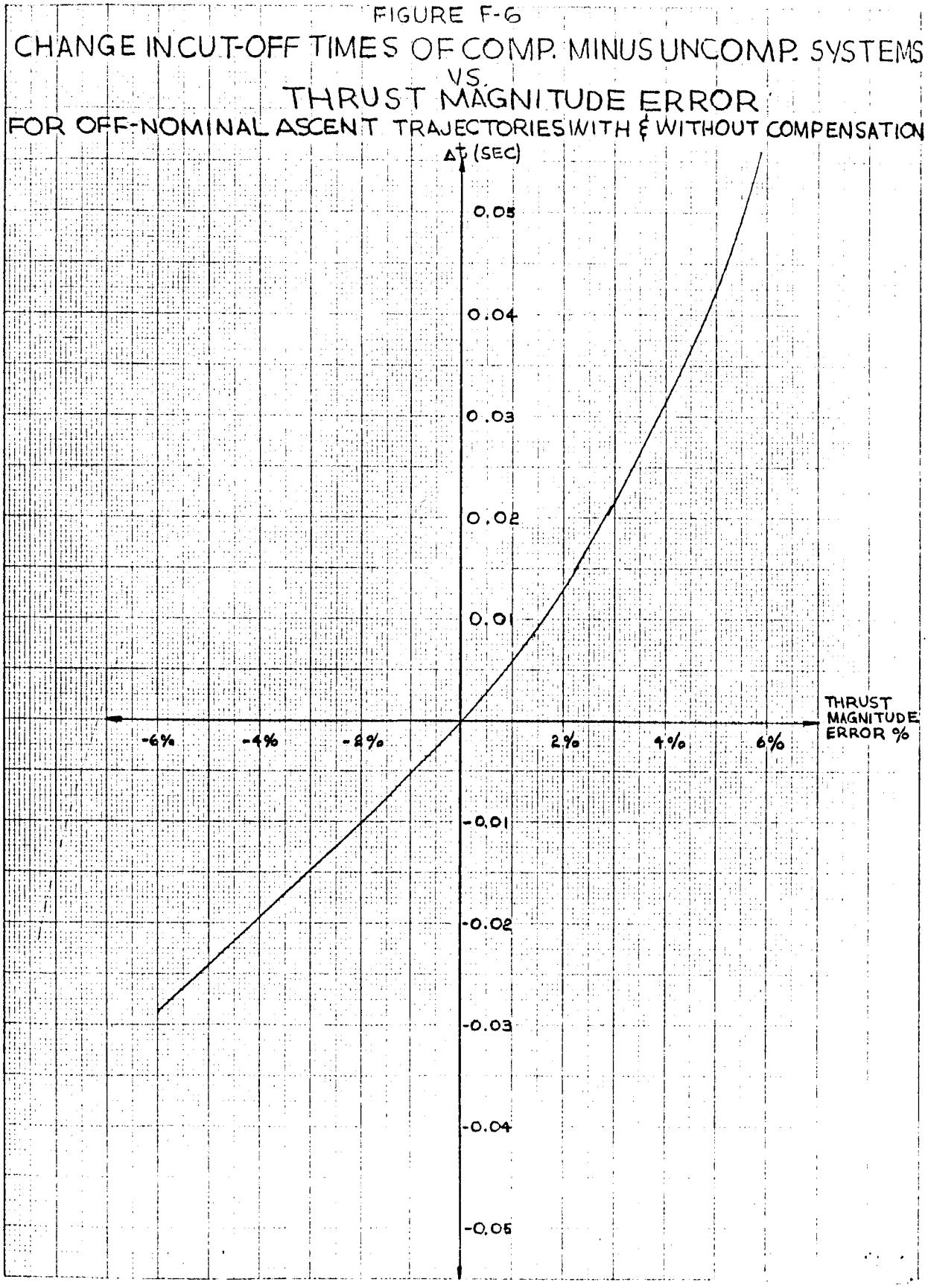

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FIGURE F-7

MINIMUM RANGE TO C.S.M. vs. SPECIFIC IMPULSE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

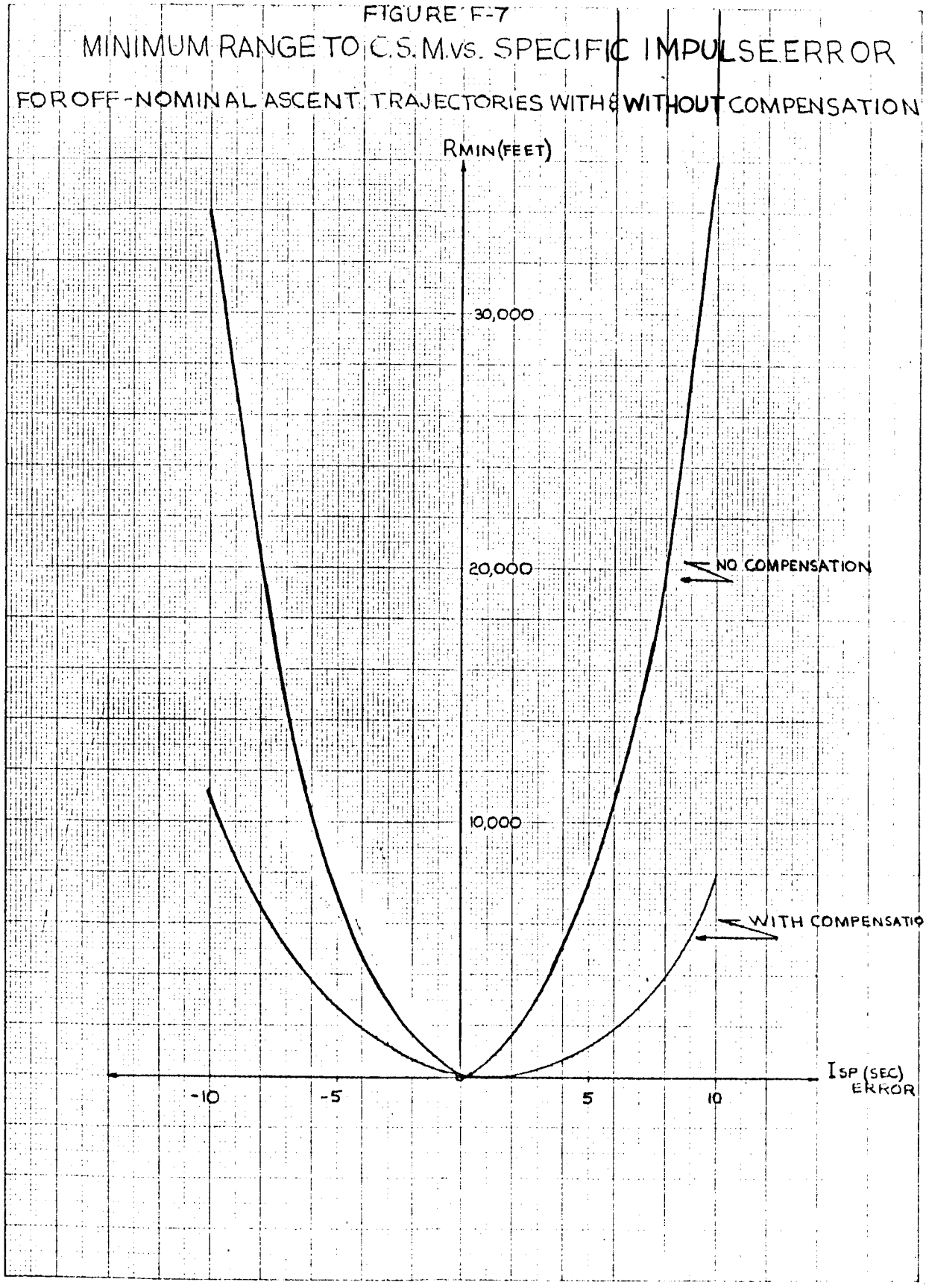

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FIGURE F-8

FLIGHT PATH ANGLE ERROR vs. SPECIFIC IMPULSE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

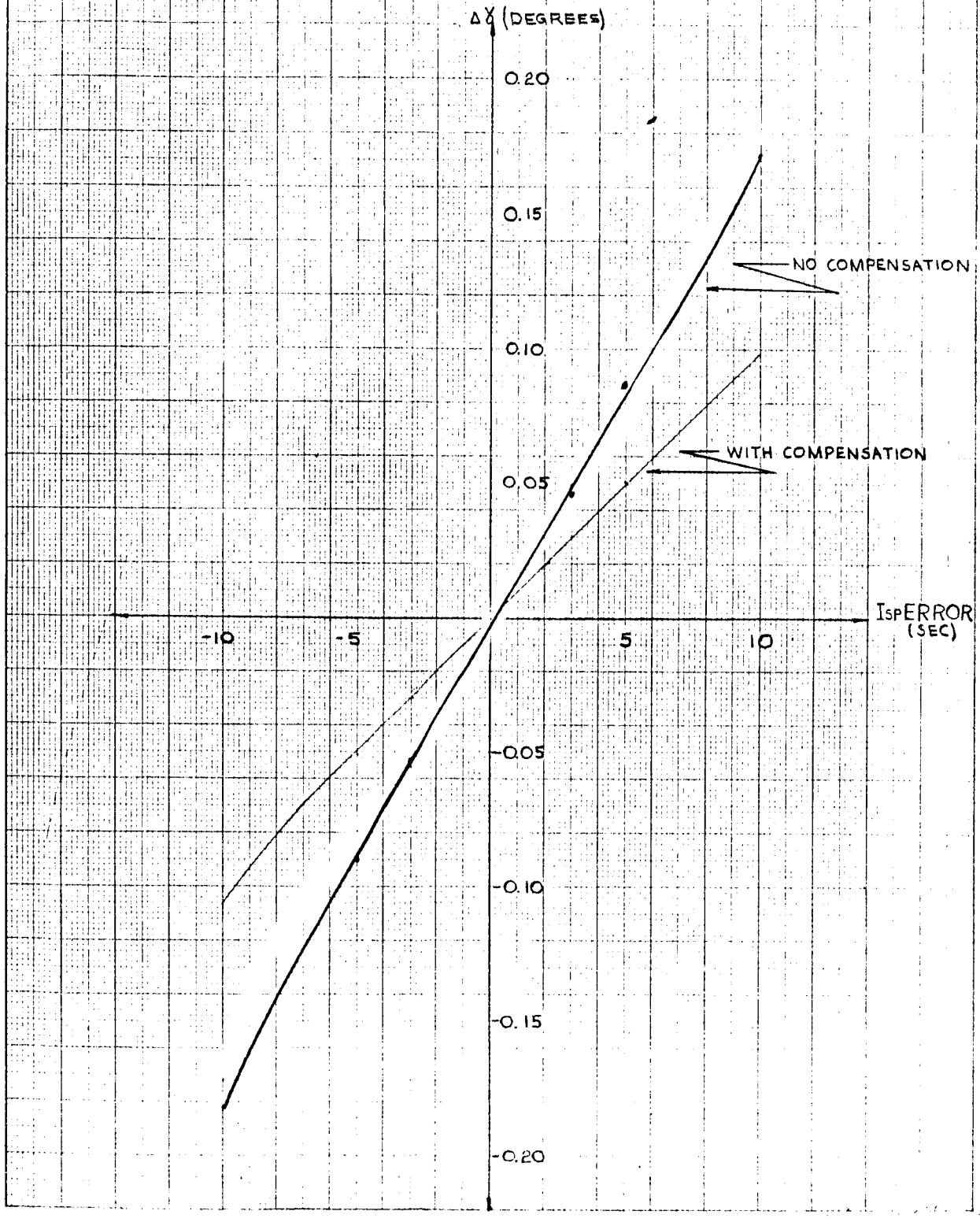


FIGURE F-9

VELOCITY ERROR AT CUT-OFF vs. SPECIFIC IMPULSE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENS

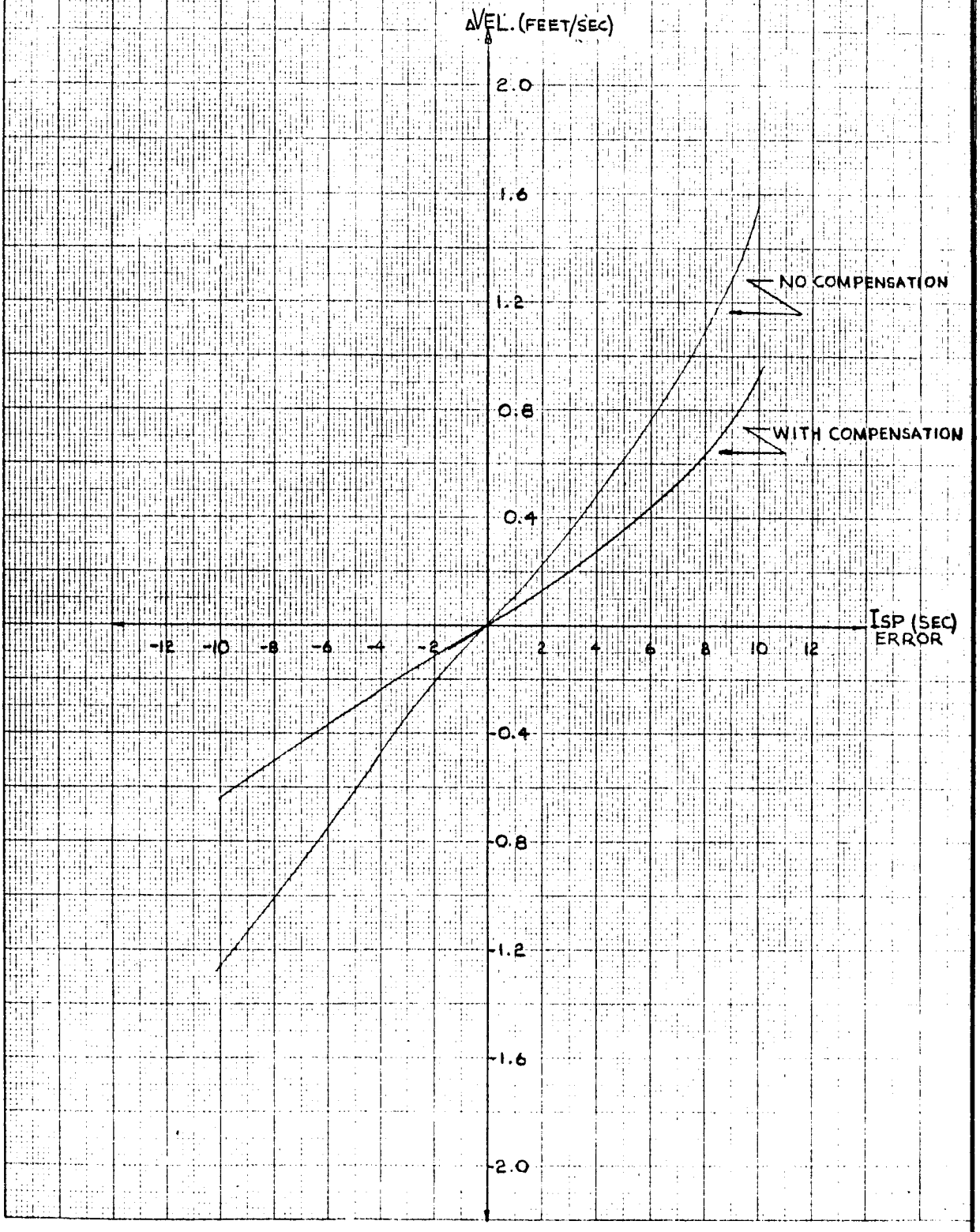


FIGURE F-10

ALTITUDE ERROR vs. SPECIFIC IMPULSE ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENS.

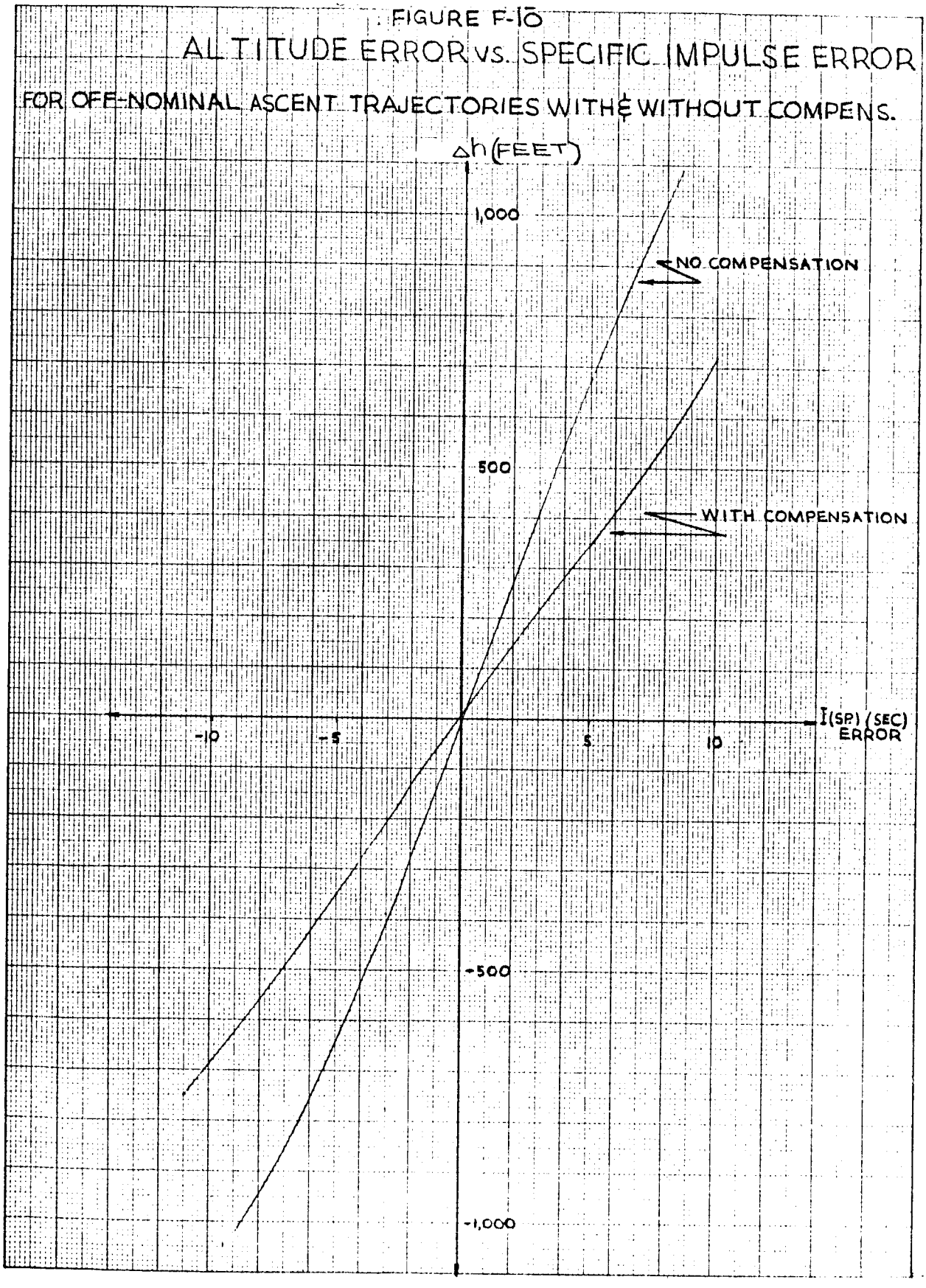


FIGURE F-11

CHANGE IN CUT-OFF TIMES OF COMP. MINUS UNCOMP. SYSTEMS
VERSUS
SPECIFIC IMPULSE ERROR
FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENS.

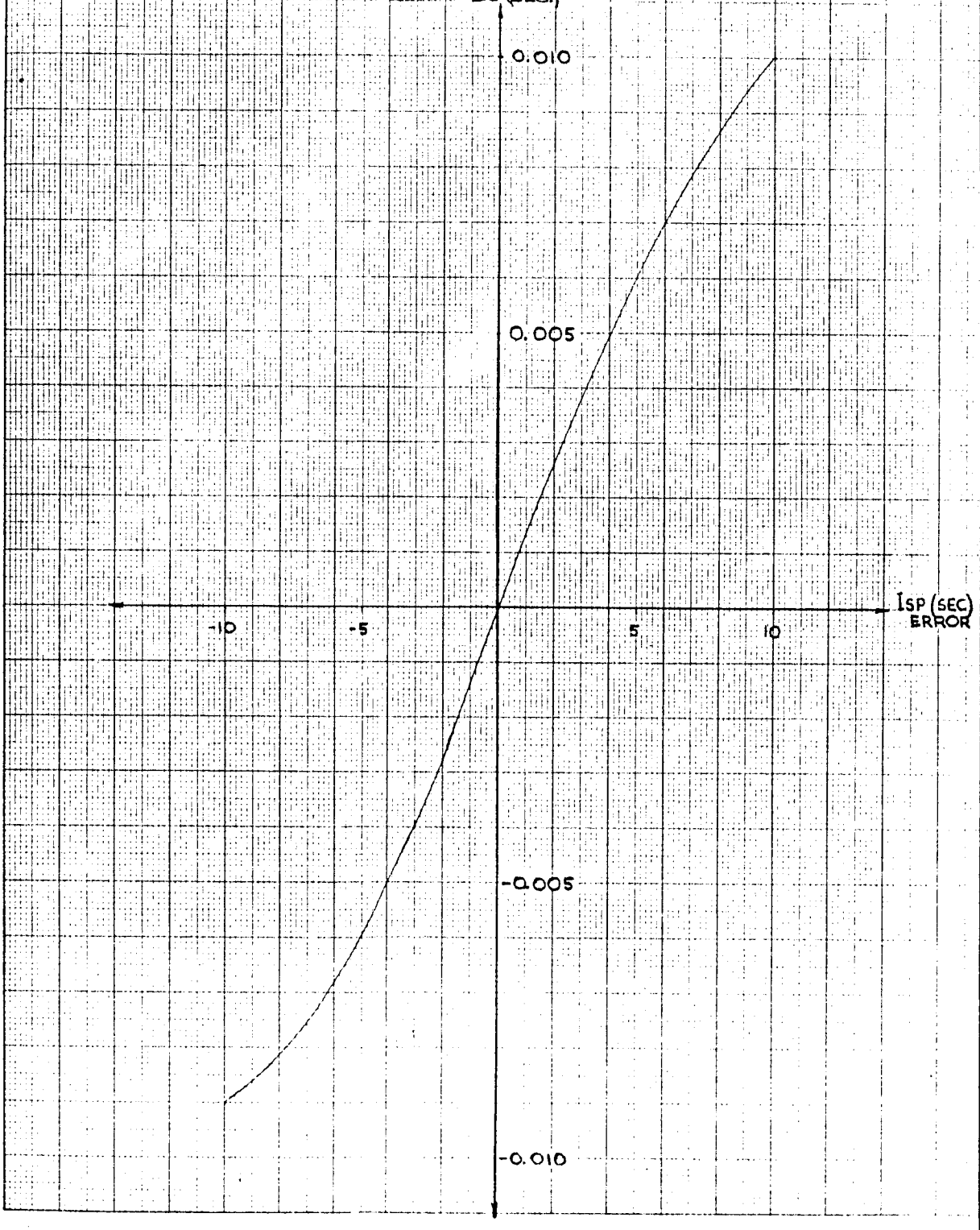


FIGURE F-12

MINIMUM RANGE TO C.S.M. vs. INITIAL MASS ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

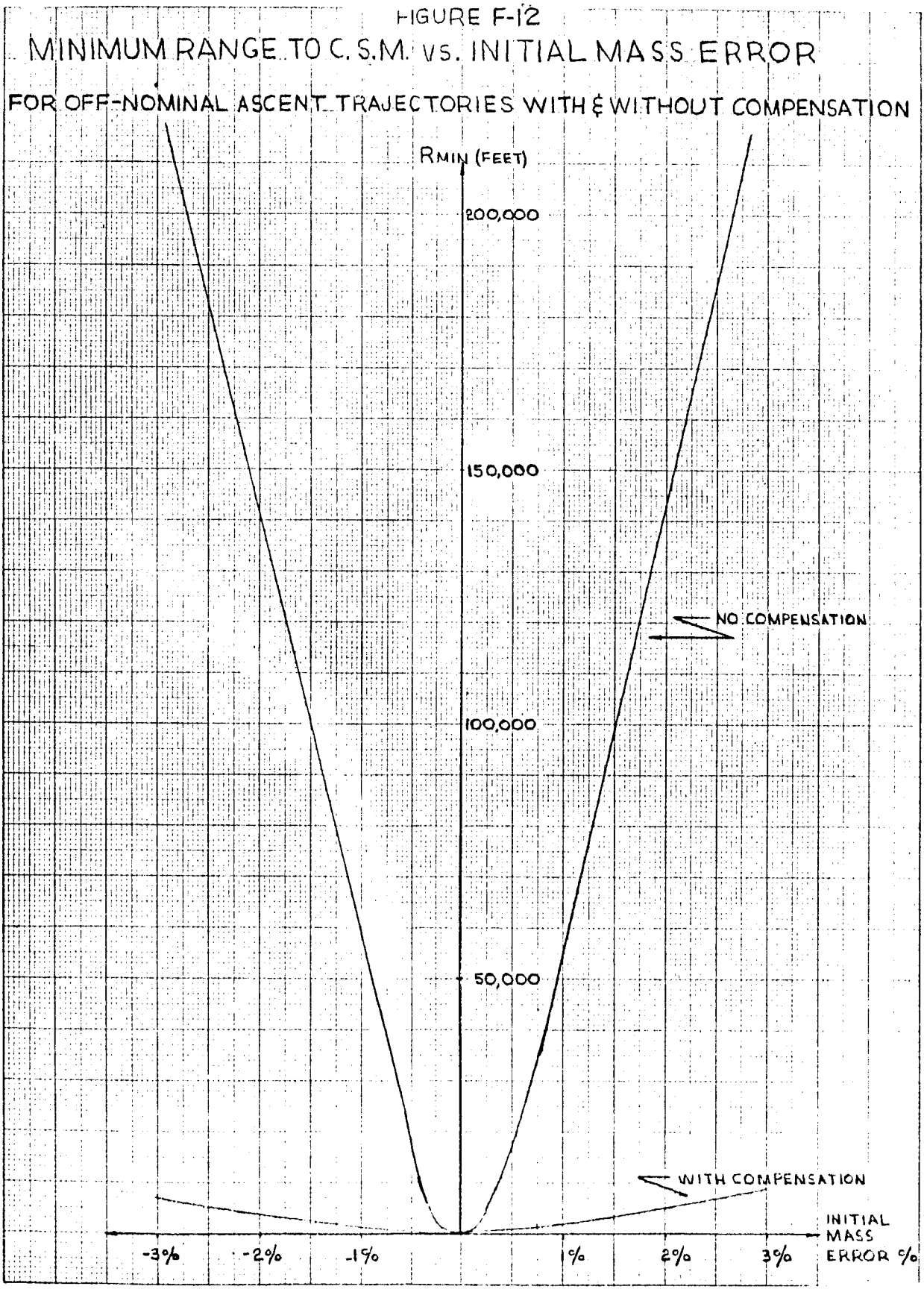


FIGURE F-13

FLIGHT PATH ANGLE ERROR vs. INITIAL MASS ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

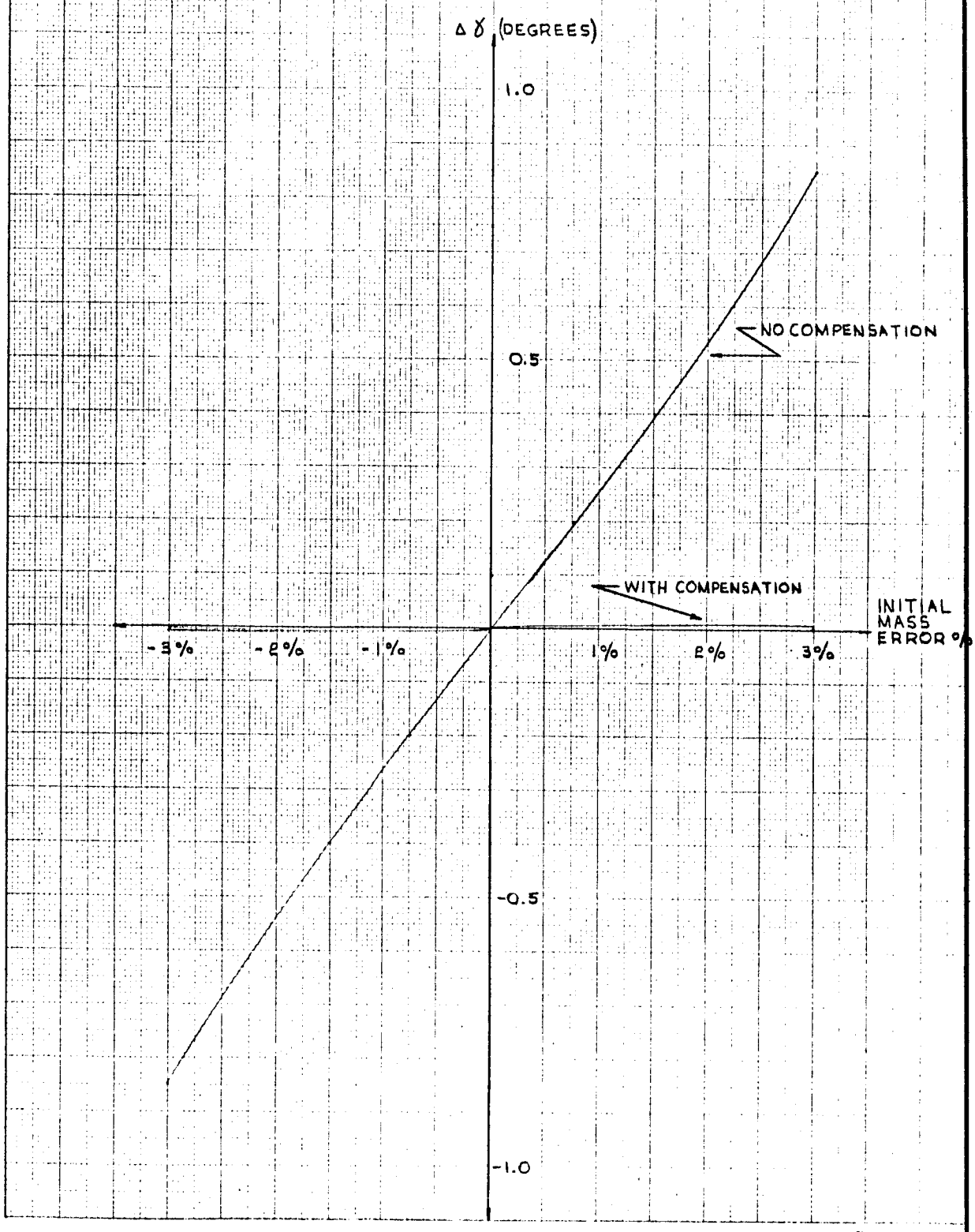
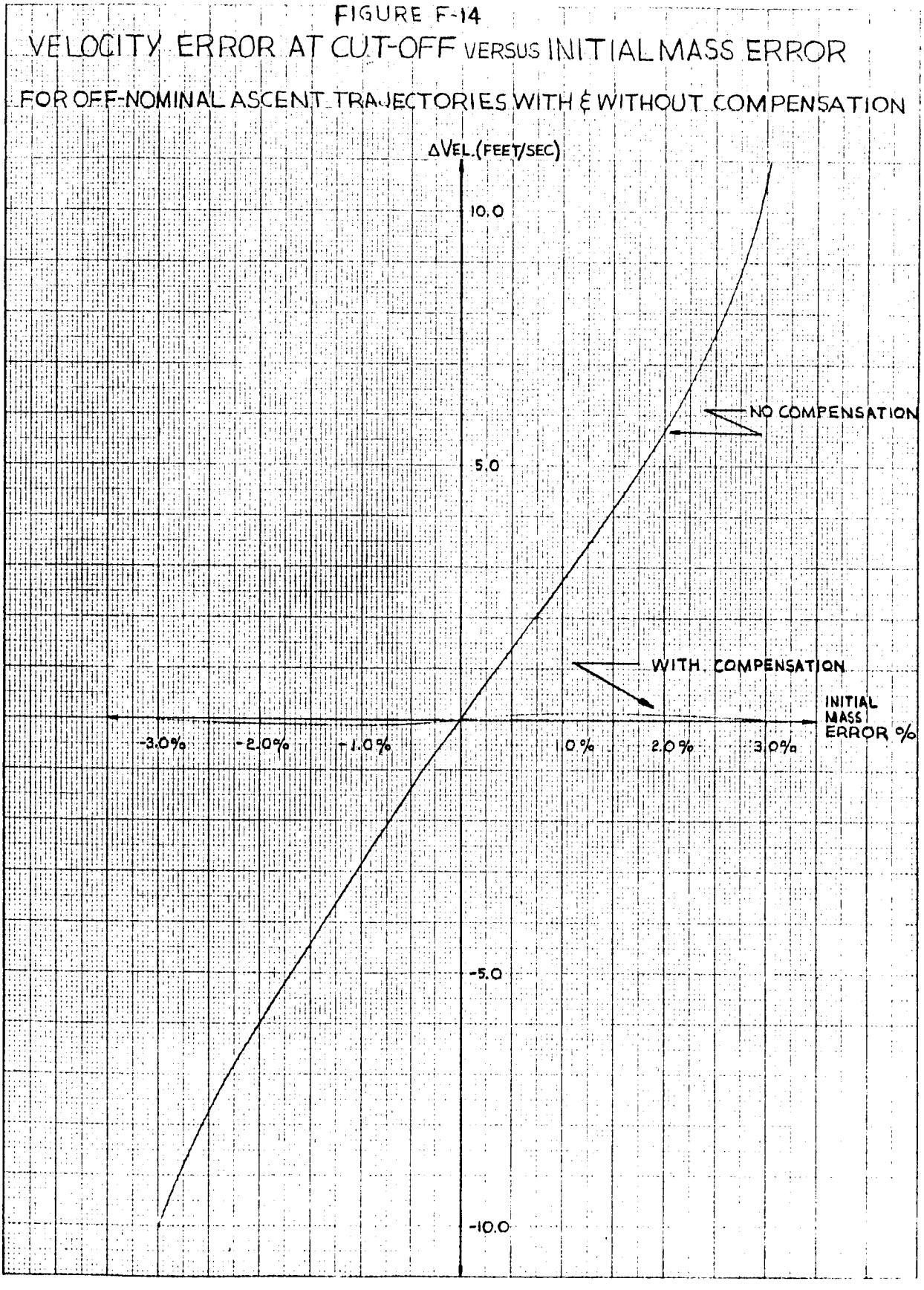


FIGURE F-14

VELOCITY ERROR AT CUT-OFF VERSUS INITIAL MASS ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION



CONFIDENTIAL

FIGURE F-15

ALTITUDE ERROR VERSUS INITIAL MASS ERROR

FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

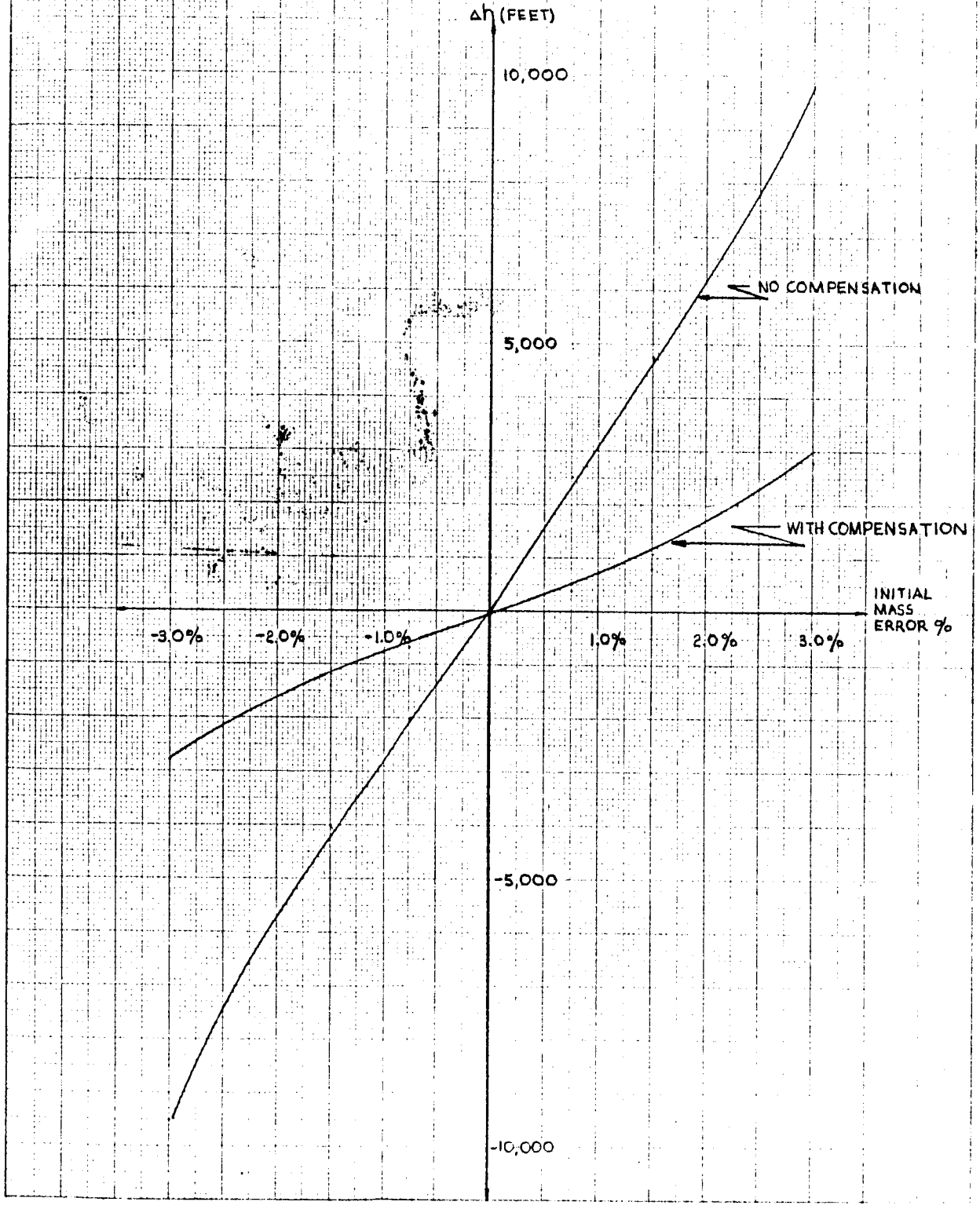


FIGURE F-16

CHANGE IN CUT-OFF TIMES OF COMP. MINUS UNCOMP. SYSTEMS
VERSUS
INITIAL MASS ERROR
FOR OFF-NOMINAL ASCENT TRAJECTORIES WITH & WITHOUT COMPENSATION

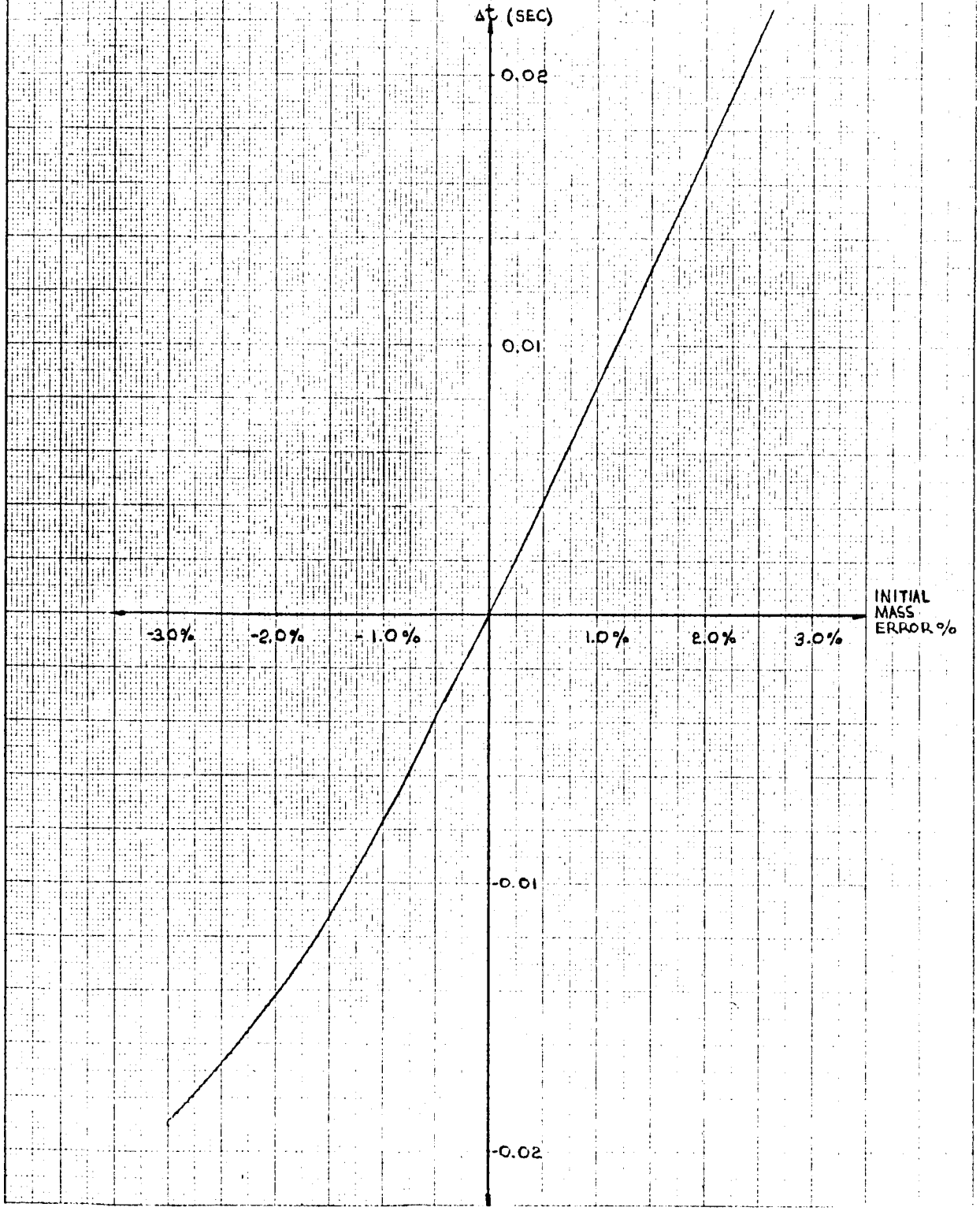


TABLE F-VI

MINIMUM RANGE TO CSM AND THRUST CUT-OFF ERRORS FOR THRUST MISALIGNMENT - WITH AND WITHOUT "T_1 COMPENSATION"

INPUT ERROR	MIN. RANGE TO CSM	FLIGHT PATH ANGLE ERROR	VELOCITY ERROR	ALTITUDE ERROR
PITCH AXIS THRUST MISALIGNMENT +0.3 deg. COMPENSATED	6,702.7 ft.	+0.002 deg.	1.2 ft/sec	5.0 ft.
UNCOMPENSATED	54,115.5 ft.	-0.282 deg.	8.0 ft/sec	-3,466.7 ft.
PTICH AXIS THRUST MISALIGNMENT -0.3 deg. COMPENSATED	3,278.0 ft.	-0.002 deg.	-1.2 ft/sec	3.7 ft.
UNCOMPENSATED	45,866.7 ft.	+0.282 deg.	-8.0 ft/sec	3,482.5 ft.

times in the hundredths of seconds must be accurately obtained to minimize errors at intercept. This effect is shown graphically in Figures F-5, 10, and 15.

The "t_1 Compensation" method can not correct for guidance hardware errors (e.g. gyro drifts, pitch rate uncertainties). These hardware errors, however, to have effects upon the minimum range to the CSM at intercept. It has been found that gyro drifts of 1 deg/hr (0.00028 deg/sec) cause approximately 1 nautical mile miss at intercept, and a pitch rate uncertainty of 0.001 deg/sec cause miss at the CSM of about 3 nautical miles.

Further studies are now underway to determine the effectiveness of "t_1 Compensation" when considering:

1. Varying thrust and varying Isp during the powered ascent.

2. Abort from powered descent and hover.

These studies have further revealed that the major portion of the velocity error at thrust cutoff is compensated by the "kick angle," and only a small part (approximately 1.5 ft/sec out of a total of 25 ft/sec for 6% thrust magnitude error) is attributable to the adjustment of the ΔV cutoff. It may, therefore, be possible to reduce the compensation system complexity by eliminating the ΔV cutoff compensation. This would add, at most, an additional miss of about 1.2 n. mi. at the CSM.

G. SUMMARY OF SYSTEMS PERFORMANCE REQUIREMENTS

Aim of the Study

The primary aim of the study, the preliminary results of which are described in this report has been to obtain the performance requirements for a back-up guidance system which can perform the nominal ascent from the lunar surface, as well as the abort from any phase of the LEM mission. In order to establish these requirements, a general theory of ascent and abort guidance has been developed, and specific techniques analyzed for applicability and feasibility with respect to the back-up guidance problem. Operational procedures are proposed to reduce the requirements of the back-up guidance system performance without significant penalties.

Guidance Technique

The prime criteria in developing a guidance technique is simplicity in concept and implementation, minimum hardware complexity and maximum allowable equipment tolerances.

The powered ascent trajectory used in this study is described in Section E. The trajectory represents a simplified approximation to an optimum ascent and consists of a vertical rise, followed by two constant pitch rate phases. An inertially fixed attitude reference and a simple programmer command the spacecraft along the desired trajectory. The allowable tolerances on powered ascent burn-out conditions are dictated by the more stringent of the two requirements to achieve a) a 40,000 ft. clear pericynthion trajectory and b) to come to within at least 10 n.mi. of the CSM at the point of closest conjunction for rendezvous initiation.

Studies performed by GAEC with respect to back-up guidance and by MIT in connection with the primary navigation and guidance system have shown that a midcourse guidance and correction technique will be required to assure proper conditions for rendezvous initiation. The midcourse guidance scheme developed for the back-up operation is described in Section F. A perturbation technique has been selected to achieve computational simplicity, in which the "observational residuals", i.e. the difference between known nominal data and radar derived data, form the basis for the determination of the corrective impulse required. The results of the studies investigating the range of applicability and the efficacy of the "Delta" midcourse guidance technique are presented in Tables F-I and F-II of Section F from which these significant conclusions may be drawn: effective midcourse correction is possible for a large range of deviations in burn-out conditions, provided the measurement accuracies are fairly good. Thus, if midcourse correction provisions are incorporated into the back-up guidance philosophy, tight control of burn-out conditions for the purpose of assuring rendezvous does not represent a governing constraint on the equipment tolerance requirements. However, the same attitude reference used during the powered ascent phase also serves as the reference for the angular observational residuals during midcourse and, as can be seen from Table F-III, a drift rate of about 1°/hr represents the tolerable limit for achieving the 10 n.mi. distance of closest approach to the CSM. Consequently, no more than 1°/hr drift in the attitude reference gyros is allowable even though this performance is not required from the point of view of control of the burn-out conditions.

Considering the problem of assuring a clear pericynthion orbit upon termination of the powered ascent, the question arises as to what are the nominal and worst off-nominal conditions at burn-out. The nominal burn-out conditions inject the LEM into a coasting transfer orbit which intersects the CSM orbit at some central angle from burn-out. This central angle may be nominally 180°, and vary between 140° to 270° for various plane change or abort situations. Under certain conditions, burn-out may result in injection into a nominally circular low altitude orbit. For example, an emergency ascent from the lunar surface may require injection into a parking orbit until the proper phasing for ascent into the CSM orbit has been attained (see Reference below). Short launch delays may also require temporary phasing in a parking orbit up to a maximum of about two hours. Another situation in which injection into a low altitude parking orbit may be required is an abort from powered descent. The transfer orbit for an abort just prior to starting of the powered descent phase requires an approximately 270° central angle to rendezvous since the abort consists of simply remaining in the synchronous descent transfer orbit. About one minute after reaching the hover point, the proper phasing exists for a 180° Hohmann transfer. Thus, the central angle range of possible transfer trajectories following abort covers the 90° sector between 270° and 180°. Since midcourse corrections are required for aborts from powered descents, considerable difficulty is introduced by attempting to use the Delta midcourse guidance technique over such a large range of off-nominal transfer trajectories. The present concept is to establish a small number of reference transfer orbits to CSM rendezvous, and store a set of guidance constants for each reference orbit. Since the shorter central angle transfer orbits require smaller LEM lead angles with respect to the CSM at transfer initiation, it is clear that transfers with larger central angles can always be attained by inserting the LEM into a low altitude phasing orbit for the short time period necessary to achieve the increased lead angle. To shorten the transfer central angle, however, requires a long (up to about 18 hours) staytime in the parking orbit. Consequently, the longest (266°) transfer orbit must be one of the reference trajectories, and the others are selected to achieve reasonable short maximum phasing times without introducing too large a set of constants to store in the back-up guidance system. If the 266° orbit were the only abort reference orbit, and abort just prior to touchdown would extend this time period to about 58 minutes. The maximum dwell time in parking orbit following abort from powered descent can be reduced by a factor essentially equal to the number of discrete reference orbits for which sets of constants are supplied. Figures G-1 and G-2 show the abort paths, the dwell times in parking orbit, ΔV requirements and central angle of transfer orbits for the case of two reference trajectories. If the two reference trajectories are selected to minimize the abort ΔV, the maximum staytime in parking orbit becomes 36 minutes.

In any event, there is a distinct probability that a situation will arise, in which injection into low altitude orbits may be required. Thus, the component requirements must be examined from the point of view of meeting the clear pericynthion constraint for low altitude orbit injection.

A General Study of Elliptic Transfer Orbits Between LEM and CSM Lunar Parking Orbits - LMO-500-30.

FIGURE G-1

ABORT REFERENCE TRAJECTORY GEOMETRY

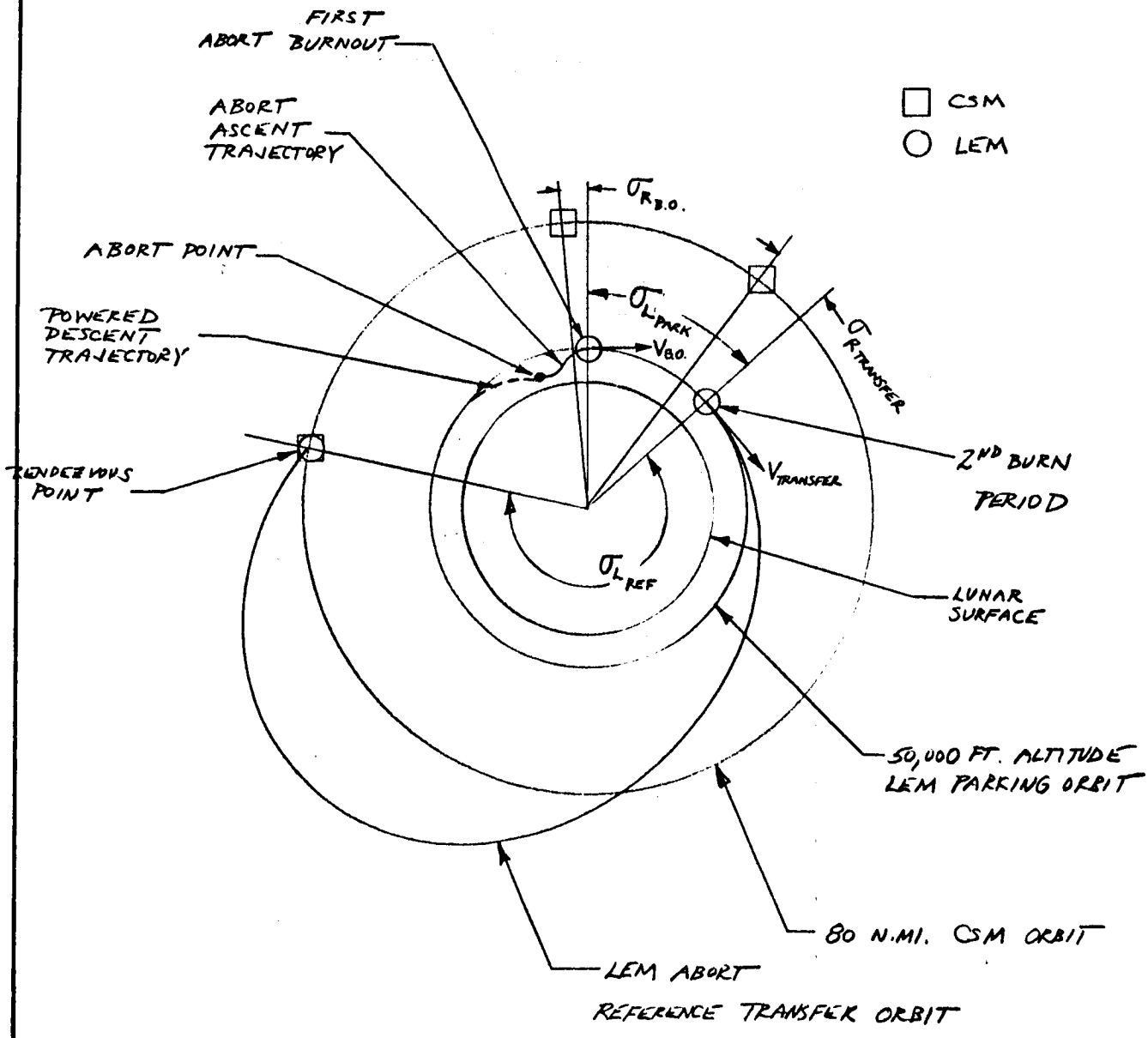


FIGURE G-2 ABORT TRAJECTORY PARAMETERS FOR TWO REFERENCE TRAJECTORIES

TRANSFER (50,000 FT CIRCULAR PARKING ORBIT ALTITUDE TO 50 N.MI. CSM ORBIT ALTITUDE)

t_{PARK} ~ MIN.

(a) PARKING ORBIT DWELL TIME

MAX. t_{PARK_1} (OCCURS PRIOR TO SWITCH PT.

MAX t_{PARK_2} (AFTER SWITCH PT.) (OCCURS AT HOVER AT. +2 MIN t_d)

22.4 MIN.

MINIMUM R.V. SWITCH POINT t_{d_s} = 327.6 SEC

$\sigma_{TRANSFER_2}$ ~ DEG

$\sigma_{L_{REF_2}}$ ~ DEG

$\sigma_{L_{REF_2}}$, ASCENT POLAR ANGLE OF LEM DURING TRANSFER FOR ABORTS AFTER t_{d_s}

$\sigma_{L_{REF_1}}$ = 265.6° FOR ABORTS PRIOR TO t_{d_s}

236.6°

(b) 2ND REF TRAJECTORY ASCENT CENTRAL ANGLE

NOTE: FOR A SINGLE REF. TRAJECTORY ($\sigma_{L_{REF}}$ = 265.6°)

$\sigma_{R_{TRANSFER}}$ = +8°

ΔV_{MAX} = 6615 FT./SEC.

MAX t_{PARK} = 58.3 MIN (OCCURS AT HOVER AT. +2 MIN. t_d)

$\sigma_{R_{EOI}}$ AT SWITCH PT. = $\sigma_{R_{TRANSFER}}$ FOR ABORTS AFTER t_{d_s}

-4.2°

$\sigma_{R_{TRANSFER}}$ FOR ABORTS PRIOR TO t_{d_s} = +8°

$\sigma_{R_{EOI}}$ AT HOVER +2 MIN t_d = -11.8°

(c) 2ND REF TRAJECTORY σ_R AT TRANSFER PT.

ΔV ~ FT/SEC

6480 FT/SEC.

MAX ΔV AFTER SWITCH PT. (OCCURS AT HOVER +2 MIN t_d)

(d) MAX. ABORT ΔV *

MAX ΔV PRIOR TO SWITCH PT.

* INCLUDES 45 FT/SEC DOCKING ΔV, 150 FT/SEC MIDCOURSE + OUT-OF-PLANE (ΔV), & MAN. ASCENT + MEN. LIMITS ΔV

t_{d_s}, TIME FROM INITIATION OF NOMINAL FLIGHT TO REF. TRAJ. SWITCH PT. SEC

The open loop error studies performed thus far show that variations in thrust-to-weight ratio would have the largest effect on the deviations in burn-out conditions. This is true even if burn-out is commanded as a function of measured ΔV rather than as a function of time. The curves of Figures F-1-F-15 in Section F show the burn-out errors as well as the associated miss-distance to the CSM (for the case of no midcourse correction) as a function of various errors connected with thrust-to-weight ratio. For example, for less than 4% engine thrust deviation, a flight path error at burn-out of 1° results. The "t_1 compensation" scheme, described in Section F has been developed in an effort to eliminate the effect of deviations in thrust-to-weight ratio from its nominal value. The success with which it accomplishes this objective can be evaluated by comparing the "with t_1 compensation" curves with the "no compensation" curves of Figures F-1 through F-15. In fact, the "t_1 compensation" scheme is so effective that T/W variations can, for all practical purposes, be eliminated from consideration as far as their contribution to burn-out errors are concerned. The "t_1 compensation" technique, does not, however, correct for back-up guidance system component errors, and the effect of these on burn-out conditions must be considered. An extensive component error analysis has shown that the only significant contributary sources to burn-out errors are alignment and drift of the attitude reference. The results of a parametric study of the effect of these errors are shown on Figures G-3 through G-6. The gyro drift value indicated on the curves is the sum of random plus acceleration sensitive drift normalized for the average acceleration throughout the powered ascent phase. The important conclusions are those that can be drawn by reference to Figure G-6. For a reasonable value of initial alignment accuracy of, say, 0.1°, 40,000 ft. clear pericynthion is not possible even with zero additional attitude drift caused by the gyros. With 0.05° initial alignment accuracy, gyro drift rate must be kept below 0.3°/hr. Thus, it would appear as if the clear pericynthion conditions would impose the controlling constraint on equipment performance.

These tight tolerance requirements can be alleviated, however, by biasing the nominal burn-out conditions. For example, it is clear that if burn-out occurs at a velocity in excess of that corresponding to circular orbital velocity at burn-out altitude, velocity magnitude errors up to at least the circular excess velocity will not reduce the pericynthion below that of the burn-out altitude. Thus, the sensitivity of pericynthion altitude to velocity errors would have been greatly reduced. It turns out, that the same effect occurs with respect to sensitivity to flight path angle errors. This is graphically demonstrated by the curves of Figures G-7 and G-8. The lower curves illustrate the allowable errors in flight path angle and velocity magnitude, respectively, for a given increase in ΔV above that for circular orbit. For zero excess ΔV, the allowable tolerances are 0.1° and 2.5 fps, respectively, confirming the conclusion reached above with respect to 0.1° initial alignment of the attitude reference.

FIG. G-3

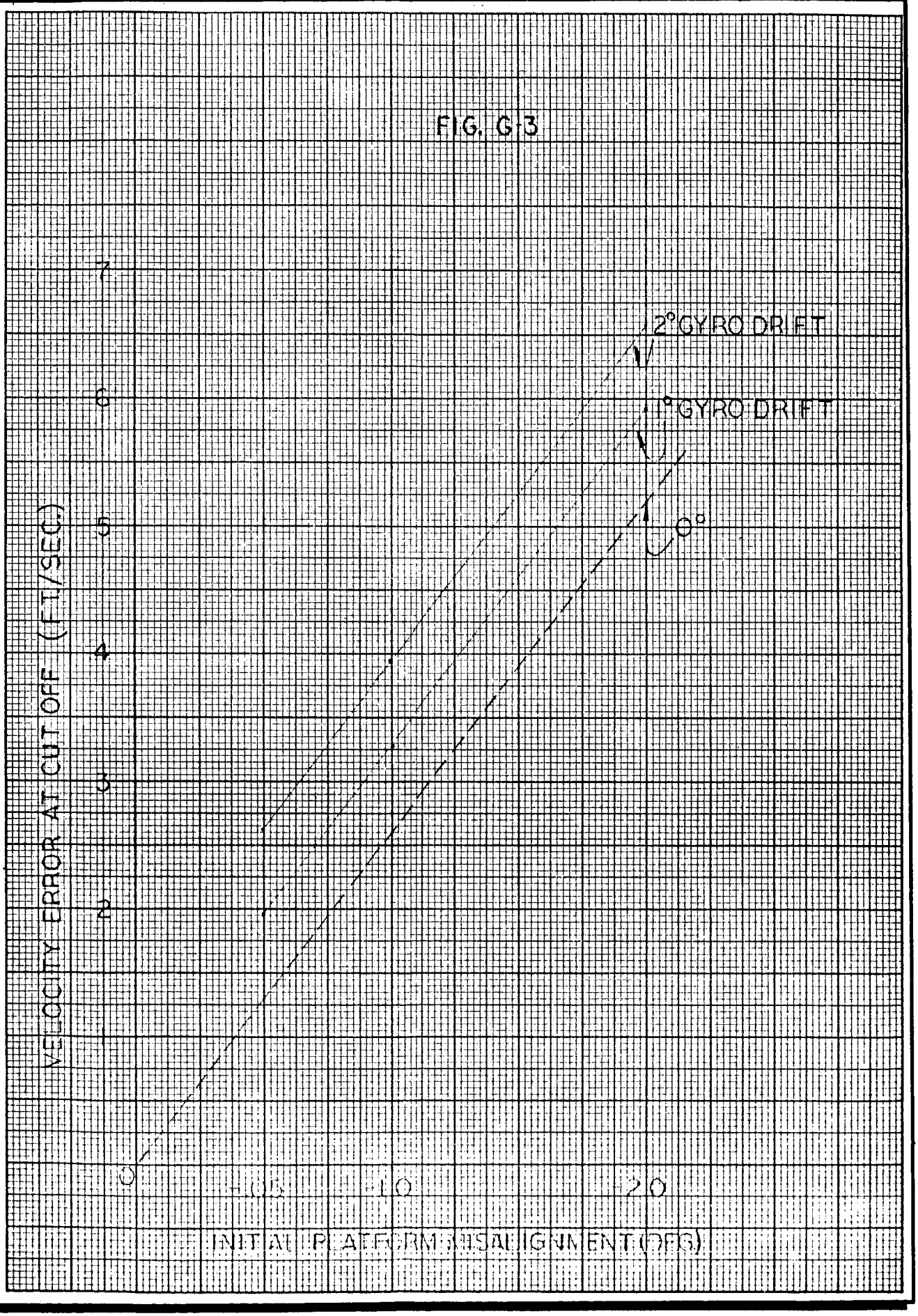

2° GYRO DRIFT
1° GYRO DRIFT
0°
VELOCITY ERROR AT CUT OFF (FT/SEC)
7
6
5
4
3
2
1
0
10
20
INIT AL PLATFORM MISALIGNMENT (DEG)

FIG. G-4

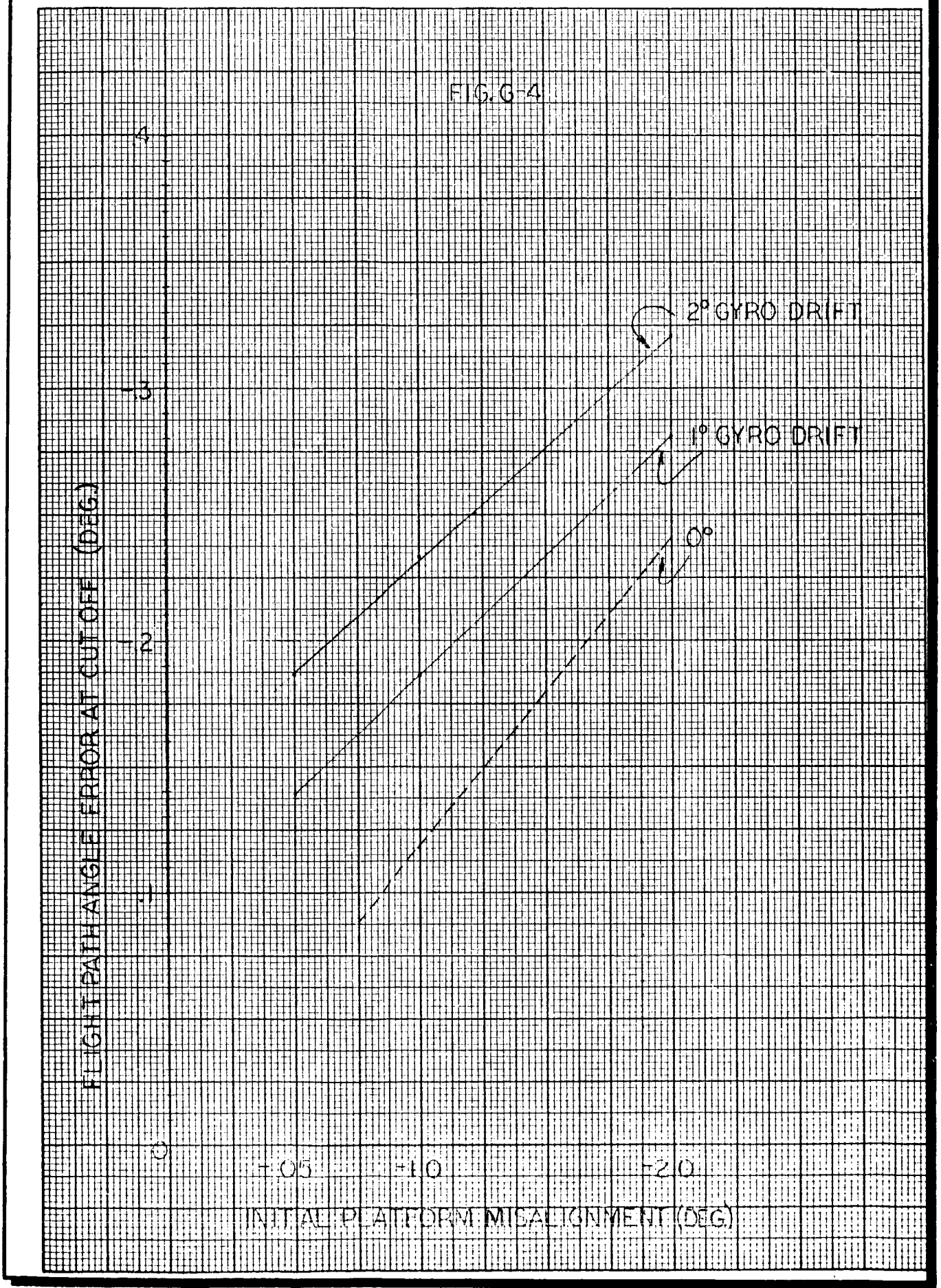

2° GYRO DRIFT
1° GYRO DRIFT
0°
FLIGHT PATH ANGLE ERROR AT CUT OFF (DEG.)
-4
-3
-2
-1
0
-.05
-1.0
-2.0
INITIAL PLATFORM MISALIGNMENT (DEG)

FIG G-5

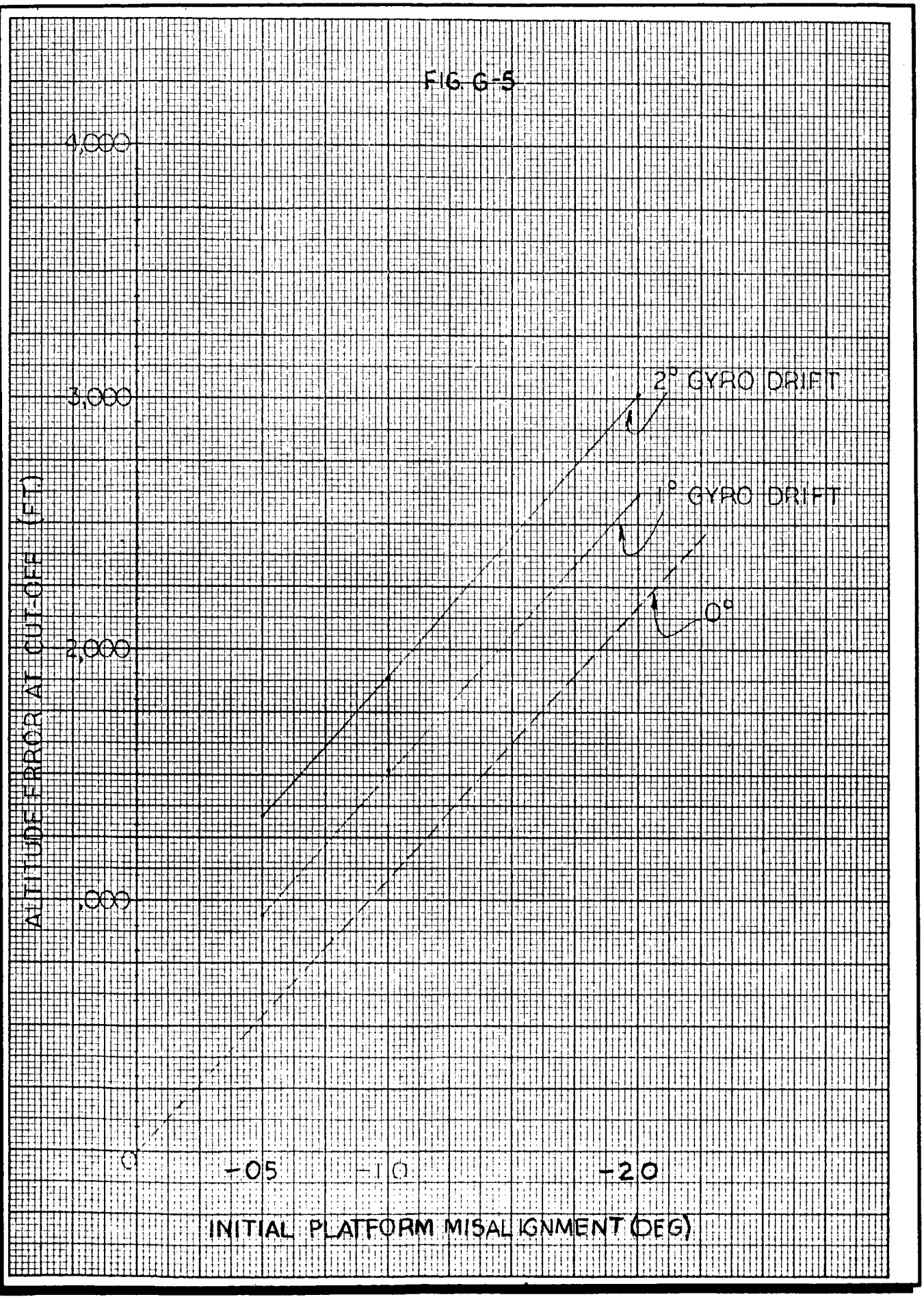
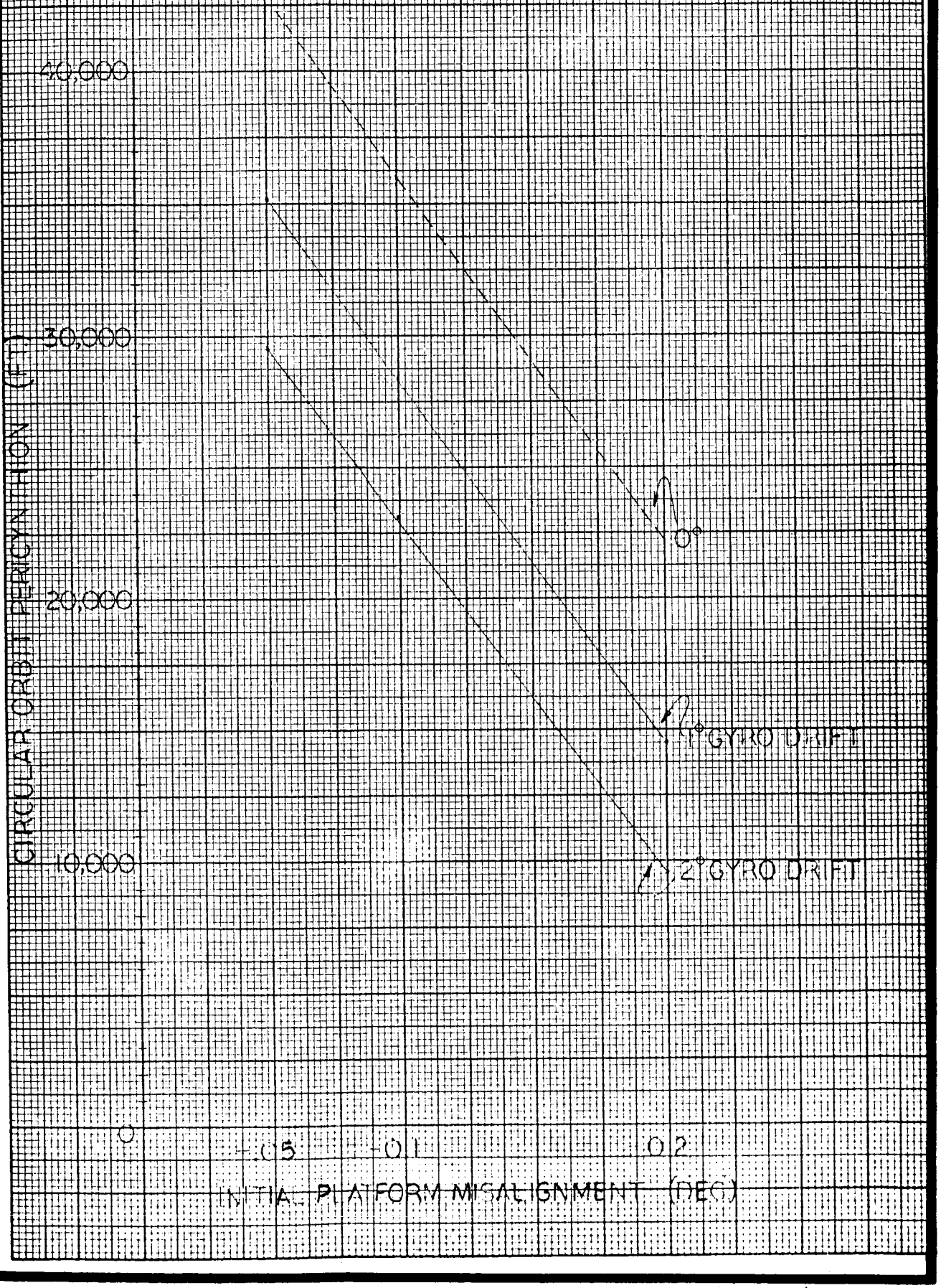

4,000
3,000
2,000
,000
0
2° GYRO DRIFT
1° GYRO DRIFT
0°
-05
-10
-20
ALTITUDE ERROR AT CUT-OFF (FT)
INITIAL PLATFORM MISALIGNMENT (DEG)

FIG. G-6


40,000
30,000
20,000
10,000
0
CIRCULAR ORBIT PERIGYNTHION (FT)
0°
1° GYRO DRIFT
2° GYRO DRIFT
.05
.01
.02
INITIAL PLATFORM MISALIGNMENT (DEG)

[redacted]

FIGURE G-7
EFFECT OF FLIGHT PATH ANGLE ERROR ON ΔV REQUIRED TO ACHIEVE A 40,000 FT. CLEAR-PERICYNTHION PARKING ORBIT

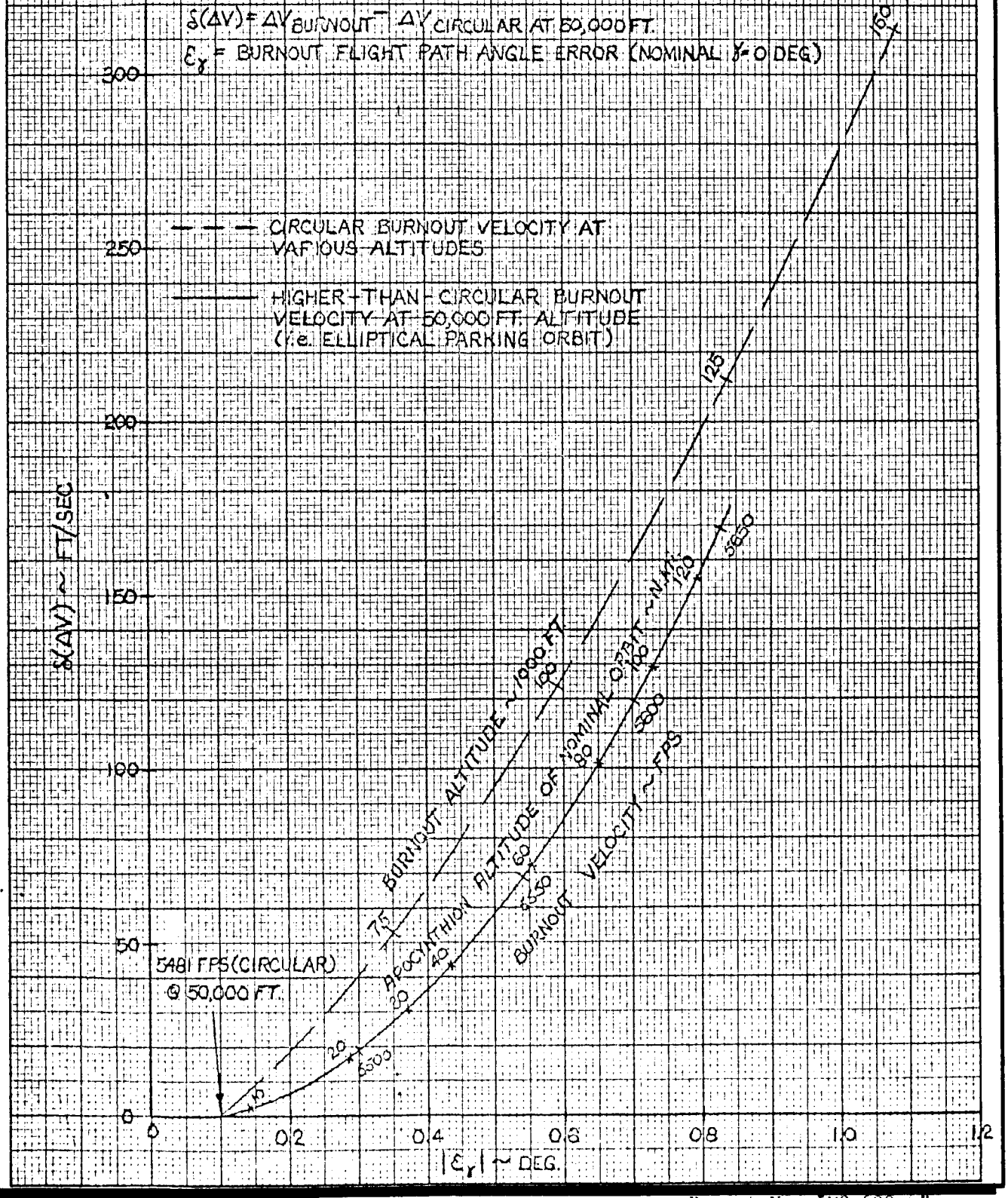


FIGURE G-8

EFFECT OF BURNOUT VELOCITY TOLERANCE ON ΔV REQUIRED TO ACHIEVE A 40,000 FT CLEAR-PERICYNTHION PARKING ORBIT

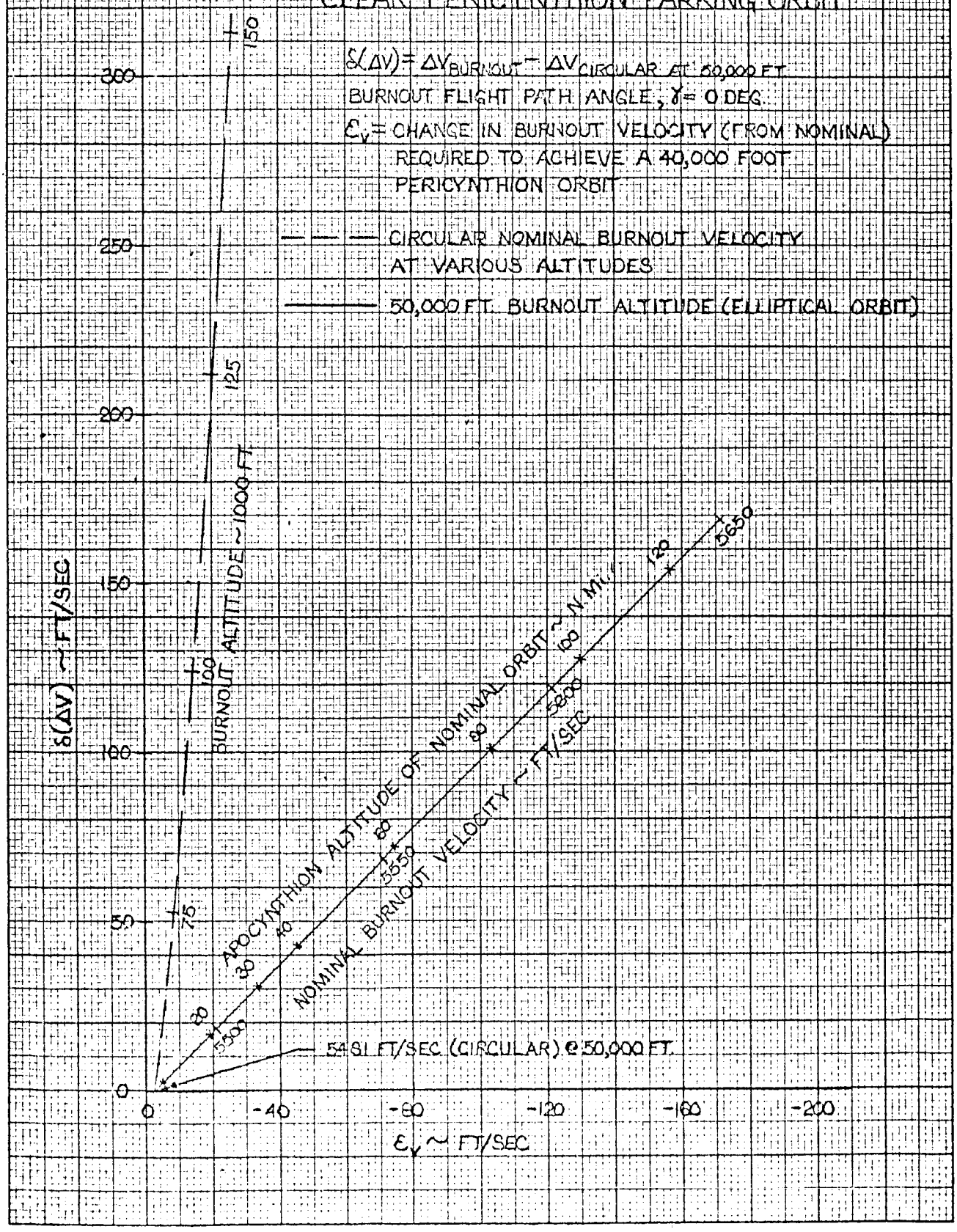
$\delta(\Delta V) = \Delta V_{BURNOUT} - \Delta V_{CIRCULAR\ AT\ 50,000\ FT}$

BURNOUT FLIGHT PATH ANGLE, $\gamma = 0$ DEG.

$\mathcal{E}_V$ = CHANGE IN BURNOUT VELOCITY (FROM NOMINAL) REQUIRED TO ACHIEVE A 40,000 FOOT PERICYNTHION ORBIT

— — — CIRCULAR NOMINAL BURNOUT VELOCITY AT VARIOUS ALTITUDES

——— 50,000 FT. BURNOUT ALTITUDE (ELLIPTICAL ORBIT)



For a 15 fps excess velocity, however, the allowable tolerances become 0.27° and 18 fps, which, from Figures G-3 and G-4 can be attained by a combination of 0.175° initial alignment accuracy and 1°/hr drift rate.

The upper curves of Figures G-7 and G-8 show the burn-out altitude to which the nominal ascent trajectory would have to be biased to assure a 40,000 ft. pericynthion with the same values of flight path angle and velocity errors. The ΔV penalty resulting from increasing the nominal burn-out altitude is described in GAEC/LEM report LMO-500-52. As an example, the 15 fps excess velocity case referred to above may be considered. From Figure G-7 , an equivalent sensitivity to flight path angle is attained by biasing the burn-out altitude to about 65,000 ft. and injecting into a circular orbit at that altitude. At a (T/W) of 0.42°, the ΔV penalty for this higher altitude ascent is between 25 and 30 fps. For an equivalent velocity sensitivity, a burn-out altitude of about 115,000 ft. is required with an attendant ΔV penalty of almost 150 fps. Thus, velocity biasing is by far the more economical method of reducing pericynthion sensitivity to burn-out errors. In fact, the velocity bias does not represent a ΔV penalty, since the ΔV for injection into the transfer orbit can generally be reduced by that amount.

Thus, in order to extend the equipment tolerances for an ascent requiring a stay-time in a parking orbit, the burn-out conditions should be biased by about 15 fps in velocity, making the parking orbit slightly elliptical. There is one adverse factor resulting from this operational concept. The synodic time for two orbits, i.e. the time interval between equal phasing conditions for spacecraft in these orbits, increases as the energies of the orbits approach each other. Thus, for the higher energy elliptical parking orbits, a greater maximum in-orbit staytime results. The relationship between circular velocity excess, apocynthion altitude and synodic time is shown on Figure G-9 . For a 15 fps excess velocity, the maximum synodic time increases from 17.6 hrs. to 18.8 hrs. This is the time that would have to be spent in the parking orbit for an emergency lunar launch just prior to the maximum early launch time permitted by the ΔV available. This does not, however, require a change in the basic design mission profile because the increase in maximum synodic time can only be realized in practice if there is a simultaneous occurance of both an emergency condition which requires immediate ascent as well as a failure in the primary Navigation and Guidance System. With the primary Navigation and Guidance System operating, velocity biasing is not required, since the burn-out errors can be kept small enough to assure clear pericynthion even with circular orbit injection. Thus, 18.8 hrs. staytime is associated with a double failure situation, whose probability of occurrence is too small to be considered in design.

There is, however, another small penalty attendant upon velocity biasing which must be taken into account in the design. The maximum dwell time in parking orbit following an abort from powered descent increases by about five minutes over the maximum of 58 minutes for circular orbit burn-out.

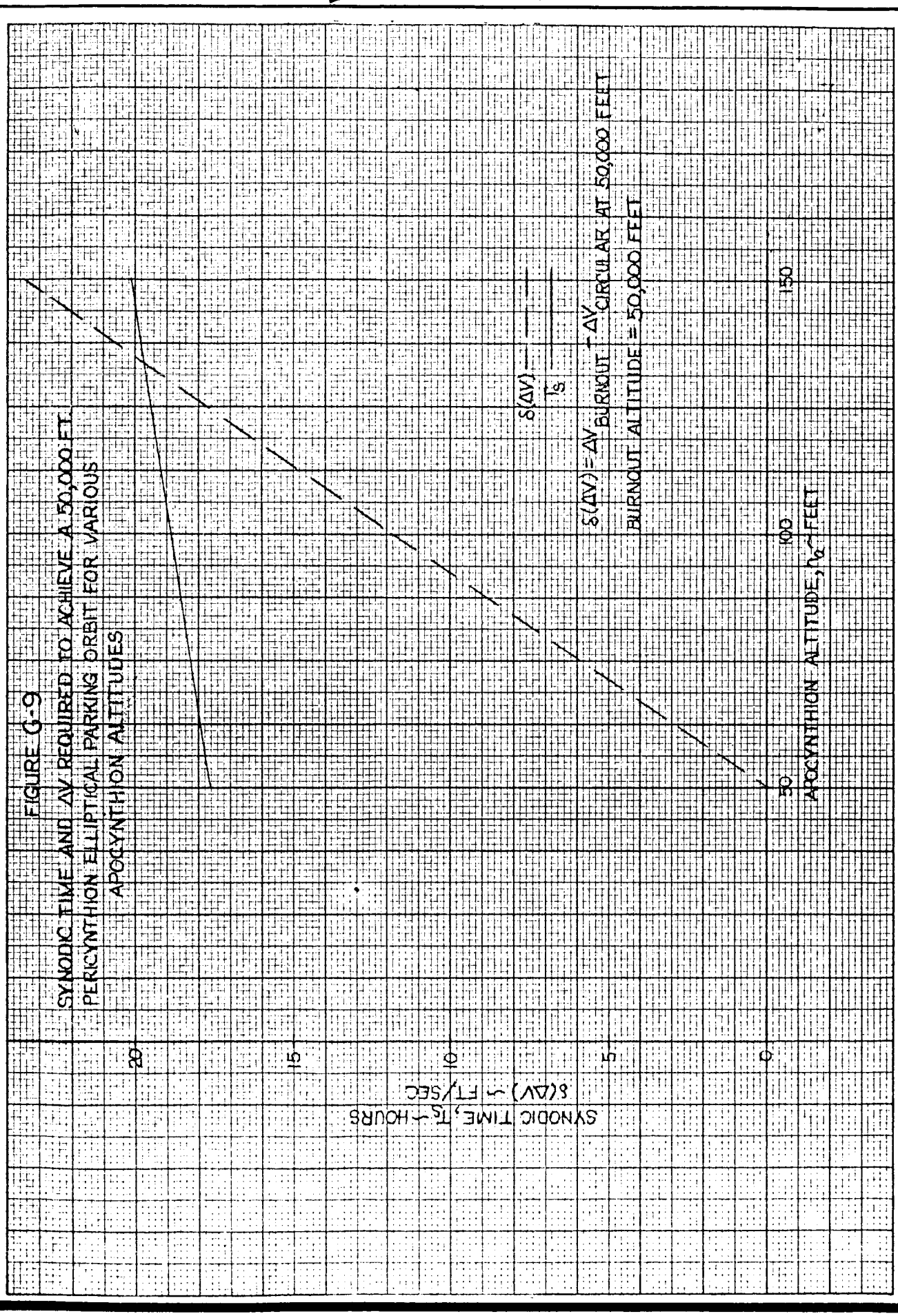


FIGURE G-9
SYNODIC TIME AND ΔV REQUIRED TO ACHIEVE A 50,000 FT. PERICYNTHION ELLIPTICAL PARKING ORBIT FOR VARIOUS APOCYNTHION ALTITUDES

This has a small effect on the allowable drift rate tolerance during the coast phase prior to midcourse correction.

Performance Requirements

General

The following performance requirements are applicable to a back-up system configuration consisting of any attitude reference, a set of three, body axis mounted, integrating, accelerometers and an abort programmer. The "t_1 compensation" technique will be incorporated in the programmer mechanization. Any additional mechanization required for the Delta midcourse guidance technique will be defined at a later date. The back-up system attitude reference can be aligned to the landing site coordinate system when on the lunar surface and to the primary system inertial coordinates while in flight. The programmer must receive initial velocity conditions as inputs and select the appropriate abort program required by the initial conditions. Detailed performance specifications for the abort programmer will be defined later, the information above being included for reference only. The ac celerometer performance requirements are based on limiting the 1σ burn-out velocity error caused by accelerometer errors to 1 fps.

Attitude Reference

1. Alignment about "vertical axes"
   On lunar surface $\leq$ 10 arc minutes (3σ)
   In flight (transfer acc.) $\leq$ 3 arc minutes (3σ)

2. Alignment in "Azimuth"
   On lunar surface = 10 arc minutes (3σ)
   In flight (transfer acc.) = 3 arc minutes (3σ)

3. Attitude Drift Rate
   Net drift rate as measured over any five minute interval in an equivalent 0.6 g field = 0.5 deg./hr.

4. Range

   Angular Range
   All attitude capability

   Angular Rate
   Maximum performance range = 20°/sec about all axes
   Maximum operating limits = 30°/sec about all axes
   Minimum tracking rates through pole = 15°/sec.
   Nominal operating range = 0°- 10°/sec.

Integrating Accelerometer (body mounted) (1σ)

1. Thrust axis accelerometer (x body axis)
   Misalignment to thrust axis = 30 arc min. (11 to x body axis)
   Accelerometer bias error = 50 ug
   Scale factor error = 1×10^{-4} g/g
   Acceleration range = 1 fps^2 - 33 fps^2
   Velocity output resolution = 0.1 fps
   Velocity output range = 0 to 6050 fps

2. Cross axis accelerometers (Y & Z body axis)
   Misalignment to cross axis = 1.5 arc min. (1 to x axis accel. in axis)
   Accelerometer bias error= 50 ug
   Scale factor error = 5×10^{-3} g/g
   Acceleration range = 0 to 3.0 ft/sec^2
   Velocity output resolution = 0.1 fps
   Velocity output range = 0 to 100 fps

Performance Capabilities

A preliminary estimate of the 3σ performance capabilities of a back-up guidance system meeting the specified requirements can be made. In arriving at this estimate, it is assumed that all errors are independent and thus covariance effects are neglected. The 3σ burn-out errors for either injection into low altitude near circular orbit or direct injection into transfer orbit to the CSM will be as follows:

Velocity magnitude error $\approx$ 60 fps
Flight path angle error $\approx$ 0.225°
Altitude error $\approx$ 2100 ft.
Pericynthion altitude (with 15 fps velocity bias) $\approx$ 40,000 ft.

Miss distance to CSM for 266° transfer orbit (with midcourse correction and no dwell time in a phasing orbit) $\approx$ 8 N.Mi

Miss distance to CSM for 266° transfer orbit (with midcourse correction and maximum dwell time in orbit) $\approx$ 10 N.Mi

In arriving at these performance capabilities, the velocity and position errors at the time of abort and their propagation to burn-out conditions have not been included. These initial condition errors will, of course, increase the burn-out errors over the values cited, and thus, may require an increase in the velocity bias. This should not, however, significantly affect the design of the back-up guidance system.

H. BACKUP GUIDANCE SUBSYSTEM CONFIGURATION STUDIES

Introduction

A Backup Guidance system must contain an attitude reference, capable of all-attitude operation and of providing vehicle orientation data for the attitude display, and a triad of integrating accelerometers to measure integrated specific force. A programmer would also be required to instrument the particular guidance techniques discussed in section F.

Two Backup Guidance System Configurations have been studied with respect to the general requirements and are shown in Figures H-1 and H-2. Alignment of the back-up guidance system on the lunar surface with its own levelling sensors must be accomplished in conjunction with the primary system, the OMU and possibly the tracking radar.

A trade-off study technique has been developed for each of the systems under investigation. The study assigns weighting factors to reliability, weight, power, size, interface and complexity, and development risk. A comparison has been made between the two configurations, but a relative rating has not yet been assigned.

Strapped Down Configuration

The inertial sensor package contains three body mounted gyros operating in a pulse torque rebalanced mode. Each gyro output pulse is proportional to an incremental angular rotation which drives the Digital Differential Analyzer. Three linear integrating accelerometers supply digital incremental velocity for direct guidance and control use.

The Digital Differential Analyzer coordinate converter takes incremental angular rotation information from the gyros in the inertial sensor package and generates the nine direction cosines which relate body attitude to inertial coordinates. A further conversion to equivalent gimbal angles for display may also be required. The error command signals are generated by comparing computed with pre-programmed direction cosines. These command signals are then transformed into body coordinates and converted to analog signals to activate the reaction control system.

Gimbal Platform Configuration

The platform supplies gimbal angles, which are equivalent to the Euler angles relating body attitude to inertial coordinates, directly to a digital programmer, displays and control electronics section. The A/D conversion is performed in the programmer. An error command signal is generated by comparing pre-programmed with measured gimbal angles. The error signals are then transformed to body co-ordinates and converted to analog control signals in the same manner as for the strapped-down configuration.

Alignment of the Backup Guidance System

The inertial reference must be aligned before initiation of powered descent or ascent and, depending on its drift, updated at various intervals from the primary system. There are three basic sensors which can be used to align the backup guidance system: the primary system, the OMU, and the radar.

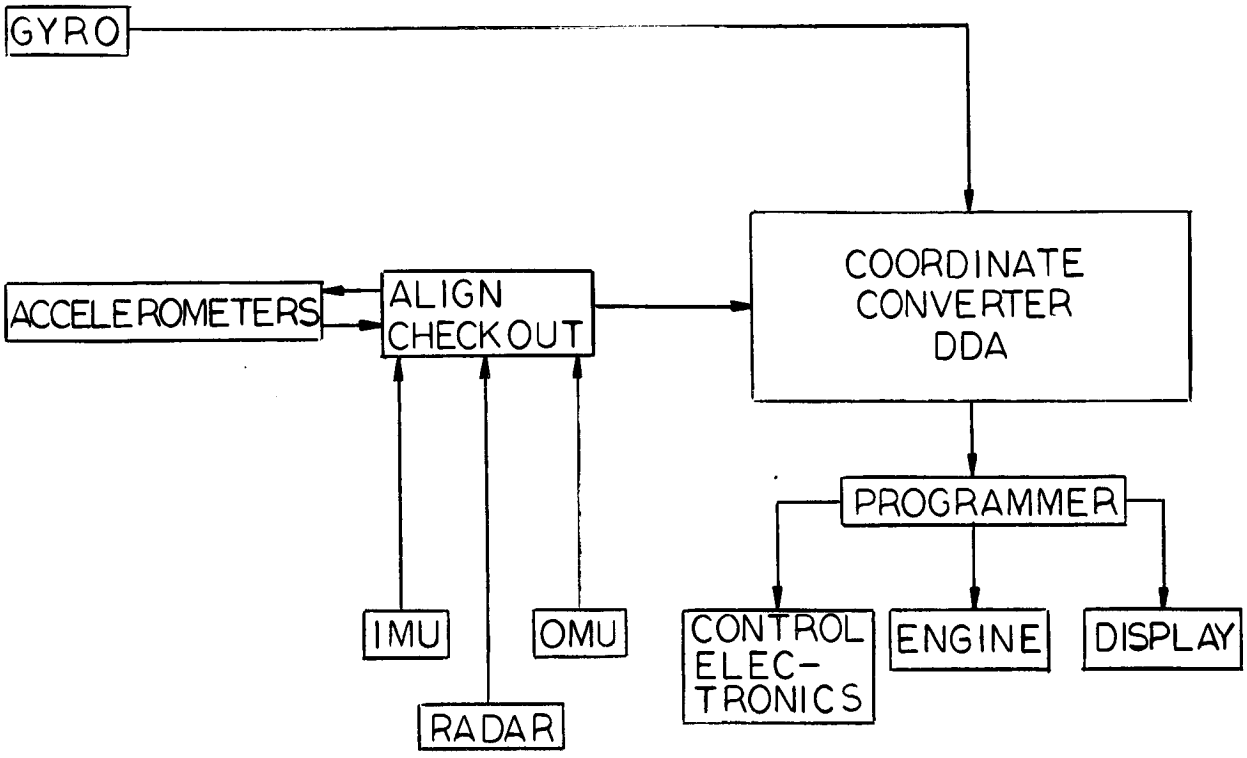


FIGURE H-1
FUNCTIONAL BLOCK DIAGRAM
OF THE
STRAPPED DOWN BACKUP GUIDANCE SYSTEM

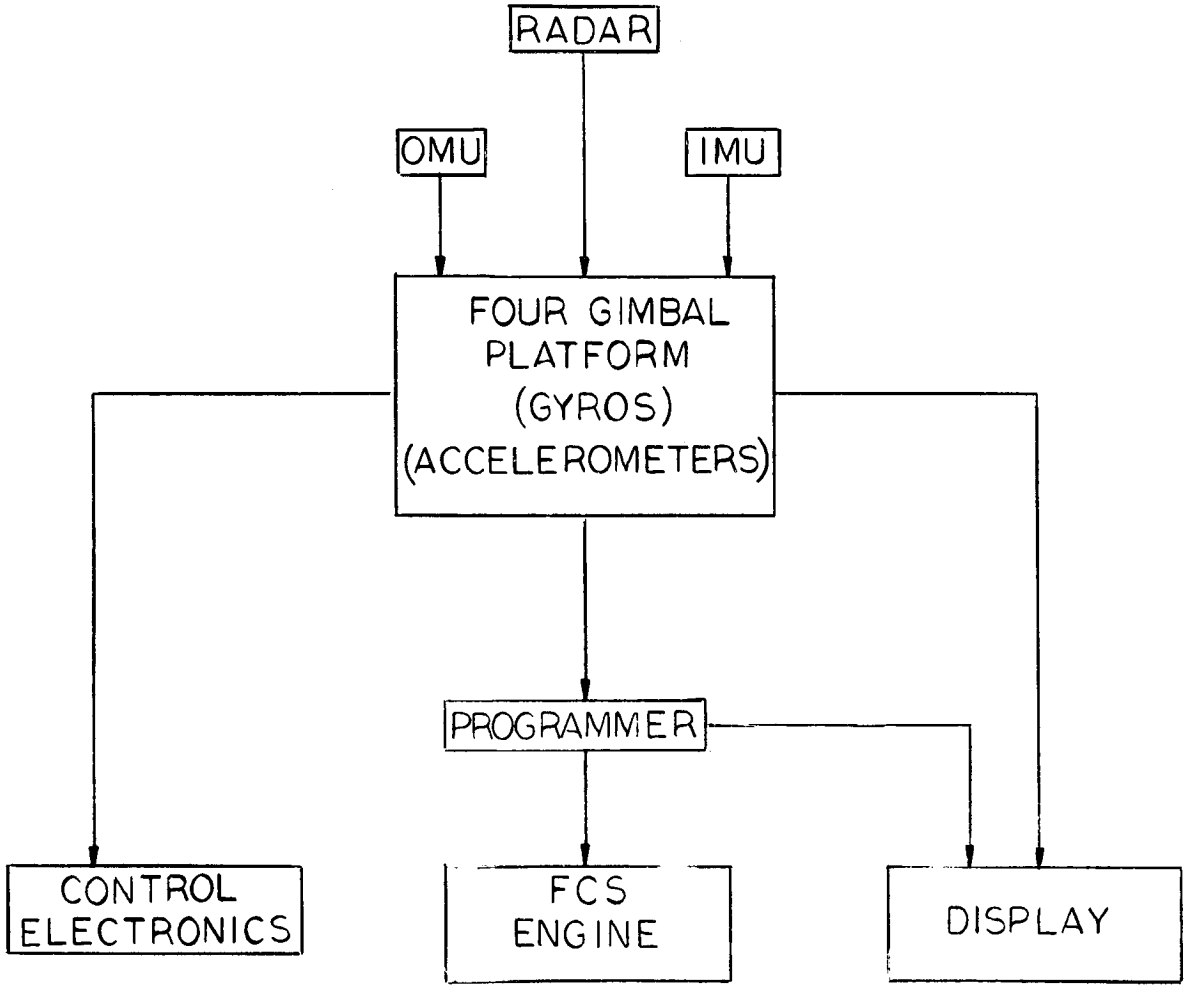


FIGURE H-2
FUNCTIONAL BLOCK DIAGRAM
OF THE
PLATFORM BACK-UP
GUIDANCE SYSTEM

Alignment of the back-up system from the operating primary system requires transfer of appropriate angular and velocity information. This can be accomplished automatically, upon command signal from the crew, or the data can be manually entered into the back-up system from primary system displays. Alignment of the back-up system when the primary system has failed need only be accomplished on the lunar surface since it will be in continuous operation during the rest of the mission and will have been updated at intervals from the primary system until the primary system failure. Lunar surface alignment requires either multiple star sightings, or alignment to local vertical by using accelerometers or bubble-level type sensors plus a single sighting of a star or the CSM to establish the reference coordinate system. Radar sightings of the CSM may be used in place of star sightings; however, radar line-of-sight direction and line-of-sight rate from LEM to the CSM as measured by the LEM rendezvous radar requires additional knowledge of LEM and CSM relative positions and the CSM orbital rate in order to transform measured attitude angles into equivalent inertial angles.

Trade-off Studies

The preliminary trade-off study as presented to NASA, MSC Houston on May 8, 1963 is given in Tables H-1 - H-9.

As shown in Table H-1 the greater weighting factor is given to weight (35) and next reliability and development risk (25 each). Power, size and interface complexity were given 5 each.

The reliability comparison shown in Table H-2 compares relative reliabilities of the strapped-down and platform sensor packages in both redundant and non-redundant configurations. The non-redundant platform configuration has two, two-degree-of-freedom gyros while the strapped-down configuration includes three single-degree-of-freedom gyros. As shown, neither arrangement meets the reliability apportionment. When redundancy is added, in the form of one additional active, two-degree-of-freedom gyro in the case of the platform, and six additional gyros in the case of the strapped-down arrangement (of which three are in active and three in standby or passive redundancy), the strapped-down configuration meets the requirement while the platform falls slightly short. One of the major contributors to unreliability in the platform configuration is the slip ring assembly. Unfortunately, this problem is not improved in the redundant configuration. The DDA required for coordinate conversion in the strapped-down arrangement is included with the gyro package in the reliability comparison.

Table H-3 is a weight comparison which shows the redundant strapped down configuration to weigh 49.5 pounds while the redundant platform weighs 45.5 pounds mainly because the electronics of the strapped down system weighs more than the platform even though the platform sensor package weighs more than the strapped down sensors.

The indicated power comparison shown in Table H-4 indicates approximately the same power consumption for both configurations (125 watts).

Similarly Table H-5 indicates approximately the same physical size for both configurations.

The interface comparison, shown in Table H-6, indicates greater complexity for the strapped down system because of the D/A conversion requirements. In the platform the corresponding complexity lies in the four resolvers and three gear trains.

The development risk is outlined in Table H-7 and indicates areas requiring development of techniques beyond those proven in existing operational hardware.

The platform problem area is outlined in Table H-8 and indicates that an apparent singularity occurs during exact 90° rotation about the 3rd gimbal axis. Tracking attitude constraints of ±65° in yaw during flight where a further contingency requires tracking through 90° position will also result in serious platform errors.

The major problem areas in development of an operational strapped-down gyro configuration listed in Table H-9 are associated with the stringent requirements placed upon the gyro torquer and its associated electronics. Achievement of the necessary sensitivity over the desired dynamic range requires a significant advance in the state-of-the-art of gyro pulse torquing techniques.

TABLE H-1

WEIGHTING FACTORS

WEIGHTING FACTORS	VALUE
. RELIABILITY	25
. WEIGHT	35
. POWER	5
. SIZE	5
. INTERFACE & COMPLEXITY	5
. DEVELOPMENT RISK	25
. TOTAL	100

TABLE H-2

RELIABILITY COMPARISON

RELIABILITY (25)

	STRAPPED DOWN		PLATFORM	
	REDUN	NON-REDUN	REDUN	NON-REDUN
. RELIABILITY ANALYSIS	.99906	.9850	.99744	.9853
. INITIAL APPORTIONMENT RELIABILITY	.9986	.9986	.9986	.9986
. NON-REDUNANT CONFIGURATION	-	3-SINGLE DEG OF FRDM GYROS	-	2-TWO DEG OF FRDM GYROS
. CONFIGURATION BASED ON 3 AXIS REDUNANT & MAJOR VOTED LOGIC	9 GYROS (6 ACT) (3 PASS)	-	3 GYROS 3 (ACT)	-
. MAJOR RELIABILITY PROBLEM	NONE	GYROS, ACCEL, & DDA	SLIP RINGS	GYROS, ACCEL, & SLIP RINGS

TABLE 3

WEIGHT COMPARISON

WEIGHT (35)

	STRAPPED DOWN	PLAT.
. ESTIMATED TOTAL WEIGHT	49.5#	45.5#
. ESTIMATED WEIGHT INCLUDES:		
- BASIC SENSOR PKG. & ELECTRONICS	25 #	35.5#
- ATTITUDE REF & CES INFCE REQ	8 #	6 #
- RELIABILITY REQ SENSOR PKG	9 #	5 #
- ELECTRONICS	7.5#	1 #

TABLE H-4

POWER COMPARISON

POWER (5)

	STRAPPED DOWN	PLATFORM
. EST TOTAL POWER REQ'D	126 W + HTRS	125 W + HTRS
. EST TOTAL PWR REQ'D INCLUDES:		
- BASIC SENSOR & ELECTRONICS	60 W + HTRS	110 W + HTRS
- ATTITUDE REF & CES INFCE	15 W	5 W
- RELIABILITY REQUIREMENT		
. SENSORS	15 W + HTRS	5 W + HTRS
. ELECT	36 W	5 W

TABLE H-5

SIZE COMPARISON

SIZE (5)

	STRAPPED DOWN	PLATFORM
. ESTIMATED SIZE	.86 CU. FT.	.80 CU. FT.
. ESTIMATED SIZE INCLUDES:		
- BASIC SENSOR PKG & ELECTRONICS	0.55 CU. FT.	.66 CU. FT.
- ATTITUDE REF & CES INFCE REQ	0.08 CU. FT.	0.01 CU. FT.
- RELIABILITY REQ SENSORS	.08 CU. FT.	.11 CU. FT.
ELECT.	.15 CU. FT.	.02 CU. FT.

TABLE H-6

INTERFACE REQUIREMENTS AND COMPLEXITY (5)

	STRAPPED DOWN	PLATFORM
. ADDITIONAL EQUIPMENT REQUIRED FOR INTERFACE		
- DISPLAYS	MATRIX MULT & DA CONV.	NONE
- CONTROL ELECT. SECT	ELECTRONIC D/A CONV.	4 RESOLVE. 3 GEAR TRAINS
- ALIGNMENT . IMU & LUNAR	A/D CONV AND/OR D/A CONV*	NONE

* INVESTIGATIONS UNDERWAY TO DETERMINE ACTUAL HARDWARE REQUIREMENTS

TABLE H-7

DEVELOPMENT RISK COMPARISON

DEV. RISK (25)

MAJOR PROBLEM AREAS

- PLATFORM
  - APPARENT SINGULARITY
  - REDUNT GYRO PLATFORMS
- STRAPPED DOWN
  - LARGE GYRO TORQUE RATES AND DYNAMIC RANGE
  - GYRO PULSE TORQUING TECHNIQUES
  - EFFECT OF TORQUE RECT ON GYRO DRIFT
  - DDA INTERFACE

TABLE H-8

PLATFORM PROBLEM AREA

PLATFORM PROBLEM AREA

- APPARENT SINGULARITY OCCURS DURING EXACT 90° ROTATION ABOUT 3rd GIMBAL AXIS
- TRACKING THRU 90° POSITION

  REQUIRED FOR FOR CONTINGENCIES (DURING FLIGHT,

  ATTITUDE RESTRAINTS ± 65° of YAW)
- REQUIRED RELIABILITY SEEMS DIFFICULT TO OBTAIN EVEN WITH GYRO REDUNANCY

TABLE H-9

STRAPPED DOWN PROBLEM AREAS

STRAPPED DOWN PROBLEM AREAS

. LARGE GYRO TORQUE RATES AND DYNAMIC RANGE

- TORQUERS PERFORM FUNCT. OF GIMBAL DRIVES
- EFFECTS OF HEAT ON GYRO DRIFTS
- BIASING GYROS

. PULSE TORQUING TECHNIQUES

- LITTLE DEVELOPMENT WORK DONE -

(ENVIRONMENTAL EFFECTS)

- ALTERNATE METHODS

BANG-BANG

PULSE DEMAND

. TORQUE RECTIFICATION EFFECTS ON GYRO DRAFT

- VEHICLE LIMIT CYCLE AFFECTS GYRO DRIFT

. DIGITIAL DIFFERENTIAL ANALYZER

- NO OPERATIONAL SYSTEMS WITH PULSE REBAL GYROS
- SETTING INITIAL CONDITIONS

I. CREW OPERATIONAL REQUIREMENTS AND CAPABILITIES

Introduction

The LEM crew's primary function is to act as the adaptive element in the guidance and control function. The design of the man-machine-mission complex will have accommodations so that:

- the crew can update and adapt the guidance and control mechanism to current and intended mission events;
- conventional guidance and control tasks are automated;
- the crew has effective flight control and capabilites with respect to safety of flight and mission success;

The discussion in this section is concerned with crew operational requirements in terms of the interaction of crew tasks and capabilites with mission requirements and with the results of preliminary mission simulation studies.

Summary of Crew Tasks

The type and number of tasks to be performed by the crew in the primary or back-up guidance modes will depend upon the degree to which sensing, guidance, and control operations are automated.

There are basically three classes of variables that will influence crew performance. They are not mutually exclusive and the variables in each will probably be found to interact with those in the other classes. The first class is that associated with the nature of the trajectories and their parameters. The second class involves the nature of the vehicle flight control system and includes handling and flying qualities as well as attitude constraints. The third class includes display and optical/visual variables.

In the fully automatic navigation, guidance, and control modes, crew tasks will consist of:

- monitoring flight director and attitude displays;
- monitoring flight path displays;
- visually verifying altitudes, attitudes, ranges, rates, topographical features, and reference alignments by observations of the external environment through optical aides and direct viewing through the window;
- comparing actual with nominal flight plan parameters and applying corrections as required;

. assessing vehicle and subsystem status, verifying or restoring normal operating conditions;

. deciding on, and performing alternative actions in the event of disabling malfunctions.

Table D-1* summarizes the interaction of crew tasks with mission requirements, while Table I-1 summarizes the crew tasks interaction with the controls and displays in both the primary and back-up guidance modes.

In the manual alternate modes, the commander directs and executes flight path control in accordance with instrument and visual/optical flight plan procedures. The systems engineer will assume copilot tasks in addition to his primary tasks of monitoring and maintaining subsystem operational status.

At present, the mission flight plan is defined only in terms of a fully automatic guidance and control mode and a semi-automatic emergency or abort mode. Selection of the criteria for making decisions regarding the division of duties between man and mechanism in the degraded back-up modes is still in progress. These criteria will be incorporated in subsequent mission simulator assessments of the feasibility of manual flight regimes.

Results of Prior Mission Simulation Studies

1. Rendezvous, Docking and Letdown

There has been considerable investigation into the capabilities of the human pilot to guide and control during the hover and letdown, docking, and rendezvous phases. The activity of the pilot becomes valuable in these phases because of his ability to select landing areas, identify objects during space intercept, choose tactics to be used, and make decisions if there are unforseen difficulties. The results of these human factors and simulation studies indicate that the pilot can manually control the vehicle in these phases within acceptable limits given the appropriate visual and instrumented flight path cues.

Brissenden (see Reference 1)* has conducted extensive human factors and simulation studies in connection with manual control of space rendezvous and docking and has been concerned with the development and simulator evaluation of an instrument flight plan approach; he concludes that:

. human pilot has the control capability to effect rendezvous successfully in the presence of relatively severe conditions if adequate vehicle control and flight-data presentation are provided;

* Tables and references in this section will be found on pages I-8 to I-30.

. single retro-rocket with multistart capability is sufficient for vehicle control if attitude controls and display information on line-of-sight rates and range and range rate are furnished. Continuously variable rocket thrust is not necessary. Pilots prefer to use intermittent thrust of constant value. Thrust misalignments up to 90 per cent of attitude-control can be handled;

. the rendezvous vehicle need not be co-planar with the satellite station prior to rendezvous, and initial conditions, within a wide band of control capabilities of the vehicle, do not adversely affect rendezvous;

. in the absence of visual aids, the instrument presentations deemed necessary by the pilots are:

  . range and range rate;

  . elevation and azimuth line-of-sight rates;

  . vehicle attitude angles and angle rates;

  . elevation and azimuth angles.

. experienced pilots, used to dial instruments, tend to follow similar space trajectories in making a rendezvous, but times to rendezvous vary somewhat, and an energy management schedule for the pilot in table form or additional display would be required for time control;

. the average amount of fuel used by the pilots is only slightly higher than the reference minimum. Perturbing effects, such as thrust-misalignment torques and on-off reaction controls, do not necessarily cause an increase in fuel consumption. Fuel use does vary moderately with specific rendezvous techniques controlling the time required;

. some artificial damping of the angular motions of the vehicle is found to be desirable but not essential;

. there is no definite correlation between pilot opinions and attitude-control input frequencies for a wide band of tolerable control characteristics and data displays.

The development and simulator evaluation of visual rendezvous techniques is located in Reference 2. The results indicate that a pilot using a simple optical sighting device and a clock can determine the parameters necessary for computing the relative range and closure rate between his vehicle and another space vehicle while arresting the angular motion between the two vehicles, and can do so with sufficient accuracy to perform the final braking maneuver safely to a point where the rendezvous can be completed from direct visual cues.

The study described in Reference 2 is continued in Reference 3 and concludes that the line-of-sight rate must be measurable to within 0.1 mr/sec in order to control a completely visual rendezvous, including the computation of range and range rate from angular measurements. It recommends that the grid resolution of the optical sight be better than 1 milliradian.

The human factors considerations affecting visual capabilities in rendezvous are discussed in References 4 and 5. The results of the study of Reference 4 are as follows:

- if the angular separation between a space target and an inertial reference is 12.5 milliradians, a pilot can detect an angular rate of 0.1 milliradian per second by observing a 1 milliradian angle traversed in 10 seconds or less;

- high brightness difference between moving objects and their background is not required for good angular detection. Visual detection ability deteriorates with fatigue, and visual tracking tasks should include relief cycles;

- the error in identifying object motion varies directly with reference separation. To maximize a pilot's visual ability, an optical device that projects a space-fixed reference onto the target grid should be provided. If this optical device has at least a 3 inch lens, the pilot can use stars as dim as eleventh magnitude as background inertial reference and thereby obtain a reference star within 12 milliradians of the target.

Target parameters which affect visual detection in the acquisition phase of rendezvous are discussed in Reference 5. An important conclusion reached is that the use of a flashing white light on the target will facilitate the detection of the target moving slowly in a star field. The flash rate must be slow enough to permit a flash duration not requiring excessive power, but still fast enough so that there will be several flashes while the pilot searches the target area.

Descent and Soft Lunar Landing

Unlike the rendezvous, docking, and letdown phases, there are no visual techniques for effecting powered descent. Sensing and tracking of the major descent trajectory parameters can only be done with the inertial and radar sensors. The degree of difficulty of the tracking task depends upon the trajectory characteristics, the guidance laws for effecting optimal and near-optimal trajectories, and the fuel consumption budget.

The powered descent simulation studies conducted to date have emphasized minimum fuel descent trajectories to the point where the requisite guidance and control could only be effected under fully automatic control. The resultant trajectory guidance laws impose such a premium on information-handling rates, execution timing and accuracy as to render a man incapable of taking over in the event of machine malfunctions; the pilot might, at best, backup the attitude and translation servo amplifier.

LEM Engineering Simulation Program

The LEM Engineering Simulation Program is intended to evolve much of the LEM design philosophy and technology. The scope, objectives, techniques and scheduling of this comprehensive program are presented in References 9 and 10.

Peske and Swanlund (see Reference 6) have formulated a descent trajectory and guidance law, and configured an automatic guidance and control system, to obtain an optimum (least fuel consumption) response to an initial lateral displacement error from the reference trajectory. The vehicle is assumed to be descending from an altitude of 90 miles at a constant two earth gravity deceleration using thrust vector control and a linear controller, and approaching an established lunar base (navigation aids on surface). The vehicle can measure position, velocity, attitude, and attitude rate, on-board and/or via lunar base data links. The optimum automatic system response is used as the performance standard of comparison between automatic and manual flight path control.

From the results of the simulator evaluations the authors conclude that:

- the pilot could perform outer loop control of the vehicle with only laterial position information, or lateral position and velocity information. (The inner stability augmentation loop is automated) However, an optimum trajectory is not achieved;

- optimum trajectories could be obtained with manual control only if the same feedback information as used by the automatic system (with feedback terms in the right proportion) are displayed on the error indicators;

- if the operator is doing only a simple error-tracking task and is thus acting in the capacity of a servo amplifier, he does not make the best use of his capabilities, his function in the control of the vehicle should be to monitor the automatic system, only taking over in the event of a system failure. Most of his attention should be directed toward locating and tracking the landing site.

Queijo has conducted analytical and pilot simulation studies in regard to circumlunar and lunar descent trajectories, lunar descent thrusting techniques, and the implementation of control displays which enable manual control from injection into lunar orbit through soft landings on the lunar surface.

An analytical six degree of freedom fixed-base simulator study of the ability of pilots to modify ballistic trajectories of a space vehicle approaching the moon and establish a circular orbit about 50 miles above the lunar surface is described in Reference 7. The unmodified ballistic trajectories have miss distances from the lunar surface of from 40 to 80 miles and a velocity change of from 8,200 to 8,700 feet per second at closest approach.

The pilot is given control of the thrust (along the vehicle longitudinal axis) and torques about all three body axes. The information display given to the pilot is a hodograph of the vehicle rate of descent and circumferential velocity, an altimeter, and vehicle attitude and rate meters. The results of the investigation have shown that pilots soon become adept at flying the simulator and can consistently establish orbits lying within an altitude from 10 to 90 miles. The indicated fuel consumption generally is about 1 to 3 percent of the initial vehicle mass more than that required by use of a two-impulse Hohmann maneuver. The use of the hodograph as a primary display is very effective and can provide much useful information.

In Reference 8, a six degree of freedom fixed-base simulator study of the ability of pilots to execute soft lunar landings from circular lunar orbits at 50 miles altitude is discussed. The pilot is given control over the vehicle thrust level and moments about each of the three body axes. With the assumption that the vehicle has no automatic damping control, the pilot's task is to acquire and fly particular trajectory hodographs and to perform a soft landing in a specified area. The task has been performed either by one pilot in complete control of the vehicle or by two pilots with one controlling vehicle attitude and the other controlling the thrust level. The results of this study have led to the following conclusions:

1. The pilots can consistently make good landings, and have generally landed with touch-down-velocity components (radial and tangential) of less than 4 feet per second and within a range of about 2,000 feet of the desired landing site. This range variation is associated primarily with the readout resolution of the indicators shown to the pilot.

2. There is no appreciable difference in touch down conditions or fuel consumption between the one-pilot and the two-pilot flights. However, one-pilot lunar landings require close concentration on the part of the pilot in order to attain acceptable touchdown velocity components, vehicle attitude, and the desired landing site. The landing task is made much easier with two pilots, one operating the throttle and the other the attitude control.

3. It is possible effectively to uncouple the range and altitude control by using throttle for range control and the attitude control to adjust altitude over most of the landing trajectory. During the vertical descent phase, the throttle is used to control altitude and rate of descent, and the attitude control to adjust range and circumferential velocity.

4. The characteristic velocity for lunar landings varies from 0 to 10 percent above the velocity computed for a perfectly flown trajectory.

5. A throttle ratio of from 7:1 to 10:1 is used to make lunar landings. It is estimated that for the maximum thrust-to-weight ratio available in this study, a throttle ratio of about 5:1 will be required for a perfectly flown trajectory.

Crew Station Controls and Displays Configuration

A preliminary Crew Station Controls and Displays Configuration is detailed and described in the following drawings and document.

LSK-480-10051 Sheet 1 of 3	Cockpit 66 - Tunnel M-1 Instrument Panel
LSK-480-10051 Sheet 2 of 3	Cockpit 66 - Tunnel M-1 Lower Instrument Panel
LSK-480-10051 Sheet 3 of 3	Cockpit 66 - Tunnel M-1 Side Instrument Panel
LED-480-2	Preliminary Subsystem Controls and Displays

References

(Section I)

1. Brissenden, Roy F., Burton, Bert B., Foudriat, Edwin C., Whitten, James B., "Analog Simulation of A Pilot-Controlled Rendezvous", NASA TN D-747, April 1961.

2. Lineberry, Edgar C., Jr., Brissenden, Roy F., Kurbjun, Max. C., "Analytical and Preliminary Simulation Study of a Pilot's Ability to Control the Terminal Phase of a Rendezvous With Simple-Optical Devices and a Timer", NASA TN D-965, October 1961.

3. Brissenden, Roy F., and Lineberry, Edgar C. Jr., "Visual Control of Rendezvous", Aerospace Engineering, 21, June 1962.

4. Brissenden, Roy F., "A Study of Human Pilot's Ability to Detect Control of Space Rendezvous", NASA TN D-1498, December 1962.

5. Pennington, Jack E. and Brissenden, Roy F., "Visual Capability In Rendezvous", Astronautics and Aerospace Engineering, 96-99, February 1963.

6. Peske, A. and Swanlund, G., "Control Considerations for a Lunar Soft Landing", Aerospace Engineering, 20, 26-27, 72-74, 1961.

7. Queijo, M. J. and Riley, Donald R., "A Fixed-Base Simulator Study of the Ability of a Pilot to Establish Close Orbits Around the Moon", NASA TN D-917.

8. Queijo, M. J., Miller, G. Kimball, Jr., and Fletcher, Herman S., "Fixed-Base Simulator Study of the Ability of a Pilot to Perform Soft Lunar Landings", NASA TN D-1484.

9. Kress, R. W., "LEM Engineering Simulation Program Review at MSC, May 10, 1963", Grumman Aircraft Engineering Corporation LEM Engineering Memorandum LMO-570-57, May 15, 1963.

10. Greene, S. and Beadle, R. Jr., "Problem Definition for Grumman Aircraft Engineering Corporation's Real Time Abort Studies to be Conducted at Ling-Temco-Vought's Manned Aerospace Flight Simulator", LMO-570-52, 3 May 1963.

11. Beadle, R., Munter, P., Wolf, H. and Wood, F., "Detailed Presimulation Report for Phase A Lunar Hover and Landing Simulation", LED 570-1, 4 March 1963.

12. Freeberg, N. E. and Wolf, H., "Lunar Landing Simulation Phase A, "Preliminary Report No. 1", LED 480-2, 17 May 1963.

TABLE I-1

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
1. Syn-Coast		
Separate	a) Execute sep. using VFR, IFR, AFR techniques.	a) Monitor situ. board and advise.
	b) Orient vehicle for LEM CSM mutual radar and VHF checks as req.	b) Monitor control displays and advise.
	c) Trim vehicle attitude for injection.	c) Monitor LEM-CSM mutual checks and advise
	d) Execute docking if status board abort situ. verify.	d) SW to BUGS if PNGS cause for abort.
	(1)*; (3) -a; (4) -a, b; (5) -b; (6) -b; (8) -a, b; (9) -a; (10) -a, b, c, d; (11) -a, b, c, e; (12) -a, b, c; (14) -a; (15) -a; (18) -a; (19) -a, b, d; (20) -a.	(8) -a, b; (9) -a; (10) -a, b, d; (12) -a, b, c; (14) -a: (15) -a; (16) -a; (17) -a, e.
Inject	a) Initiate inject command to PNGS.	a) Monitor situation board and advise.
	b) Monitor control and situation displays WRT nominal flight plan.	b) Monitor radar and communication contacts.
	c) Execute manual override either for vernier correct and/or abort. Follow abort procedures.	c) SW to BUGS if PNGS cause for abort.

* See Code

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
1. Syn-Coast Separate	(2); (3) -b; (4) -a; (5) -b; (6) -b; (7) -b; (8) -a, b; (9) -a, b; (10) -a, b, c, d; (11) -b, c, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a, b; (20) -a	(8) -a, b; (9) -a, b; (10) -a, b, d; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e, g
1. Syn-Coast Coast	a) Monitor control and situation displays w.r.t. nominal flight plan.	a) Monitor situation board and advise.
	b) Execute manual override as required for reorientation accord. flight plan attitude constraints and star sight.	b) Monitor control displays and advise as req.
	c) Update PNGS via AOT star sightings.	c) Transfer aline BUGS
	d) Revert to manual FCS mode and BUGS in event of PNGS abort situation.	d) Monitor radar alt. near Pericynthion.
	e) In abort situation, determine abort maneuver. (direct, coast to rendezvous)	e) SW to BUGS in event of PNGS abort situation. Assist comander in deciding abort maneuver requirement.
	(2) ; (3) -b; (4) -a, b; (5) -b; (6) -b; (8) -a, b; (9) -b; (10) -a, b, c, d, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a, d; (20) -a.	(8) -a, b; (9) -b; (10) -a, b, d, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e, g.

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
Pow-Descent		
2a) Nominal Inject - 20 K Ft.	a) Monitor control displays w.r.t. nominal flight plan.	a) Monitor control and situa. displays.
	b) Monitor abort situat. display and anticipate continue or abort decision and procedures.	b) Monitor and correlate radar alt. w PNGS target data.
	(2) ; (3) -b; (4) -a, b; (5) -b; (6) -b; (8) -a, b; (9) -c; (10) -a, b, c, d, e, f, g, h, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a, d; (20) -a.	(8) -a, b; (9) -c; (10) -a, b, d, e, f, g, h, i, j, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e, g.
2b. Abort from Pow-descent	. Commun. with CSM - verify pres. position w.r.t. phasing planarity.	. Locate and verify "killer"
Inject - 20 K	. If killer FCN confirmed, execute descent engine staging per flight program	. Follow emerg. procedure. Life Support Fuel Management Electrical Power Supply Enter plane change parameter into PNGS-AGC
	. Actuate abort command to AGC.	. Monitor PNGS Abort Nav.-Guid. function.
	. Monitor ascent and rendezvous displays.	. Monitor situation displays.
	. Maintain VHF contact with CSM.	

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
2b. Abort from Pow-descent	(2) ; (3) -b; (4) -a; (5) -b; (6) -a, b; (7) b; (8) -a, b; (9) -c, d; (10) -a, b, c, d, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -b; (15) -b; (19) -a, d; (20) -a.	(8) -a, b; (9) -c, d; (10 -a, b, d, i, j, k; (12) -a, b, c; (14) -b; (15) -b; (16) -a; (17) -a, b, d, e;
2c. Abort From Pow-Descent Inject - 20 K	. Comm. with CSM - verify pres. position w.r.t. phasing - plane change for prospective ascent trajectory.	. Correlate LEM orbit params transmitted from CSM and reset BUGS ascent programmer.
	. SW FCS guidance coupler to BUGS	
	. Follow abort proced. w.r.t. descent engine staging and re-orientation of vehicle in pitch and roll.	. Assist in abort procedures.
	. Activate BUGS abort programmer.	. Monitor BUGS ascent guidance operation.
	. Monitor ascent displays.	. Monitor situation displays.
	. Standby for midcourse and rendezvous phases.	. Standby for midcourse and rendezvous phases.
	(2) ; (3) -a, b; (4) -a; (5) -b; (6) -b, a; (7) -b; (8) -a, b; (9) -c, d; (10) -a, b, c, d, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -b; (15) -b; (19) -a, d; (20) -a.	(8) -a, b; (9) -c, d; (10) -a, b, d, i, j; (12) -a, b, c; (14) -b; (15) -b; (16) -b; (17) -a, b, d, e.

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
2d. Abort from Pow-Descent Inject - 20 K	. Comm. with CSM for verification of phasing - plane change instructions.	. Monitor status board.
	. SW FCS guidance coupler to S - Attitude Hold Mode.	. Standby for manual control of ascent countdown.
	. Follow flight plan abort procedure w.r.t. descent engine staging and reorientation of vehicle in pitch and roll.	. Start countdown on Pitch-thrust ascent manual program.
	. Activate and hold ascent engine thrust on cue.	. Cue commander to thrust vertical.
	. Execute pitch prog. on eng-nav. cues	. Cue commander on execution of pitch angle prog. (set in attitude and attitude rate to control displays).
	. Monitor and null attitude and attitude rate error displays via att. controller.	
	. Cut off thrust on cue.	. Cue commander on thrust cut off.
	. Terminate pitch prog. on cue.	. Cue commander on pitch prog. term.
	. Standby for mid-course correct. and rendezvous.	. Standby for midcourse correction and rendezvous.

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
2d. Abort from Pow-Descent	(1) ; (3) -a; (4) -a; (5) -a; (6) -a, b; (7) -b; (8) -a, b; (9) -c, d; (10) -a, b, c, d, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -b; (15) -b; (19) -a, d; (20) -a.	(8) -a, b; (9) -c, d; (10) -a, b, d, i, j, k; (12) -a, b, c; (14) -b; (15) -b; (16) -b; (17) -a, b, e.
3a. Pow-Descent 20 K - Hover	. Monitor control displays w.r.t. nom. flight plan. . Auto steering should roll vehicle 180° to put landing sight within view of window and opt. aid. . Begin surveillance of prospective landing site. Maintain commun. with CSM. . If new land site elected enter coord. data to AGC. . Standby to take over manual control at hover. (2) ; (3) -b; (4) -a, b; (5) -b; (6) -b; (8) -a, b; (9) -c; (10) -a, b, c, d, e, f, g, h, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (18) -b, c; (19) -a; (20) -a.	. Monitor situation displays . Monitor radar track of CSM. . Standby for possible co-pilot duty at hover. (8) -a, b; (9) -c, (10) -a, b, d, e, f, g, h, i, j, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e, g; (18) -d.

CREW TASK INTERATION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
3b. Abort from Pow-Descent 20 K - Hover	Similar to 2b.	Similar to 2b.
3c. Abort from Pow-Descent 20 K - Hover	Similar to 2c.	Similar to 2c.
3d. Manual Pow-Descent 20 K - Hover	∘ Commun. with CSM. ∘ SW FCS to A.H. Mode ∘ Manually control descent to hover per instrument flight rules ∘ Piloting will consist of nulling errors on command displays via attitude and thrust controller. ∘ Execute visual-opt. surveillance of lunar identification points. (1) ; (3) -a; (4) -a; (5) -a; (6) -b; (7) -a; (8) -a, b; (9) -c; (10) -a, b, c, d, e, f, g, h, i, j, k; (11) -b, c, d, e; (12) -a, b, c; (14) -b; (18) -a, b, c; (19) -a, d; (20) -a.	∘ SW BUGS to control dis. ∘ Use nom. descent - hover hodographs. ∘ Cue Commander on hodograph sequence of parameters per manual pow-descent schedule (Discrete Settings of parameters on command displays). ∘ Monitor situa. displays ∘ Assist in visual surveillance. (8) -a, b; (9) -c; (10) -a, b, d, e, f, g, h, i, j, k; (12) -a, b, c; (14) -b; (16) -b; (17) -a, e; (18) -d.

CREW TASK INTERACTION WITH CONTROLS AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
4a. Nominal Hover - Landing	. Execute manually controlled let-down to chosen landing site - or translation to new site and let-down via thrust and attitude and translation controllers.	. Monitor situation displays.
		. Assist in visual surveillance.
		. Execute post land. Checkout of subsystems.
	(1) ; (3) -a; (4) -a, b; (5) -a; (6) -b; (7) -a; (8) -a, b; (9) -c; (10) -a, b, c, d, e, f, g, h, i, j, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (18) -a, b, c; (19) -a, d; (20) -a.	(8) -a, b; (9) -c; (10) -a, b, d, e, f, g, h, i, j, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e; (18) -d.
4b. Abort From Hover	Similar to 2b, except for point of abort.	Similar to 2b, except for point of abort.
	Similar to 2c, except for point of abort.	Similar to 2c, except for point of abort.
5a. Nominal Pre-ascent from Lunar Surface	. Establish local vertical via star sightings.	. Rough erect to vertical using accel. level loops.
	. Receive direct communication from CSM as it comes over horizon.	. Fine erect using optically defined vert.
	. Establish LOS to CSM.	. Verify Subsystem on and operating.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
5a. Nominal Pre-Ascent From Lunar Surface	. Track CSM with rendezvous radar and establish CSM orbit. . Execute launch on cue from engineer navigator. (2) ; (3) -b; (4) -a, b; (5) -b; (6) -a; (8) -a, b; (10) -a, b, c, d, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a; (20) -a.	. Enter CSM track data to AGC. . Begin launch countdown - (launch when CSM overhead). (8) -a, b; (10) -a, b, d, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, c, d, e.
5b. Backup Guid. - Pre-Ascent From Lunar Surface	Similar to 5a.	Similar to (5a) except that computation of LEM-CSM phasing, plane-change and ascent trajectory data to be entered into BUGS programmer either done with nomagraphs or with aid of CSM commun. link.
6a. Nominal Pow-Ascent	. Commun. with CSM. . Monitor control displays and correlate ascent trajectory with flight program nominal. . Standby for manual ascent control in event of PNGS malfunction.	. Monitor Situa. displays. . Standby for switch-over to BUGS AR in event of PNGS malfunction.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
6a. Nominal Pow-Ascent	(2) ; (3) -b; (4) -a, b; (5) -b; (6) -a; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a; (20) -a.	(8) -a, b; (9) -d; (10) -a, b, d, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, c, e.
6c. BUGS Pow-Ascent	Similar to 2c except that data entry to BUGS program. are w.r.t. lunar surface datum.	Similar to 2c except that data entry to BUGS are w.r.t. lunar surface datum.
6d. Manual Pow-Ascent	Similar to 2d except ascent pitch-thrust program is w.r.t. lunar surface datum.	Similar to 2d except ascent pitch-thrust program is w.r.t. lunar surface datum.
7. Coasting - Ascent Midcourse Correction		
7a. Nominal	a) Monitor control displays and correlate with flight plan. Note length of ascent coasting orbit and times to midcourse events (t_1, t_2, τ_1), (t_3, t_4, τ_2). b) At time for first measurement (t_1), read radar range, pitch angle, and antenna elevation angle. Compare these readings with flight plan and note differential quantities.	a) Monitor situation displays.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
7a. Nominal	c) At time for second measurement (t_2), read radar range, pitch angle, and antenna elevation angle. Compare these with flight plan and note differential quantities. d) Enter the differential range and inplane pitch angle quantities noted at t_1 and t_2 on the slide rule scales and compute the first midcourse corrections in ΔV_x and ΔV_z. e) Compare slide-rule solution with AGC displayed solution. f) Note time of execution of first midcourse correction and compare with nominal time (τ_1) for first midcourse correction. g) At time for third measurement (t_3), read radar range, pitch angle, and antenna azimuth angle. Compare these readings with flight plan and note differential quantities.	

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
7a. Nominal	h) At time for fourth measurement (t_4), read radar range, pitch angle, and antenna azimuth angle. Compare these readings with flight plan and note differential quantities. i) Enter the differential range and out-of-plane angle noted at t_3 and t_4 on slide rule scales and compute the second midcourse correction in ΔV_y. j) Compare the slide-rule solution with AGC displayed solution. k) Note time of execution of second midcourse correction and compare with nominal time (T_2) for second midcourse correction. (2) ; (3) -b; (4) -a, b; (5) -b; (6) -a; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, c, d, e; (12) -a, b, c; (14) -a; (19) -a; (20) -a.	(8) -a, b; (9) -d; (10) -a, b, d, k; (12) -a, b, c; (14) -a; (16) -a; (17) -a, c, e.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
7c. BUGS	a) Monitor control displays and correlate with flight plan. Note event time (t_1, t_2, τ_1), (t_3, t_4, τ_2) determined by engineer-navigator and standby for cues.	a) Monitor situation displays and determine length of ascent coasting orbit and time to midcourse events (t_1, t_2, τ_1), (t_3, t_4, τ_2) and cue commander at each event.
	b) At t_1 cue, note radar range, pitch angle, and antenna elevation angle. Compare with flight plan and note differential quantities.	b) Cue commander at t_1.
	c) At t_2 cue, note radar range, pitch angle, and antenna elevation angle. Compare with flight plan and note differential quantities.	c) Cue commander at t_2.
	d) Enter differential range and in-plane pitch angle quantities noted at t_1 and t_2 on the slide rule scales and compute the first corrections in ΔV_x and ΔV_z. Set computed ΔV_x and ΔV_z into RCS controller.	
	e) At τ_1 cue, fire RCS ΔV_x and ΔV_z jets.	e) Cue commander at τ_1.
	f) At t_3 cue, note radar range, pitch angle, and antenna azimuth angle. Compare with flight plan and note differential quantities.	f) Cue commander at t_3

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
7c. BUGS	g) At t_4 cue, note radar range, pitch angle, and antenna azimuth angle. Compare with flight plan and note differential quantities.	g) Cue commander at t_4.
	h) Enter differential range and out-of-plane angle quantities noted at t_3 and t_4 on the slide rule scales and compute the second midcourse correction in ΔV_y. Set computed V_y into RCS controller.	
	i) At τ_2 cue, fire RCS ΔV_y jets.	i) Cue commander at τ_2.
	j) Standby for rendezvous phase.	
	(1) ; (3) -a; (4) -a; (5) -a; (6) -a; (7) -a; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, c, d, e; (12) -a, b, c; (14) -b; (19) -a; (20) -a.	(8) -a, b; (9) -d; (10) -a, b, d, k; (12) -a, b, c; (14) -b; (16) -b; (17) -a, e.
8. Rendezvous		
8a. Nominal	. Maintain commun. with CSM.	. Monitor situation displays.
	. Monitor control displays - use AOT to visually monitor rendezvous.	
	. Standby for manual docking.	

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
8a. Nominal	(2) ; (3) -b; (4) -a; (5) -b; (6) -a; (7) -b; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, c, d, e; (12) -a, b, c; (14) -a; (18) -a, b, c; (19) -a; (20) -a.	(8) -a, b; (9) -d; (10) -a, b, d, k; (12) -a, b, c; (14) -a; (16) -a; (17) -a, c, e, f; (18) -d.
8b. Manual Alternate (IFR)	. Monitor rendezvous radar display and note lock-on signal and range marker. . Orient vehicle in pitch and roll until rendezvous radar gimbal angles nulled and CSM visible through forward cabin window. This effects alinement of radar gimbal axes with vehicle X and Y axes. . At first rendezvous range marker, yaw until radar gimbal inertial rate is detected and displayed on either radar azimuth or radar elevation display. . Use translation controller to thrust along vehicle axis corresponding to radar gimbal axis displaying rate. Thrust until radar gimbal intertial rate is nulled. . Use thrust controller to reduce range rate to predetermine value.	. Monitor situation display.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
8b. Manual Alternate (IFR)	. At 2nd rendezvous range marker repeat LOS nulling and range rate braking. . Continue procedure at subject range marker to achieve acceptable terminal rendezvous range and closure rates. . Standby for manual docking. (1) ; (3) -a; (4) -a; (5) -a; (6) -a; (7) -a; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, d, e; (12) -a, b, c; (14) -a; (15) -a; (19) -a; (20) -a.	(8) -a, b; (9) -d, (10) -a, b, d, k; (12) -a, b, c; (14) -a; (15) -a; (16) -a; (17) -a, e.
8c. Manual Alternate (IFR)	Similar to 8b except range markers not displayed. Range marking cued by systems engine. (1) ; (3) -a; (4) -a; (5) -a; (6) -a; (7) -a; (8) -a, b; (9) -d; (10) -a, b, c, d, k; (11) -b, d, e; (12) -a, b, c; (14) -b; (19) -a; (20) -a.	Similar to 8b except range markers not displayed. Cue commander on range marks. (8) -a, b; (9) -d; (10) -a, b, d, k; (12) -a, b, c; (14) -b; (16) -b; (17) -a, e.
8d. Manual Alt.-Degrad. VFR - IFR	. Commun. with CSM for rendezvous instructions.	. Monitor situ. display.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
8d. Manual Alt.-Degrad. VFR-IFR	. Reorient vehicle attitude till CSM flashing lights seen through window.	. Visually assist commander in detection of CSM flashing lights.
	. Align CSM on x-y intercept of reticle by yawing and/or pitching vehicle.	
	. Reorient vehicle attitude in pitch or yaw to achieve vertical alignment of star background motion in reticle.	
	. Select star closest to CSM in reticle as reference star for determinging LOS rate of change.	
	. Determine CSM LOS rate w.r.t. referenced star by timing motion of CSM w.r.t. referenced star on reticle.	. Start timing on cue from commander.
	. Compute required time or RCS impulse count to null LOS.	. Check rendezvous flight plan for range markers vs braking.
	. Hold translational control for required time of count.	
	. Repeat until optically measured LOS rate $< 0.1 \frac{mr}{sec}$.	. Monitor radar range, range rate or obtain these data via commun. with CSM.

CREW TASK INTERACTION WITH CONTROL AND DISPLAYS (cont.)

MISSION PHASE	TASK DESCRIPTION	
	COMMANDER	SYSTEMS ENGINEER - NAVIGATOR
	. Execute thrust braking for duration cued by engineer-navigator.	. Cue commander on range rate braking duration.
	. Repeat LOS nulling and range rate braking on cue from engineer-navigator.	. Repeat cueing of commander on LOS nulling and thrust braking per flight plan.
	(1) ; (3) -a; (5) -a; (6) -a; (7) -a; (8) -a, b; (9) -d; (10) -a, b; (11) -b, d, e; (12) -a, b, c; (14) -b; (18) b, c; (19) -a; (20) -a.	(8) -a, b; (9) -d; (10) -a, b; (12) -a, b, c; (14) -b; (16) -b; (17) -a, e; (18) -d.

CODE FOR TABLE I-I

FCS UTILIZATION

MODE

1. Manual - M

2. Automatic - A

CONTROLLER FUNCTION

3. Attitude

   a. Attitude Hold - AH

   b. Automatic

4. Translation

   a. Minimum Impulse - $\delta_{min.}$

   b. Direct

5. Thrust

   a. Manual - M

   b. Automatic

6. Engine Mode

   a. Ascent

   b. Descent

7. Engine Control

   a. Manual

   b. Automatic

CONTROL DISPLAYS

8. Attitude

   a. Vehicle Attitude w/s Landing Site Vertical - VOA

   b. Vehicle Attitude w/s Inertial Ref. Coord. - VAR

9. Flight Path Parameters
   a. Circumlunar Orbital Parameters
   b. Coasting (Synch. Descent) Orbital Parameters
   c. Descent Trajectory Parameters
   d. Ascent Trajectory Parameters
10. Target Parameters
   a. Range to CSM
   b. Range Rate to CSM
   c. Line-of-Sight to CSM - Rendez. Radar Gimbal Angles
   d. Line-of-Sight Rate to CSM - Rendez. Radar Gimbal Inertial Rate
   e. Cross Range to Landing Site
   f. Down Range to Landing Site
   g. Cross Range Velocity
   h. Down Range Velocity
   i. Altitude to Lunar Surface Datum - h
   j. Altitude Rate Lunar Surface Datum - $\dot{h}$
   k. Central Angle Range to CSM - Phase $\measuredangle$
11. Fuel Management
   a. Count of δ_{min} Increments - $\Delta V_x, \Delta V_y, \Delta V_z$
   b. Totalized ΔV - ΔV_I
   c. ΔV Set (ΔV Required) - ΔV_R
   d. Program ΔV Profiles, (ΔV-h), (ΔV-θ)
   e. ΔV Remaining
12. Clock
   a. GMT Time
   b. Time From Injection into Synch. Orbit - T_i
   c. Time of Initiation of Powered Descent

SITUATION DISPLAY

14. Subsystem Status

    a. Go

    b. No Go

15. Abort Situation

    a. Off

    b. Abort

NAVIGATION - GUIDANCE UTILIZATION

16. Mode

    a. Primary

    b. Backup

17. Function

    a. Attitude Reference

    b. Abort Programmed Steering

    c. Ascent Programmed Steering

    d. Thrust Vector Control

    e. Flight Path Reference

    f. Rendezvous Terminal Guidance Schedule

    g. Descent Programmed Steering

LOS SENSOR UTILIZATION

18. Visual/Optical

    a. Window-aided Stadiametric Measurements

    b. Alignment Optical Telescope/or SCT (whichever is available)

    c. Window - Reticle

    d. 2 x 70 Binoculars

19. Radar

    a. Rendezvous Radar Track CSM

    b. Rendezvous Radar Track Beacon

    c. Rendezvous Radar Backup Altimeter

    d. Altimeter Radar

20. Communications

    a. VHF