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CAPABILITIES OF THE ENTRY GUIDANCE

EQUATIONS FOR MISSION AS-202

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ABSTRACT

This report has been prepared as part of Bellcomm participation in the activities of the Guidance Software Control Panel of the Manned Spacecraft Center. It examines the entry guidance equations used for Mission AS-202 from two viewpoints. One is the determination of the variations in conditions at the entry point over which the guidance can cause a miss of 20 NM or less at the target landing point. The other is to analyze why the trajectory is as it is and determine whether or not the equations seem suitable for use in other Earth orbital missions as well as for AS-202. Since another version of entry guidance has been approved by MSC for lunar mission entry speeds, the utility of the present guidance equations was examined only for Earth satellite speeds.

The conclusions and recommendations resulting from the study are:

1. The guidance equations perform well. For mission AS-202, the critical parameter will be the flight path angle at entry. This angle is nominally -3.57° : at $> -2.9^\circ$ the CM will skip out, at $\leq -3.87^\circ$ it will fall short of the target. The guidance can cope with wide variations in position, velocity magnitude and azimuth at the entry point, with relatively large errors in ground update data on CM position and velocity, with L/D ratios as low as 0.25 and greater than 0.55, with large IMU errors (given the ground update), with 10% variations in air density, with 100 knot winds from any direction, with gravity variations, with 75% reductions in CM roll acceleration and 75% reduction in maximum roll rate, with $> 5^\circ$ errors in trim angle of attack, with $< 30^\circ$ roll errors in lift vector position, and with $> 10\%$ variations in spacecraft weight.
2. Accuracy is good despite errors in the equations used to predict the trajectory in its early phases. These early errors tend to be mutually compensatory; the final phase guidance equations are the chief reason for accuracy.
3. The equations are possibly more complex than necessary, but the performance is good, and the equations have already been coded and tested for use in the flight computer. Therefore, unless there is urgent need for computer storage space, it is recommended that these equations be used for all missions with entry speeds at approximately Earth satellite speed and that the development effort be considered complete.

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CAPABILITIES OF THE ENTRY GUIDANCE

EQUATIONS FOR MISSION AS-202

1.0 Introduction

This report has been prepared as part of Bellcomm participation in the activities of the Guidance Software Control Panel of the Manned Spacecraft Center. The report examines the guidance equations used in the entry phase of Mission AS-202 from two viewpoints. One is the determination of the range of conditions at the initial entry point (when the spacecraft descends through 400,000 feet altitude) which the equations can cope with successfully enough to give a terminal miss of 20 NM or less.* The other is to analyze, albeit briefly, why the entry trajectory is what it is and to indicate areas where improvements may be possible.

Put another way, the first part of this report is concerned with entry guidance as a black box, while the second part considers what is inside the box.

The report is not concerned with dispersions or with the CEP at the landing point. This topic is covered in the "Guidance and Navigation System Operations Plan - Mission AS-202" (GSOP) issued by MIT [1] and by MSC reports [2]. However, analysis during the preparation of this report does confirm the approximately 1.0 NM CEP suggested by MIT, provided that the exact position and velocity of the spacecraft at the entry point are known in the spacecraft guidance computer, and provided that trajectory dispersions in earlier positions of the flight do not result in entry conditions much different from nominal. (The allowable departures of entry conditions from nominal are given in this report.) Inaccurate knowledge of the entry conditions leads naturally to a larger CEP.

The nominal trajectory used in this report was taken from the GSOP. This is because the Operational Trajectory (OT) was not available while the report was being prepared and because the Reference Trajectory [3] employs constants in the entry guidance equations which differ from those to be used in the actual flight (and in the OT).

*See Section 3.1 for a discussion of this criterion.

The guidance equations also were taken from the GSOP but were corrected to agree with the actual flight computer program [11,12,13]. Computer simulation work was conducted at Bellcomm using two different computer programs (see Appendix 2).

This report begins with a description of the GSOP nominal trajectory and with comments on the requirements put on the mission. The effects of position and velocity deviations and uncertainties on guidance accuracy are then examined, followed by similar discussions on the effects of Inertial Measuring Unit errors, of lift to drag ratio variations, of variations of other spacecraft parameters, and of variations in environmental factors. These discussions form the first part of the report.

The second part of the report is concerned, as noted earlier, with why the trajectory is as it is. The guidance equations used are shown, and comparisons are drawn between the theoretical predictions made by the guidance computer concerning the trajectory and what actually occurs.

It is originally planned that this report would conclude with an evaluation of the usefulness of the AS-202 entry guidance equations for a super-circular entry velocity more typical of lunar missions (36,000 fps vice the 28,706 fps of AS-202). It turns out that this is not appropriate, since the equations proposed for use at such higher velocities are different (as suggested by the equations proposed for Mission AS-501). Therefore, this topic is not considered in this report, which is thus limited to the applicability of the equations for earth orbital satellite speed missions.

2.0 AS-202 Nominal Entry Description

Mission AS-202, among its other objectives, is designed to test the heat shield of the Command Module (CM). To produce the necessary heat load, the flight is a high lob shot, with an apogee in excess of 650 NM and with entry being preceded by one 84 second and two 3 second burns by the Service Propulsion System (SPS). The spacecraft passes through the approximate entry altitude of 399,188 feet* with an inertial velocity of 28,706 feet per second, a flight path angle of 93.57° below the radius vector and an azimuth relative to true north of 58.65° . The entry point is at 5.11°S geodetic latitude and 134.79°E longitude. The target for parachute opening (at 25,000 feet altitude) is at 17.25°N geodetic latitude and 170.0°E longitude, 2,640 NM from the entry point.

*400,000 ft is the definition of entry altitude. The data given are those available from the GSOP [1].

Figure 1 shows the trajectory relative to the Earth, beginning at the first entry point, while Figure 2 shows altitude and inertial velocity vs. time. Lift is down when the spacecraft (S/C) passes through 399,188 feet altitude ($t=0$ sec), and then is changed to lift up at $t=94$ sec when the deceleration first exceeds 0.2 g. Range and lateral control, using roll control of the lift force, begin at $t=128$ sec, continuing the pull up to a peak altitude of 265,157 feet (436 NM from the target at $t=370$ sec). A second entry follows and final phase control begins at $t=436$ sec, leading to the parachute opening at $t=850.3$ sec at 25,000 feet altitude. The S/C reaches this point with an accuracy of 0.5 NM under nominal conditions and has a relative velocity of 423 fps, a flight path angle of -86.5° and an azimuth of 249.5° . (No requirement is specified for AS-202 entry accuracy, see Appendix 1.) To achieve such accuracy, a position and velocity update is transmitted to the spacecraft prior to the SPS burns noted.

Guidance accuracy is achieved in Apollo entry by rolling the CM to put the lift force in the proper direction. Both range and lateral error are controlled simultaneously in this way. The computer program computes a quantity called Y, which is a conservative estimate ($Y < 40$ NM) of the lateral capability of the CM. If full lift up or down is required for range control, then lateral control is permitted only to the extent of $\pm 15^\circ$ rotation of the lift from full up or down; the rotation is reversed when the predicted lateral miss exceeds $Y/2$. If full lift up or down is not required, then the roll changes sign when the predicted lateral miss exceeds Y. Figure 3 shows* the roll command, lateral miss and Y (or $Y/2$) versus range-to-go (note that the trajectory events move from the right to the left in this plot). The trajectory is well planned, with only 6 NM lateral miss being predicted at the start of entry. The roll command is seen to change slowly over most of the flight, with two to three oscillations at the end of the flight (when lateral capability is least and large errors cannot be tolerated).

The heating requirement for this mission is that the heating rate at its first peak exceed $70 \text{ Btu/ft}^2/\text{sec}$, exceed $50 \text{ Btu/ft}^2/\text{sec}$ at the second peak, and that the total heat exceed $20,000 \text{ Btu/ft}^2$. These rates are due to convective heating only, and apply for the hot spot. Note that they are per square foot, not for the total entry heat shield. Formulas for this calculation are issued by the Structures and Mechanics Division, MSC; simulation results and heating requirements are consistent with each other.

*The marks on all of the plots are 10 seconds apart.

Figure 4 shows the time history of convective heat rate and total convective heat for the GSOP nominal mission. The first heating rate peak occurs only 150 sec. after entry begins, and reaches $74.1 \text{ Btu/ft}^2/\text{sec}$, an acceptable value. It reduces rapidly after that to low levels which fulfill one further requirement, that the heat rate fall below $19 \text{ Btu/ft}^2/\text{sec}$. for more than 60 seconds. In this nominal trajectory, this interval lasts for 101 sec. Heat rate then rises rapidly again as the CM makes its second entry into the atmosphere, but peaks at only $48.2 \text{ Btu/ft}^2/\text{sec}$.*

Entry guidance is also required to keep deceleration to less than $10g$. Figure 5 shows that the AS-202 entry does much better than this, the peak being only $3.6g$. On the first entry, the peak is only slightly above $1g$ with the larger decelerations coming after the second entry.

This trajectory is representative of orbital speed entry trajectories, which are designed to keep heating to acceptable levels but still permit range control. The initial phase reduces the velocity below skip-out levels, but the trajectory is kept at high altitudes to reduce heating and achieve range. Figures 6 and 7 show altitude vs inertial velocity, and range-to-go vs inertial velocity, bearing out these statements. Second entry occurs relatively close to the landing site but with velocity and flight path angle combinations which keep deceleration and heating low while permitting landing point control.

3.0 Limiting Conditions for Guidance Accuracy

3.1 Effects of Position Variations on Footprints

This section considers the effects of position variations from nominal at the entry altitude upon the accuracy at the target, and also notes the effects on heating and deceleration. Perfect navigation data at the start of entry, and nominal L/D are assumed.

Figure 8 shows two major curves superimposed on a projection of the Earth, while Figure 9 shows the same curves with additional data on rectangular graph paper. The outer curve is the so-called footprint: it locates where the CM would go if the roll angle (0° roll puts lift up) were held constant (throughout the flight) at the values marked. The inner curve shows that part of the footprint in which the target could be located and reached by the CM with a miss less than 20 NM. These curves are thus

*The Operational Trajectory may enter at higher velocity, and meet this heating rate requirement also. With the addition of radiative heating effects, the $50 \text{ Btu/ft}^2/\text{sec}$ will be met in the GSOP trajectory also.

an indirect approach to finding the effect of position variations at entry. By showing where the CM could go given nominal entry conditions, they also imply where it could come from to reach the target. Footprints are a somewhat traditional method of illustrating this topic.

There is no accuracy requirement for AS-202 (see Appendix 1), although the CEP specified in the general Apollo Program Specifications [9] is 10 NM, i.e., 50% of the flights should miss by 10NM although downrange and crossrange misses are neither equiprobable nor uncorrelated. Using 20 NM is an arbitrary choice of the authors of this report as a criterion of accurate guidance.

The unguided footprint extends from 840 NM to 5,253 NM distant from the entry point, and is + 186 NM wide at its broadest point. It is interesting to note that most of this longitudinal range is achieved by having the lift upwards and that + 100° roll trajectories correspond closely with 10g peak deceleration trajectories.

It is also interesting to note that the nominal target can be reached (theoretically) by having a constant vertical lift-to-drag ratio (L/D) equal to $(L/D)_{\text{nominal}} \cos 45^\circ$, i.e., if the L/D is purely vertical and equals 0.707 of the nominally 0.34 L/D ratio. Use of this L/D ratio is, in fact, one of the guiding principles in mission planning. In a guided trajectory, this result is achieved by varying the roll angle, as occurs in AS-202.

The accurate guidance region extends from 1,460 NM to 3,470 NM from the entry point, and is + 150 NM wide at its broadest point. (Accuracy falls off rapidly as this inner boundary is exceeded.) The Apollo Program Specification [9] requires 1,500 NM to 2,500 NM, which is, therefore, more than adequately met with AS-202 entry conditions. It is stressed that this curve applies only for the initial position and velocity conditions at the start of entry.

While it is unlikely in AS-202 that the target will be changed in flight, or just before flight, to avoid bad weather conditions, this inner curve shows the bounds within which this could be done. Figure 10 shows the effects on heating and maximum deceleration of such a change.

Figure 10 shows that as the downrange distance is decreased, the first heat rate peak increases, the second heat peak decreases (and eventually disappears), the time below $19 \text{ Btu/ft}^2/\text{sec}$ heat rate decreases (to zero) and the total heat decreases. At no range does the second peak convective heat rate rise above the desired $50 \text{ Btu/ft}^2/\text{sec}$ for the entry conditions used.

It should be noted that the heat rates and loads for AS-202 are insufficient to qualify the heat shield for supercircular velocity entries. Mission AS-201 embodied a high heat rate test of the heat shield (see Figure 11). Together AS-201 and AS-202 qualify the heat shield for Earth orbital missions. They also offer sufficient data so that, by extrapolation, the heat shield may be qualified for shallow LOR velocity entries, but not for the full range of LOR entries. Full qualification of the heat shields for LOR missions must await missions AS-501 and AS-502.

Figure 10 also shows that peak deceleration increases as range decreases, but this effect is relatively slight, the maximum deceleration remaining less than 10 G in the accurate guidance region. Total heat load is least when deceleration is greatest, but peak heat rate is then increased and range capability reduced. Mission planning coordinates these factors with requirements and heat shield capability.

In passing, however, it is proper to note that the adjustment of certain constants in the guidance equations can, for example, change the second peak heat rate. This happened to be the case in the Reference Trajectory [3] and was also borne out by simulation work at North American Aviation [7]. Second peak heating rates as high as $70 \text{ Btu/ft}^2/\text{sec}$ can be achieved in this way, seemingly without deleterious effects on other aspects of the mission. Why the guidance constants were changed to present values is not known to the writers, except that they do reduce the heating (which is appropriate for later missions).

Returning to the original topic of this section, Figures 8 and 9 may be used to indicate the possible variation in the entry point position. But in so doing it must be remembered that the inertial velocity vector must stay at its nominal value. From a flight mechanics viewpoint, this is a difficult condition to get in practice - shorter flights would probably enter more steeply, etc. Changing the entry flight path angle from -3.57° to -3.27° , for example, increases the maximum and minimum accurate ranges by over 300 NM. Thus, a trajectory which has an initial entry further downrange than nominal, and which is also shallow (for the same reason) has a double penalty to pay. The initial range to the target is reduced and, by being shallow, the minimum accurate range is increased: for both reasons, the margin between target range and minimum accurate range is reduced.

The GSOP lists the indication errors (1σ) in position at entry start as some 61,622 ft downrange and 29,053 ft cross range. Indication errors refer to the difference between the position estimate computed by the G&N systems and the actual position. Considering that the purpose of the G&N guidance is to make the estimated position equal the target position, then the

indication errors can be used to suggest the variation of the actual trajectory from nominal. Having the ground update the G&N systems estimate does not, of course, make the actual trajectory become the nominal trajectory. Rather, it informs the G&N system that the actual trajectory is not nominal. Hence, the ground update will reduce only the indication errors, promoting accurate guidance, but not restoring actual trajectory conditions to nominal. (See Section 3.3 for effects of indication errors).

Thus these errors of some 10 and 4.5 NM can be interpreted as trajectory variations. Comparison with the guidance accuracy footprint dimensions shows clearly that position variations at the entry altitude will have no practical effect on accuracy at touchdown.

3.2 Effect of Velocity Variations

Velocity variations, following the discussion of the previous sections, are variations in the actual velocity from nominal values. In what follows, again as in the previous section, the indication errors are considered to be zero, i.e., the ground has perfectly updated the position and velocity estimates in the G&N systems. The position at entry altitude is considered nominal, (as is the L/D ratio).

Figures 12, 13 and 14 are of interest. Figure 12 shows flight path angle and velocity magnitude effects, considering azimuth nominal. The upper curve is perhaps the most important curve in these figures, for it shows the skip out boundary. The spacecraft should not be allowed to skip out, if it is all preventable by controlling the SPS burn attitude. One reason is that it is very difficult to predict where the CM will go once it skips out - some simulated trajectories for conditions near the curve showed two full orbits before landing. By this time, spacecraft electrical power will be exhausted and the spacecraft will be uncontrollable.

Unfortunately, the data in the GSOP suggest that this possibility must be considered. At the nominal velocity of 28,706 fps, the minimum tolerable flight path angle is some -2.9° , where the nominal is -3.57° . This represents a variation of + 335 fps up. The GSOP gives 64 fps as 1σ for this variation, so 335 fps is + 5.2σ . By pure, theoretical Gaussian probability theory, 5.2σ is some 10^{-7} probable, and might be considered ignorable. But it is far better to be prudent and prepared than theoretically pure. Or, mathematically, the common assumptions of Gaussian probability distribution may be appropriate only near the mean of the distribution and not near the tails. Certainly the present situation is not "tail trivial". Flight controllers should carefully monitor the flight path angle predicted at entry.

The lower curve in Figure 12 shows the boundary beneath which the CM will fall short of its target. At nominal velocity, the critical flight path angle is -3.87° , a -150 fps variation. This is some 2.3σ , for which the associated Gaussian probability is 0.01. Preparedness is again justified.

This lower boundary is not a $10g$ boundary, which is much further below and is not shown. High deceleration rates are not a significant factor in Earth orbital entry with the AS-202 guidance equations. It is, of course, possible to purposely provoke high g 's by assuming rather unusual conditions: the entry guidance, in such cases, abandons range and lateral control in an effort to prevent high g 's, and performs rather well. Section 5 notes an abort trajectory in which high g 's occur.

Figure 13 shows azimuth and velocity magnitude effects, considering flight path angle nominal. At nominal velocity, fairly broad ranges of azimuth can be tolerated, between 54.6° and 61.6° with 58.65° nominal. Since 1° of azimuth represents approximately 500 fps laterally, these limits are some 2,000 fps left and 1,500 fps right. The GSOP gives 23 fps as 1σ .

Figure 14 completes the discussion, showing flight path angle and azimuth effects for nominal velocity magnitude. The skip out boundary forms the upper curve - it breaks somewhat rapidly (and with results varying between skip out and poor accuracy) as the azimuth limits are reached.

These curves show that the flight controller must strive most importantly for control of the flight path angle at the entry point - not so shallow that a skip-out can occur or so steep that the CM will fall short. This he can do with little regard for azimuth or velocity magnitude, since large variations in both of these are tolerable.

3.3 Effect of Position and Velocity Indication Errors

In this section it is assumed that the actual trajectory conditions are nominal, but that there are uncorrelated position and velocity indication errors. This situation corresponds to errors in the position and velocity update furnished by the ground to the G&N system when a nominal flight has occurred. For G&N indications, the reverse situation is more generally true - the G&N indication is more likely to be nominal and the actual trajectory not, which situation is considered in the previous sections.

Table I shows the actual misses (when the actual altitude is 25,000 ft) caused by 1σ levels of G&N indication errors, using values taken from the GSOP. The errors expected from the update, as given in Reference 2, are some 6,000 ft and 5 fps, 1σ ,

both less than the values used for Table I. Thus Table I also represents errors at greater than 1σ levels for the update.

The table indicates that guidance accuracy is not remarkably affected by reasonably expectable levels of the update errors.

Addition analysis determined approximately the levels of error which can be tolerated. These are: over 100,000 ft. in downrange position, over 20,000 ft. in altitude, over 80,000 ft. in crossrange position, over 50 fps in downrange velocity, over 200 fps in vertical velocity and over 100 fps in cross range velocity, all given at the start of entry. In particular, an uncertainty in vertical velocity does not cause skip out as an actual variation may.

Considered in light of the update errors, (nominal trajectory, update errors) these limits suggest there is no problem. In the light of G&N system indication errors, (indication nominal, actual trajectory not nominal) the limits in downrange position, altitude and vertical velocity are at approximately 2.5σ levels; larger limits are desirable. Still, the latter condition is less realistic than having the G&N indication nominal and the actual trajectory off nominal, and so the analysis in the previous sections is preferred for this latter condition.

The coupling between downrange and cross range miss is partially due to rotation of the downrange direction as the orbit changes during flight. The azimuth, for example, changes from 58.65° at start of entry to 90.36° , the orbit inclination from 31.72° to 17.14° and the right ascension of the ascending node from 161.47° to 100.75° .

The coupling is also due to the guidance equations themselves. If only the indicated position and velocity were used by the guidance system, the errors could be expected to propagate fairly intact through the trajectory except for the rotation effect noted above and for the effect of the nearly 45° range angle covered in the flight. There would be some differences since the actual deceleration is sensed by the accelerometers and incorporated into the indicated position and velocity. But the guidance equations also use the measured deceleration directly in an important way, and thus there is a direct effect of the actual trajectory on the roll command as well as the indirect effect through position and velocity. For example, the lift is rotated from down to up when the measured acceleration first exceeds $0.2g$, and the next major guidance action occurs when the estimate of vertical velocity becomes more positive than -700 fps.

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Table 1

Effect of Position and Velocity Indication Errors
at Start of Entry on Guidance Accuracy

(Actual Condition are nominal)

Error	Error Magnitude	Downrange Miss, NM	Crossrange Miss, NM
Downrange position, ft.	+61,622 -61,622	+2.3 -2.5	-1.1 +0.9
Altitude, ft.	+11,542 -11,542	-1.6 1.8	+0.8 -1.0
Track (positive right), ft.	+28053 -28053	+1.2 -1.1	+2.4 -2.1
Downrange velocity, fps	+9.25 -9.25	+0.8 -0.5	+0.1 +0.7
Vertical velocity, fps	+64.05 -64.05	-4.7 +5.4	+3.5 -3.0
Track velocity, fps	+23. -23.	+1.1 -1.2	+2.3 -2.3

Errors are 1σ for G&N system.

It is interesting to note that Table I shows that a downrange position uncertainty of 61,622 ft (10.1 NM in range) at the start of entry becomes only some 2.3 NM range uncertainty at parachute altitude. This is due in part to the guidance acting on a mixture of measured data and estimated data but in greater part is a natural consequence of orbital mechanics. As a check, the transition matrices in circular orbit were numerically evaluated for a range angle equal to the range angle of this entry trajectory. The downrange and cross-range uncertainties to be expected agree well with what is shown in Table I. A circular orbit was used because most of the entry trajectory is spent at fairly high altitude with little percentage variation in distance from the center of the earth. What Table I does not show is the uncertainty in altitude at parachute opening: for the plus 61,622 ft downrange initial uncertainty, the final altitude uncertainty is 51,952 ft with the estimate being at 76,952 ft. altitude.

The fact that indication errors do affect the actual trajectory has important bearing on computational techniques for the development of CEP estimates. MIT's method [1], for example, is to take the time history of the accelerations from a perfectly nominal trajectory and to develop the indication errors by linear methods, without considering any alteration of the trajectory. This method seems, on the basis of the discussion above, inappropriate for the entry trajectory. The data presented in the table are the results of separate trajectory simulations for each uncertainty and are the differences between the error-free landing point and the landing point reached when the particular error is included.

3.4 Inertial Measuring Unit Effects

The Inertial Measuring Unit (IMU) in the CM is aligned before launch by optical and, primarily, by gyrocompassing techniques. The X axis of the stable member is upwards, along the local gravity vector (as it is sensed by the accelerometers). The declination of apparent gravity at KSC is given as 28.523169° [1]. The Z axis is aligned 104.9901° East of true North, and the Y axis is to the right, looking downrange. The IMU ceases gyrocompassing and goes inertial on receipt of the Guidance Reference Release signal, approximately 5 seconds before lift-off. During these 5 seconds the apparent azimuth increases to 105° , the desired launch azimuth. The platform is not realigned during flight.

As noted earlier, errors in the IMU cause indication errors, and are one cause of deviations of the actual trajectory from nominal. When the position and velocity of the CM are updated from the ground, the indication errors are reduced, but the platform is not realigned. Thus the initial misalignment of the IMU

at the start of entry is dependent upon the misalignment at launch and the drifting errors which accumulate during flight. Total flight time is short enough that platform drift error coefficients may be assumed constant throughout the flight (the correlation time for errors is said to be of the order of 20 hours).

The errors considered here are the uncompensated errors. Average drift in the gyros is accounted for by software programs which cause the gyros to be torque pulsed appropriately. Software equations compensate for average accelerometer errors. Thus the errors considered here are the variations from the average errors.

The 1σ values of these errors are given in the GSOP [1]. They are not given here because they are classified. Trajectories were computed to determine the individual effect of each error source at $+3\sigma$ and -3σ levels. For the $+3\sigma$ errors, the RSS effect is 1.29 NM downrange (which is at a true azimuth of 90.36° at the target) and 2.15 NM crossrange. For the -3σ errors, the RSS effects are 1.31 NM downrange and 1.71 NM crossrange. The net miss on a 1σ basis is thus approximately 0.775 NM due to IMU errors only. The correlation coefficient is high, about 0.7, between downrange and crossrange miss. The CEP corresponding to these errors is 0.91 NM, in good agreement with the GSOP estimate of 1.0 NM (although the GSOP data is based on indication errors, not actual miss).

The largest contributors to downrange miss are Z gyro drift, X axis initial misalignment, Z accelerometer bias, and misalignment of the Y accelerometer in the XY plane. For crossrange miss, the Z gyro drift dominates, with Y and Z accelerometer biases, X axis initial misalignment, and misalignment of the Y accelerometer in the XY plane as additional contributors.

It appears from this investigation that only a malfunctioning IMU can cause misses at the target greater than 20 NM., provided the position and velocity of the CM are updated prior to entry. Without the update, the GSOP suggests a CEP of 9.9 NM. Requiring the update is not unreasonable; the IMU was designed to be used only for short periods, with realignment to be made before use.

3.5 Effect of L/D Variations

The G&N guidance equations expect that a lift-to-drag (L/D) ratio of 0.33 will be available from the spacecraft at high velocities. By the Apollo Program Specification, the L/D is to be 0.34 ± 0.04 . But it is known that the measurement of L/D in hypersonic tunnels is difficult, that small c.g. motions

can cause considerable change (+ 0.015 L/D change for + 0.3" X axis c.g. motion or 0.2" Z axis c.g. motion), and that ablation effects change the L/D.

Consider then that the drag coefficient is nominal and that the lift coefficient is not. Investigation shows that flight path angle limits are most affected by L/D variations, so Figure 15 has been prepared to show the effects.

The skip out curve again shows itself in this figure and is seen to be affected by L/D in only a minor way. This skip out curve was obtained under active guidance - initial lift down followed later by lift up - but the same curve is obtained for lift down only for all practical purposes.

With a nominal flight path angle, entry guidance handles L/D increases with no problem and can handle L/D's as low as 0.25.

With steep trajectories there is the natural tendency to fall short. Increased L/D forestalls this effect. When L/D is low and the trajectory is shallow, it becomes difficult to predict whether a skip out or a fall short will occur, which is why the region near the "intersection" of the skip out and fall short curves is marked as it is in the figure.

Additional analysis shows that if both L and D change (keeping the same ratio) by as much as + 50%, that successful trajectories still result.

This is so, of course, because the lift and drag are sensed by the G&N system as they cause CM accelerations. Thus the G&N estimates of position and velocity are kept accurate, the probable landing point can still be predicted, and corrective action taken to reach the target (if enough L/D is available).

The probable L/D variations are, therefore, not to be feared.

3.6 Other Spacecraft Parameter Variation Effects

In addition to the L/D variation effects discussed in the previous section, the effects of other spacecraft parameters were investigated.

Roll Orientation of Lift, that is, the angle by which the lift is not in the spacecraft X-Z plane, was investigated. A 15° error was inserted: the miss remained under 0.5 am. This is because the actual lift force is sensed by the accelerometers. The roll angle time history is affected by this variation, but the guidance equations successfully cope with it. This error

would also correspond to readout errors (or misalignment) of the roll gimbal resolver. Up to 30° roll error can be tolerated.

Roll Acceleration and Rate Limits were also checked. The simulations used in this study did not include a perfect simulation of the autopilot, although the simulations are thought to be reasonable. Nominal roll acceleration was $10^\circ/\text{sec}^2$ and nominal roll rate limit was $15^\circ/\text{sec}$ (about the S/C X axis).

Reducing the acceleration to $2.5^\circ/\text{sec}^2$ did not affect the guidance accuracy. Reducing the roll rate as low as $4^\circ/\text{sec}$ also does not affect the guidance accuracy. In both cases, the guidance equations adjust the roll angle time history to compensate for the variations. This is of interest especially for manual control of entry or for use of only one of the two roll thrusters. Autopilots for attitude control during entry should be as simple as possible - high accuracy is simply not required.

Weight of the spacecraft, for a 10% variation, has no important effect on the guidance accuracy. This is because the deceleration is sensed. If the weight is reduced, the drag force causes greater deceleration. There is some self stabilizing effect in that the resulting slower velocity would reduce the drag force, but, in any case, the entry guidance equations adjust the roll angle time history to compensate for the change.

Angle of Attack for trim aerodynamic moment also has little effect on the guidance, for a variation of 5° . It is considered here that C_L and C_D , the trim lift and drag coefficients, are not

changed as trim angle of attack is changed, although this is an artificial assumption. (This error is also equivalent to a pitch misalignment of the IMU, although 5° is far beyond such a misalignment.) Angle of attack variation affects the RCS fuel usage: roll and yaw are coordinated so that the net motion occurs about the velocity vector, which is displaced by the angle of attack from the spacecraft X axis. If the actual angle of attack differs from the nominal, the amount of yawing will be in error, aerodynamic trim attitude will be disturbed, and additional RCS thrusting will occur to restore trim. But this is considered minor.

3.7 Environment Variation Effects

Atmospheric density variations of $\pm 10\%$ have little effect on guidance accuracy. This is because the acceleration of the spacecraft is sensed by the G&N system.

But peak heating rates are affected. An increase in air density of 10% reduces the magnitude of the first heating rate peak 2%, increases the second peak by 11% (to $53.7 \text{ Btu}/\text{ft}^2/\text{sec}$, meeting the requirement) and reduces the total heat by 7.5% (to 19,793 Btu/ft, less than the requirement).

The value of gravity also does not affect guidance accuracy. In the bulk of computer simulations made, the oblateness of the Earth was included using the J, H and D terms [14]. The effect of the Sun and Moon was not considered. In the guidance computer, navigation is computed using only the J term, and using the so-called "Average-G" method, see Figure 18. This Average-G technique works very well, causing an error of only 73 feet in position and 0.14 fps in inertial velocity at the end of entry. If the major gravitational attraction constant u is changed by .0007% (which is the probably uncertainty in it), the effect on miss is nil. Also negligible are the gravitational attraction of the Sun and the Moon (for which special tests were made).

Winds of 100 knots blowing steadily from any direction throughout the trajectory will have little effect on the accuracy to the 25,000 ft altitude point. This effect is sensed by the accelerometers and the trajectory compensated.

Winds blowing only below some 65,000 ft can cause misses of, say, two or three miles. This is because the G&N system ceases active guidance when its estimate of the relative velocity (ignoring winds) falls below 1,000 fps, which occurs at approximately 65,000 ft. The CM takes some 34 seconds to descend from 65,000 ft to 40,000 ft, 19 seconds more to reach 30,000 ft, and 11 seconds more to reach 25,000 ft. With the parachute open, it is especially affected by winds, but winds below 2,500 ft should be low (<28 kts. [14]) to permit rescue and recovery operations.

The jet stream type of wind is thus of chief interest. At, say, 300 kts., it exceeds the 99 percentile winds (of some 200 kts). But 300 kts. is only some 1/12 mile/second, so the miss should be less than 5 miles and more probably less than 3 miles since the jet stream is not deep.

4.0 Entry Guidance Equations

4.1 Entry Trajectory Phases

Figures 17-30 are flow charts for the entry guidance computations. They are taken from the GSOP [1], and have been annotated to show how they operated for mission AS-202. Figure 16 is a repeat of Figure 1, but is marked to show the different phases of the entry trajectory.

The nominal mission sequence of events, insofar as entry is concerned, begins some 4,322 seconds after launch when the CM/SM separation sequence is initiated. Entry mode is commanded 5 seconds later and by 4,373 seconds the CM is at the entry attitude. The pitch and yaw angles are set for aerodynamic trim of the CM, and the roll gimbal angle is some -165, for lift down (nominally 180°, but the target is

to the right of the trajectory so the lift is directed 15° to the right).

Although an indicator light is lit when the deceleration first reaches $0.05g$, the first real event in the entry is when $0.2g$ is experienced. Lift is then reversed from -165° to 0° (lift up) and the CM begins the Initial Roll phase (fig. 20) of flight. This occurs $2,194$ NM from the target and 94 seconds after the entry point (for the remainder of this section $t=0$ when the CM passes through $400,000$ ft altitude - the "entry" altitude).

This roll attitude is held until the vertical velocity becomes more positive than -700 fps at 128 seconds and with $2,034$ NM to go. The CM trajectory is considered to be somewhat stabilized at this point (captured by the atmosphere, but still with enough energy to have useful range capability).

At this point, the remaining trajectory to the target is planned and the control numbers computed which govern the rest of the trajectory. (More is said about trajectory planning in section 4.2). It turns out that the initial attempt at trajectory planning is satisfactory, i.e., that the velocities predicted are not too great, and so the Constant Drag phase of the trajectory is not needed (it being used as required to reduce velocity to safe levels). The CM therefore enters the Upcontrol phase immediately. Lift up is continued, with zero vertical velocity being achieved at 170 seconds, $1,843$ NM to go.

Lift up is still continued until at 318 seconds and $255,339$ ft. the acceleration falls below 6 fps². This point is $1,243$ NM from the target. This low acceleration is defined as the beginning of the Ballistic phase (figure 24). The flight is not really ballistic, however, since the deceleration never falls below 4.2 fps². The roll angle is held constant (at 48° , the last command given before entering this phase).

The Final phase (figure 25) begins when the deceleration rises above 6.5 fps², which occurs at 436 seconds, at an altitude of $255,339$ feet, with a range to go of 791 NM. In this phase, the range which will result from the present trajectory is computed, using present velocity magnitude, vertical velocity and deceleration as the important variables. The technique employs a reference trajectory (range, vertical velocity and deceleration vs. velocity magnitude) and influence functions about that trajectory to account for deviations. This predicted range to be covered is compared with the range to the target, and the error converted into L/D command using another influence function.

Note that range control is attempted only during this final phase of the trajectory. The technique in the earlier phases aims simply at carrying out the planned trajectory - at entering the Ballistic phase with the planned velocity and deceleration. The obvious implication is that if the planned trajectory is achieved, the miss will be small. This is discussed more fully in section 4.2.

Lateral Control (figure 28) is attempted throughout the flight, except for the Ballistic phase. But vertical L/D requirements for longitudinal control take precedence, and lateral control is reduced to a question of rolling right or left. That is, suppose the vertical L/D wanted can be achieved by a roll angle of $+60^\circ$ or -60° ($L/D \text{ vertical} = L/D_{\text{max}} \cos(\text{roll angle})$): then the lateral control determines whether the $+60^\circ$ or -60° is to be used. To reduce the amount of rolling, the lateral logic permits the lateral error to drift back and forth between limits given as the quantity Y. That is, suppose the roll angle actually is $+50^\circ$, and the commanded roll is $+60^\circ$. Lateral logic will use $+60^\circ$ unless the lateral miss exceeds Y, in which case the -60° will be used.

If full lift up or down is required for longitudinal control, not much is available for lateral control. The logic in Figure 28 permits roll angles of $\pm 15^\circ$ from 0° or 180° in such a case. This gives some lateral control, but not as much as might be desired. So $Y/2$ is used as the switching criteria, i.e., a more strenuous effort is made to keep the target dead ahead. Y is always conservative and so is $Y/2$.

Lateral motion turns out to be rather smooth and should be comfortable for the crew when this guidance is used for manned missions. Toward the end of the flight there are two or three roll oscillations, required because lateral capability is reducing rapidly, but they should not be annoying.

4.2 Trajectory Planning and Executive Compared

As can be seen from Figure 17 and the previous discussion, the entry trajectory can be divided into six phases. These are

1. Pre-0.2g phase (lift down)
2. Initial Roll (lift up)
3. Constant Drag
4. Upcontrol

5. Ballistic Phase

6. Final Phase

The first phase is used to promote capture and prevent skip-out: as is seen earlier, it cannot do this if the flight path angle is too shallow (figures 12 and 15). The second phase is to prevent too deep a penetration into the atmosphere after capture is insured.

After these two unguided phases are completed, the G&N program attempts to plan the remainder of the trajectory. What happens depends upon the present speed and the range to go. In the nominal mission (and for flights with total ranges greater than about 2,300 NM) it turns out that the constant drag phase is not needed and the Ballistic phase is. So the planning routine (see Figures 21 and 23) computes four major numbers, called V_1 , A_0 , V_L and Q_7 , as well as the ranges (ASKEP, etc., in Figure 22) to be traversed in the different phases. V_1 and A_0 apply to the first part of the upcontrol phase - if the theoretical trajectory implied by the equations at top left in Figure 24 occurs, the velocity V will equal V_1 when the deceleration D equals A_0 , and the vertical velocity $RDOT$ will then be zero. ASPDWN is the range predicted to be necessary for this to occur.

V_L and Q_7 apply for the second part of the Up-control phase. If the theoretical trajectory implied by the equations in the lower half of Figure 24 occurs, then V will equal V_L when D equal Q_7 and the flight path angle will be equal to $GAMAL$. ASPUP is the range predicted necessary for this to occur.

The range ASKEP is of next interest - it is the range predicted for the Ballistic phase, defined as that part of the trajectory between $D < Q_7$ and $D > Q_7 + 0.5 \text{ fps}^2$.

The Ballistic phase leads to the final phase, for which a stored reference trajectory is used to compute the guidance commands. The predicted range in this phase is the sum of ASP1 and ASP3. ASP1 is an approximation based on the stored reference trajectory: ASP3 is a correction for the difference between the predicted initial flight path angle (which equals $- GAMAL$ by the assumption of Ballistic flight) and the flight path angle implicit in the stored reference trajectory.*

*This report will not go into the approximations involved in the development of the equations used in entry guidance, for that subject should be considered by itself in a separate report. Such work is in progress. [25]

Type of Flight	Range at Planning Point-NM	UP CONTROL						Ballistic Range at ASKEP NM	Range at Start of Final Phase-NM	Terminal Miss NM	Remarks
		V1 fps	AO fps ²	Range at V=V1 NM	VL fps	Q7 fps ²	Range at D=Q7 NM				
AS-202 Nominal 2640 NM Total Range	2030	26871	47.1	1805	23887	6	1467	665	802	-	Planned
		26810	44.4	1860	23706	6	1243	452	791	0.5	Actual
Long Range 3282 NM	2685	27513	47.7	2461	24502	6	2115	1301	814	-	Planned
		27522	41.8	2579	24438	5.7	2067	1255	812	1.0	Actual
Shortest Range To Use Ballistic Phase, 2380 NM	1777	26490	47.6	1544	23495	6	1212	412	800	-	Planned
		26499	46.3	1568	23094	6	887	187	699	0.6	Actual

Table II - Comparison of Planned and Actual Entry Trajectory for Three Ranges

How this all turns out is summarized in Table II, which gives a comparison of the planned and actual trajectories for three ranges - the nominal AS-202 mission, a maximum range trajectory, and the shortest range trajectory (2,380 NM) which was the Ballistic Phase. Figures 31 and 32 show the long and short trajectories.

The maximum range mission shows up best. This is proper, for the Ballistic phase is then a relatively high lob with a predicted flight path angle of 1.16° at exit. That is, the assumption of zero drag in Ballistic flight is most closely realized in this trajectory. As the ranges get shorter, the predicted Ballistic range is seen to be less and less accurate with an error of 225 NM (412 NM predicted, 187 NM actual) for the shortest range trajectory.

Planning for the Upcontrol phase has similar errors. It is best for the longest range mission and makes errors similar to those for the ballistic phase. The errors compensate for each other, however, which is one reason the guidance works well.

Thus it appears that the entry guidance philosophy has two major features - a Ballistic phase used to control the range in a fairly gross way, and a Final phase which is capable of high accuracy. It would seem that the early phases might be simplified - the computations seem complex for the accuracy achieved and there are many "tuning factors" in the equations - but the results are very good, and all of the requirements are met handily (at least for the type mission considered in this report).

Historically, the approach used follows from the work by Sanger on the original rocket-glide bomber [16], and from work on the Dynasoar vehicle [17, 18]. Much of this early work is discussed in reference 19, which is an excellent survey of entry techniques and problems. Initial MIT proposals are documented in reference 20, prepared when the Apollo L/D was to be 0.5. Sanger's bomber and the Dynasoar were both high L/D vehicles (> 1 or 2). When the Apollo L/D was reduced to 0.34, the approach in reference 20 was modified to the present version. The AIAA Journal of Spacecraft and Rockets in particular continues to show new approaches [21, 22, 23] which generally aim towards simplifications in the techniques.*

The accuracy capabilities of the final phase guidance were brought out in a special set of simulation runs. In these runs the initial conditions were those of the beginning of the final phase in the nominal mission, but the target points were arbitrarily (and, effectively,

*Reference 24, MIT's discussion of entry guidance, has been distributed since this report was prepared.

suddenly) moved uprange or downrange. For a miss of 20 NM or less, the runs showed that the targets could be moved as much as 200 NM downrange or 260 NM uprange. The downrange limit was determined by the full lift up capability and the uprange limit by the 10g limit. At the g limit the guidance behaved well, permitting no more than 11.7 g to occur. Figure 28 shows the g limiter logic and, in particular, indicates an improvement to be incorporated for later missions, which should further improve g limiting and keep the peak closer to the required 10g max.

The Ballistic phase type of trajectory is not used for ranges less than some 2,300 NM. What happens then is that the trajectory range prediction overshoots the range to the target. As noted in Figure 22, when this happens the CM begins a Constant Drag phase. The equations in Figure 23 control the L/D to get the deceleration D to equal D_0 , which is always greater than 1.25g. Thus the CM slows down. Every two seconds, the G&N system tries again to plan a trajectory, and continues to fail until finally the computed VL falls below 18,000 fps, and the final phase is begun (see the middle left of Figure 21). Figure 33 shows a 1,553 NM trajectory which does this.

Thus there is no formal planning of the short range trajectory - the guidance hangs up in the Constant Drag phase until the final phase is begun. Of course, this is what was intended by the developers of the guidance scheme.

It appears, therefore, that the designers of the Apollo entry guidance have done a thorough and complete job. One can hypothesize that the technique could be simplified: more importantly, the technique gives high accuracy and satisfies all requirements, given the AS-202 trajectory conditions. Within those requirements it withstands off-nominal conditions well.

5.0 Abort Trajectories in the AS-202 Mission

Two abnormal conditions in the AS-202 mission are considered in this section. One is when the first Service Propulsion System (SPS) burn is unsuccessful and the second is when the second SPS burn is unsuccessful.

In the G&N computer, two landing points are stored - the Pacific site (at 17.5°N, 170°E) and the Atlantic site (at 4°N, 31°W). Which site is to be used is determined by the following process. The G&N computer computes the free fall time (T_{ff}) to 280,000 ft. When this falls below 160 seconds, CM/SM separation is commanded (if the ground has not already

ordered the separation). 5 seconds later, the velocity magnitude is checked; if it is greater than 24,196.2 fps, the Pacific site is used; if less, the Atlantic. This criterion is to be compared with the following normal inertial velocities: 21,751 fps at first SPS ignition; 25,416 at SPS cut-off; 22,440 fps at apogee; 25,116 fps at second SPS ignition; and 27,959 fps at final SPS cut-off. It appears that difficulties can arise if the first SPS burn is partially, but not fully, completed, for a CM touch-down in Africa is then possible. This has not been fully investigated by the writers, considering it beyond the scope of this report.

It turns out that if the SPS does not ignite at all the first time, that the Atlantic site can almost be reached (see Fig. 34). Full lift up is used (even the initial attitude is changed to full lift up), and the CM falls some 36 NM short. The entry is very steep (-7.5° flight path angle) and the g's immediately rise to 12.3 g even though lift is full up all the time. The first heating rate peak is 90.9 Btu/ft²/sec; there is no second peak, and the total heat is only 4,064 Btu/ft².

It also turns out that if the second SPS burn fails, the CM cannot reach the Pacific site (see Fig. 35). It falls short by 1,634 NM. Here again the entry is steep (-5.38° flight path angle) and 7.5 g's occur in the initial plunge. Lift is up continually after the 0.2 g level is sensed, but the range capability is just not sufficient. The target is 2,690 miles from the entry point, but the entry is too steep - too much energy is lost in the initial plunge into the atmosphere. The first heating rate peak is 115.5 Btu/ft²/sec, the second 10.5, and the total heat is only 8,924 Btu/ft².

6.0 Summary and Conclusions

This study has investigated the conditions under which accurate guidance in the entry phase can be maintained. Accurate guidance is arbitrarily defined as a miss less than 20 NM at the 25,000 ft altitude (at which the parachute opens). For this study the trajectory condition and similar data were taken from the G&N System Operations Plan [1]. The results of the study indicate the following:

1. The G&N system can accurately guide the command module to any point in an area extending from 1,460 NM to 3,470 NM down-range to the entry point and up to ± 150 NM to the side. The

target point lies 2,640 NM downrange, in the center of this area.

2. Variations in actual spacecraft position as it passes through 400,000 ft (the "entry altitude") have no effect on guidance accuracy, considering expected levels of the variations.
3. Guidance accuracy is most importantly affected by variations in flight path angle (at the entry altitude). Nominally, the angle is -3.57° . If it is more positive than -2.9° , the spacecraft will skip out of the atmosphere, and can make several full orbits after skip out, exhausting spacecraft electrical power while still in orbit. Commanding constant full lift down has negligible effect in such cases. If the flight path angle is more negative than -3.87° , the trajectory will fall short of the target.

Expectable variations in velocity magnitude and heading have negligible effects on guidance accuracy.

4. Errors in G&N system estimates of position and velocity have unimportant effects on guidance accuracy, provided the actual trajectory conditions are satisfactory. This applies considering the level of errors expected from ground determination of the spacecraft orbit. At the level of errors expected from G&N system navigation, effects become important if the errors in downrange position, altitude and vertical velocity individually exceed the 2.5σ level, approximately.
5. Inertial Measuring Unit (IMU) errors are tolerable up to levels at which the IMU would be considered to be malfunctioning.
6. Lift to drag (L/D) ratio can be as low as 0.25 without affecting guidance accuracy. The upper limit was not determined but is greater than 0.55. This applies for nominal drag, with lift varying from nominal. But $\pm 50\%$ variations in lift and drag, simultaneously, are also tolerable. L/D variations have only a minor effect on skipout conditions.
7. Guidance accuracy is not affected by 10% variations in air density, by 100 knot winds from any directions, by the oblateness of the earth or by the nominal uncertainty in its gravity constant.

8. Guidance accuracy is not affected by 75% reductions in roll acceleration capability of the spacecraft, by 75% reductions in roll rate limits, by 5° variations in trim angle of attack, by 30° roll offset of the lift force, or by 10% variation in spacecraft mass. Manual control using one set of thrusters should be satisfactory. Autopilots for entry should be as simple as possible since attitude accuracy is not critical.
9. The AS-202 nominal entry trajectory includes 5 phases: lift down, lift up, uncontrol, ballistic, and final phase. At the end of the lift up, the G&N system plans the remaining trajectory and then attempts to execute the plan. This planning is not done very well due, apparently, to poor analytical approximations in the equations, but the errors tend to compensate each other. In the subjective opinion of the writers, the guidance equations used are probably more complex than necessary.

Accuracy is achieved principally in the final phase in which errors of some 200 NM can be corrected.

10. The g limiter section of the guidance equations is not used in the nominal entry, in which the maximum deceleration is only 3.6 g. When tested under adverse conditions the maximum g's experienced rose as high as 11.7g, above the 10g maximum specified. Improvements in this section are to be incorporated for AS-204 and later.

Based on these results the conclusion of the study is that the entry guidance program performs well, fulfilling all of the requirements put upon it by Apollo Program Specifications. While it may be complex, it is also a complete, working program, already coded and tested for Apollo computer use. It is recommended for use in AS-202 and in Apollo missions where entry is approximately at earth orbital speed. Its utility for supercircular entry speeds was not determined in this study; a modified version of these equations has already been proposed by MIT for supercircular speeds, and approved by the Guidance Software Control Panel, MSC, NASA.

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APPENDIX I

AS-202 Requirements and Directives - Entry Phase

Insofar as entry is concerned, the following directives and requirements are pertinent.

"Apollo Flight Mission Assignments" (M-D MA 500-11, SE 010-000-1, September 10, 1965) states that mission AS-202 primary objectives include verification of: the operation of the CM heat shield (adequacy for entry from low earth orbit), G&N, and evaluation of the heat shield at high heat load during entry at approximately 28,000 ft/sec velocity.

"Apollo Program Specification" (SE 005-001-1, Revision A, March 1, 1966) states requirements as follows: (also see later) 3.5.1.22 Entry The CM shall be capable of controlled flight through the earth's atmosphere to any preselected impact point having a ground range between 1500 NM and 2500 NM from the entry point (defined as the point at which the vehicle first descends through the 400,000 ft altitude level). Additionally, the CM shall be capable of safe flight to all extended ranges between 2500 NM and 3500 NM. Both of these shall be possible without exceeding a 10 g deceleration for inertial velocities up to 36,500 fps and equatorial inclinations between $\pm 90^\circ$. The design limit entry load for all CM systems shall be a 20 g deceleration.

3.5.1.23 Aerodynamic Characteristics The CM shall have an offset center of gravity (cg) which will produce a lift-to-drag ratio of 0.34 ± 0.04 at Mach 6. The direction of the lift vector shall be controllable through the use of the attitude control subsystem. 3.5.2.5.2 Accuracy The Primary Navigation Guidance and Control System shall be capable of guiding the CM during entry to the preselected point of parachute deployment within a 10 NM CEP.

"AS-202 Appendix to the Apollo Program Specification"

(March 1966) states as follows:

1.0 Scope ... These requirements are presented in this Appendix as deviations to the requirements specified for equipment for the lunar landing mission... Unless otherwise noted, the paragraphs in this Appendix replace in their entirety the identically numbered paragraphs in the body of the specification. 3.5.1.22 The CM shall be capable of controlled flight through the earth's atmosphere to a preselected impact area having a ground range of 2500 NM from the entry point... This shall be possible for a nominal inertial entry velocity of 28,500 fps and

Appendix I

a flight path angle of -3.5° . The design limit entry load for all CM systems shall be a 20 g deceleration.

3.5.1.23 Aerodynamic Characteristics No Change.

3.5.2.5.2 Accuracy Deleted.

Heat Load Requirements The Structures and Mechanics Division (ES) of the MSC has placed the following requirements on AS-202. For heat due only to convection effects, as calculated by the formula

$$\dot{Q} = 7201.321 \sqrt{\frac{\rho}{\rho_{s.1}}} \left(\frac{V}{26,000} \right)^{3.15} \text{ Btu/ft}^2/\text{sec}$$

(which applies for an $L/D = 0.34$), the first heat pulse shall exceed 70 Btu/sq. ft./sec, and the second shall exceed 50 Btu/sq. ft./sec. Between the first and second pulse, the heat rate shall fall below 19 Btu/sq.ft./sec for more than 60 sec, and the total heat shall exceed 20,000 Btu/sq.ft. This formula is appropriate for heating at the hot spot on the CM heat shield.

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APPENDIX II

Simulation Computer Programs

The trajectories discussed in this report were computed using two computer programs called "Monte Carlo Entry" (MCE) and Bellcomm Apollo Simulation Program (BCMASP). Features of these programs are discussed in this section.

MCE is a program developed by W. G. Heffron and J. E. Holcomb of Bellcomm, with the assistance of Mrs. S. B. Watson. It is largely in FORTRAN. Integration is accomplished using the Adams-Bashforth four point predictor equation with "precision-and-a-half", with Runge-Kutta equations to start the process. MCE programming includes the effects of up to 106 changes in nominal data, e.g., position, velocity, and attitude readout uncertainties, deviations and/or perturbations from nominal values (correlated or uncorrelated), atmospheric density changes, vehicle mass change, Inertial Measuring Unit (IMU) errors, etc. It has three modes of operation called Monte Carlo, Sensitivity Matrix and Single Run. In a Monte Carlo run, error sources are assumed to be normally distributed, 1 sigma values are input, random values are computed based on the input, and a trajectory computed. This is repeated as often as requested; mean value and standard deviations (for some 78 quantities) are computed at selected trajectory events or times, as requested. In a Sensitivity Matrix run, a no-error trajectory is run first, followed by single error source trajectories for each non-zero error source input. The difference between the no-error trajectory and, say, the trajectory in which X gyro bias drift is non-zero is thus due to the non-zero X gyro bias drift and shows the sensitivity of the trajectory to that error source. An assumption of linearity in the sensitivity is implied (which turns out to be appropriate for the entry trajectory). From the sensitivity matrix and knowledge of the standard deviations of the error sources, the program develops covariance matrices at selected trajectory events or times by matrix methods.

In the Single Run mode of operation, the error sources are incorporated at the values input and a single trajectory computed.

The MCE program, in addition to normal listings of trajectory results, also creates a binary tape of data. A program called PLOTAP, developed by Mrs. S. B. Watson with the assistance of Mr. J. E. Holcomb, uses these data to create the plots shown in this report.

Appendix II

In calculating a trajectory, MCE performs the following computations. It determines the CM attitude for aerodynamic trim with the roll angle controlled and computes the lift and drag forces and orientation (1962 standard atmosphere is used; trim lift and drag coefficients and trim angle of attack come from tables). IMU drift rates and accelerometer readings are then computed. Gravity forces are computed for an oblate earth. Every two seconds the entry guidance computations are performed (including the average g navigation algorithm) resulting in a roll command (roll about the relative velocity vector). Required gimbal angles are computed and a "bang-bang" minimum time with limited rate and acceleration autopilot is simulated which produces a time history table of roll gimbal angles. The variables are then integrated and the program returns to the attitude computation routine. A north pole inertial axis system is used, with the x-z plane being the Greenwich meridian at Guidance Reference Release (GRR), except that platform computations are made in platform axes (x axis is local plumb bob at the pad at GRR, z axis is a given azimuth from true north at GRR). One entry trajectory, which requires some 840 seconds flight time, takes 1.8 minutes on the IBM 7044 computer using 1 second time steps for all computations except the guidance routine, or 1.2 minutes using 2 second time steps.

The BCMASP program was developed by Bell Telephone Laboratories for Bellcomm and modified for the present study by Mr. I. Bogner with the assistance of Mr. W. G. Heffron. BCMASP is largely in FORTRAN and uses four point Runge-Kutta equations for integration. It includes the JPL ephemeris for sun and moon, the effects of earth oblateness, and the 1962 standard atmosphere. (BCMASP is being furnished to MSC and MSFC at the present time).

For purposes of this study, BCMASP has two basic modes of operation - Single Run (as in MCE) and Perturb. Perturb mode is similar to the Sensitivity Matrix mode in MCE, one error source being inserted at a time and trajectory differences being listed. BCMASP includes the simulation of an Inertial Measuring Unit having the standard error sources.

In calculating a trajectory with BCMASP the roll attitude is computed using a simplified rate limiting autopilot, with pitch and yaw assumed to be in aerodynamic trim. Lift and drag are computed using tabulated lift and drag coefficients and resolved along aerodynamic trim axes. IMU drift rates and accelerometer readings are calculated. Gravity forces are computed for an oblate earth. (Sun and moon effects can be included.) The entry guidance computations are entered every two seconds (including the average g navigation algorithm) producing the new roll angle command. Variables are integrated and the program returns to lift and drag

Appendix II

computations. Inertial axes of 1950.0 (first point of Aries) are the basic coordinate axes, but inertial axes of date and platform axes are also used as appropriate. One entry trajectory requires some 1.5 minutes on the IBM 7044 using 2 second time steps.

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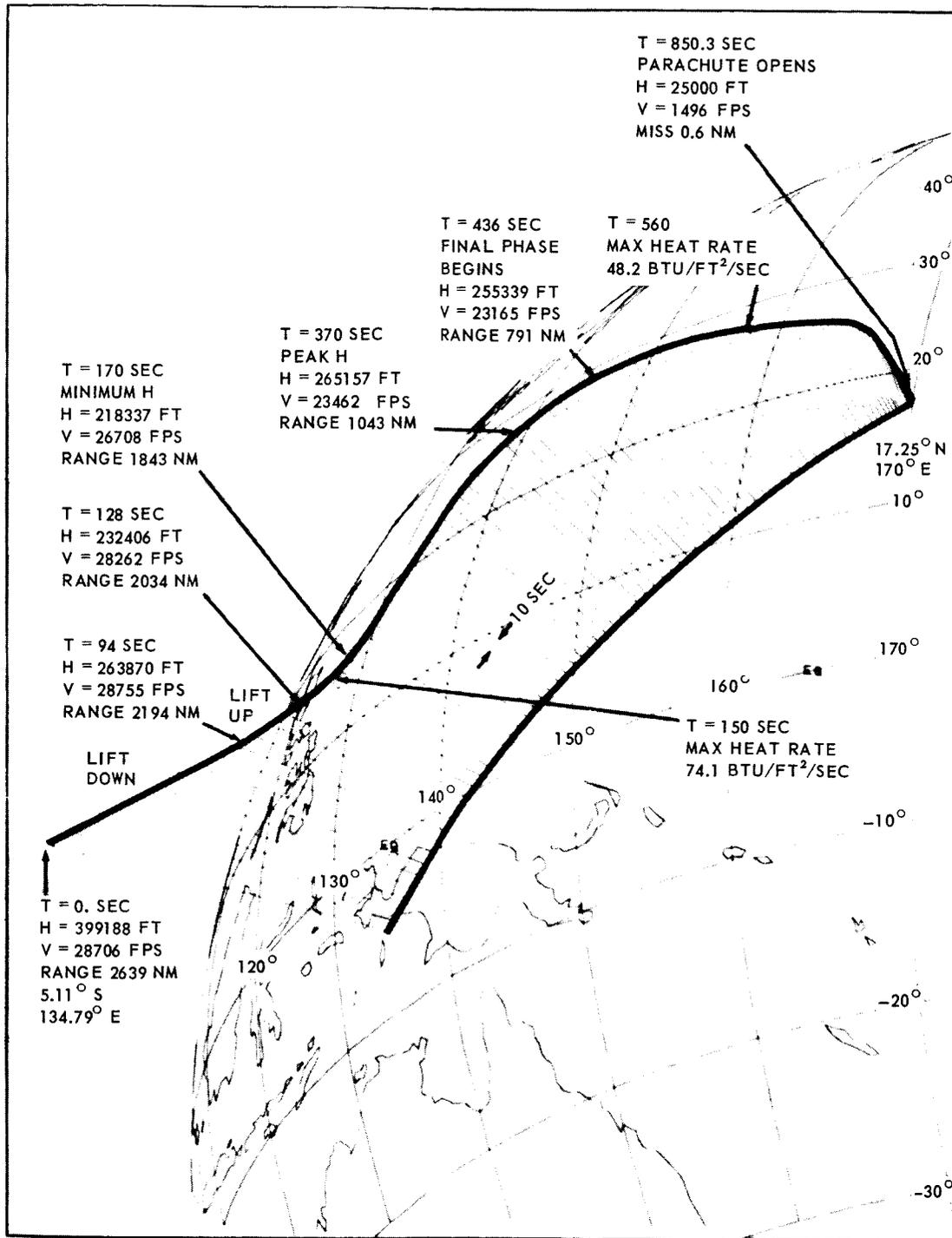


FIGURE 1. NOMINAL AS-202 ENTRY TRAJECTORY
(ALTITUDE EXAGGERATED 20X)

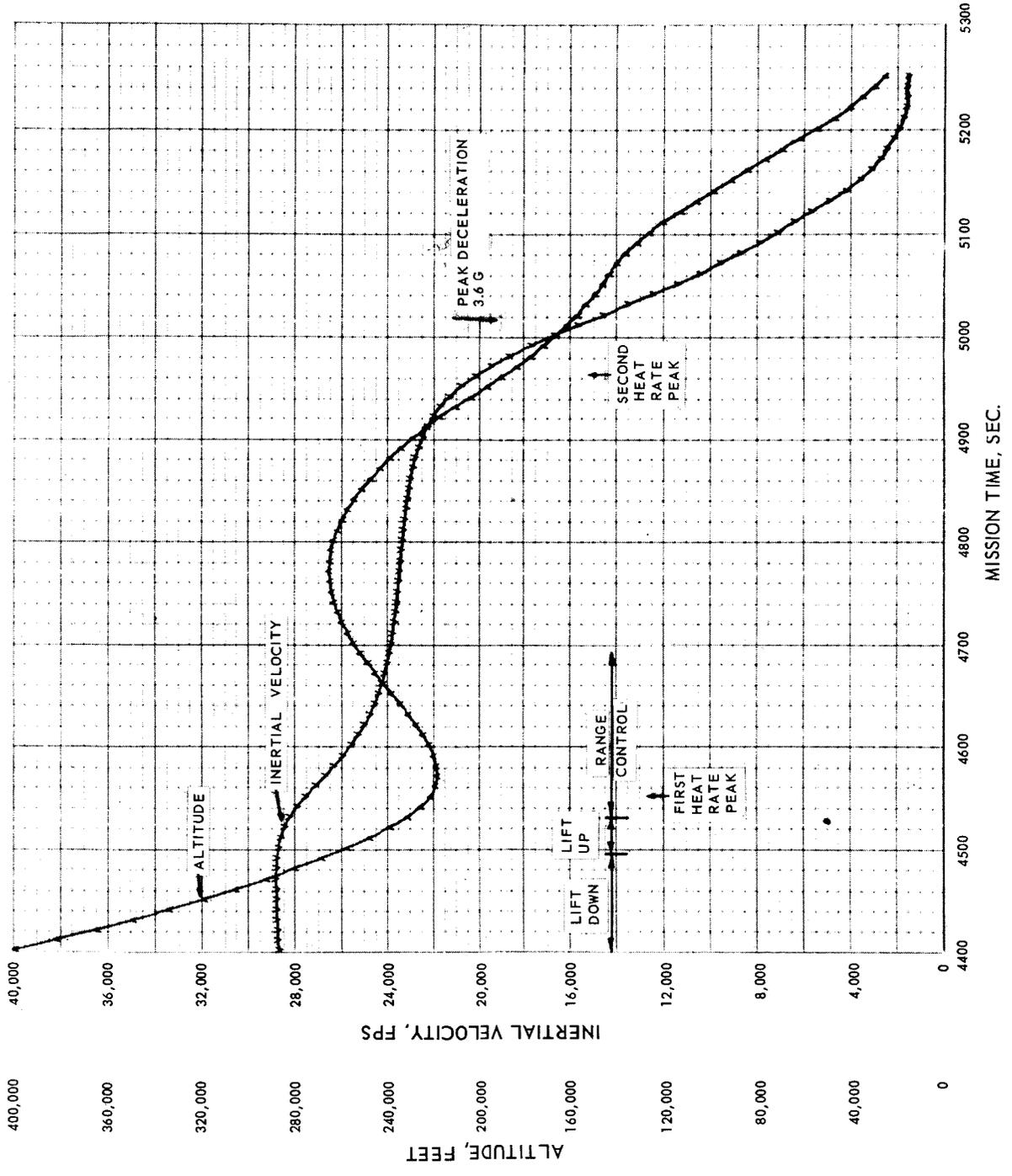


FIGURE 2. ALTITUDE AND INERTIAL VELOCITY FOR THE NOMINAL TRAJECTORY
(10 SEC BETWEEN MARKS ON PLOTS)

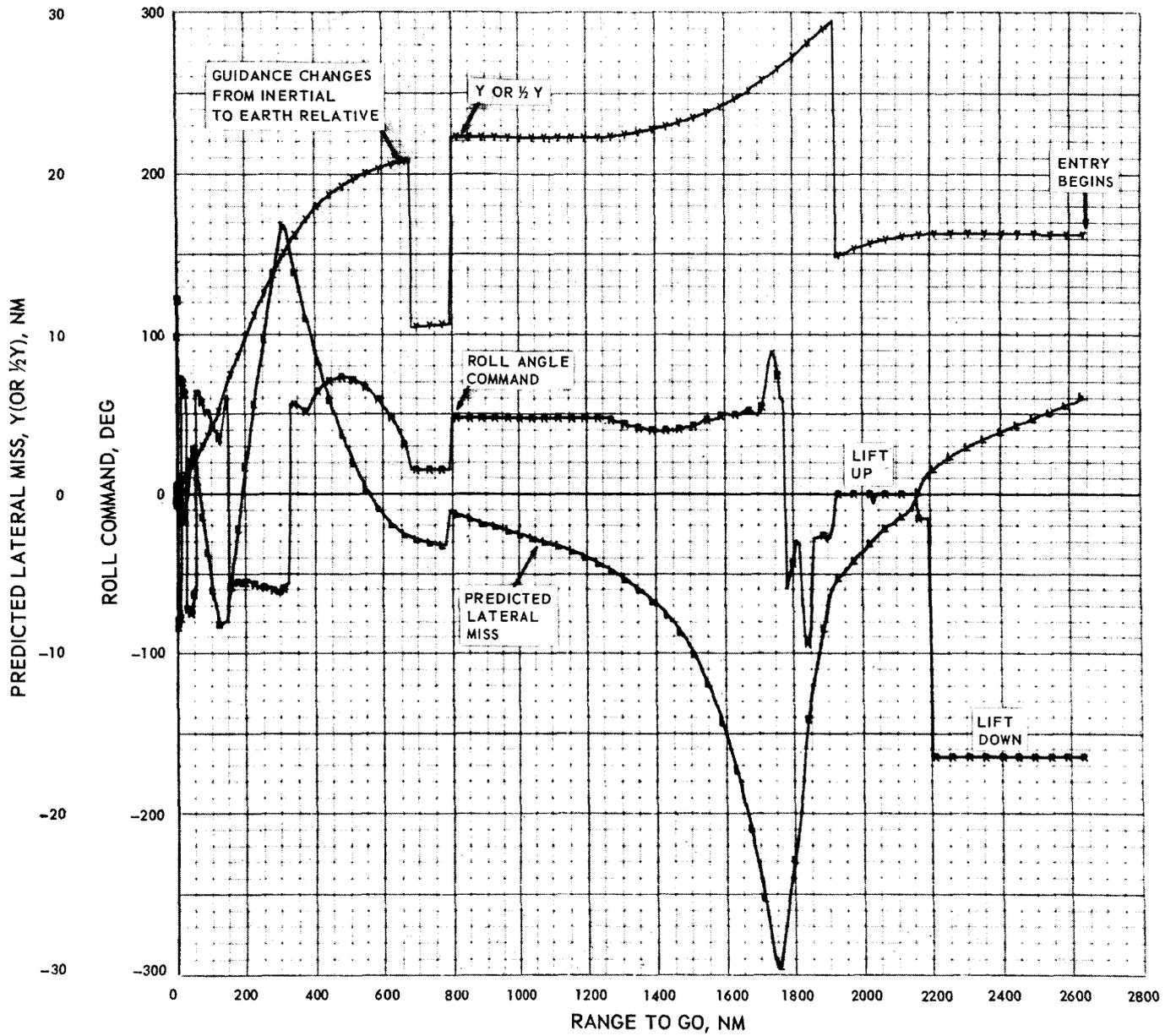


FIGURE 3. ROLL COMMAND, PREDICTED LATERAL MISS AND Y VERSUS RANGE TO GO FOR THE NOMINAL TRAJECTORY. (10 SEC BETWEEN MARKS ON PLOTS)

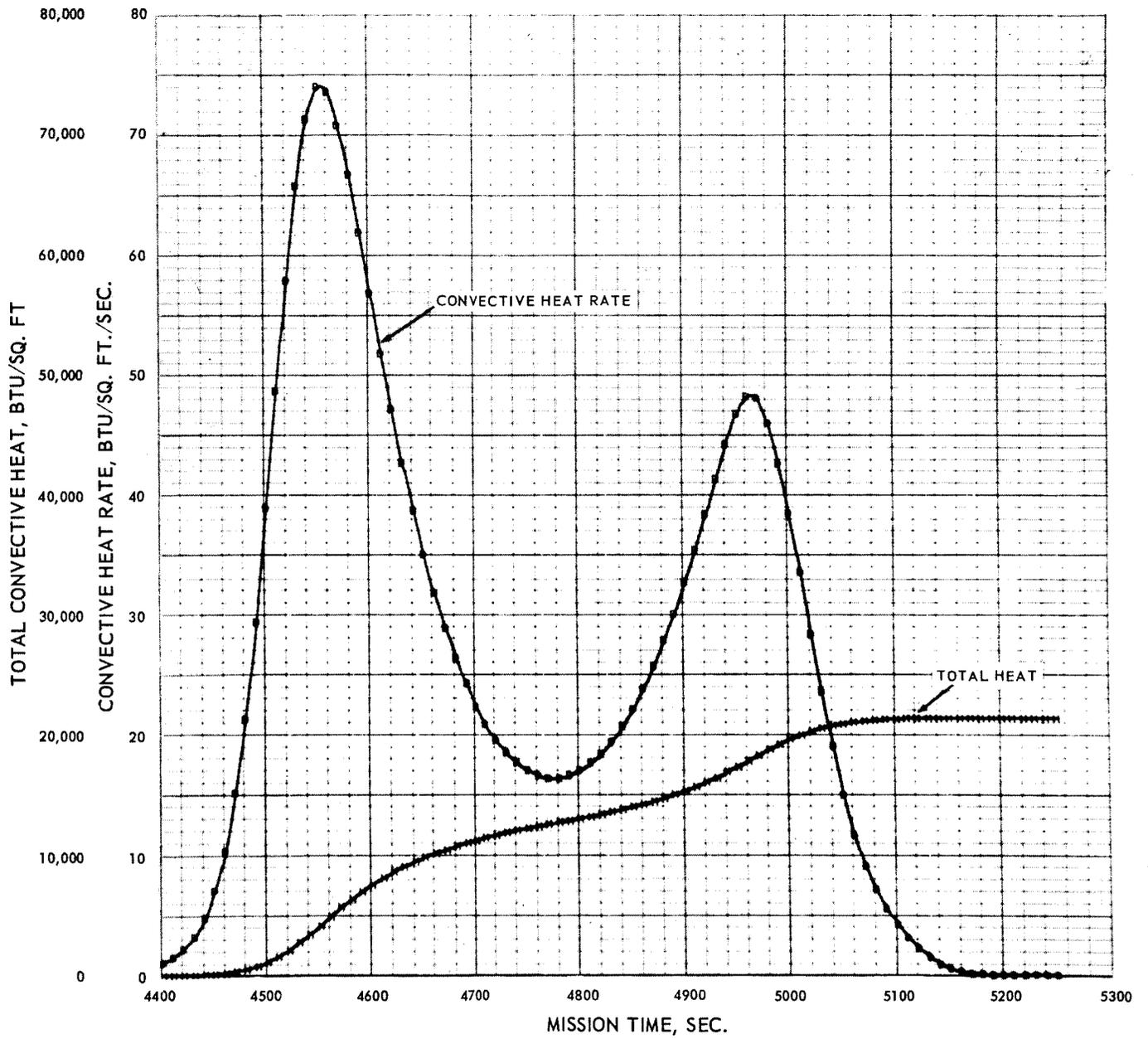


FIGURE 4. HEAT RATE AND HEAT FOR THE NOMINAL TRAJECTORY

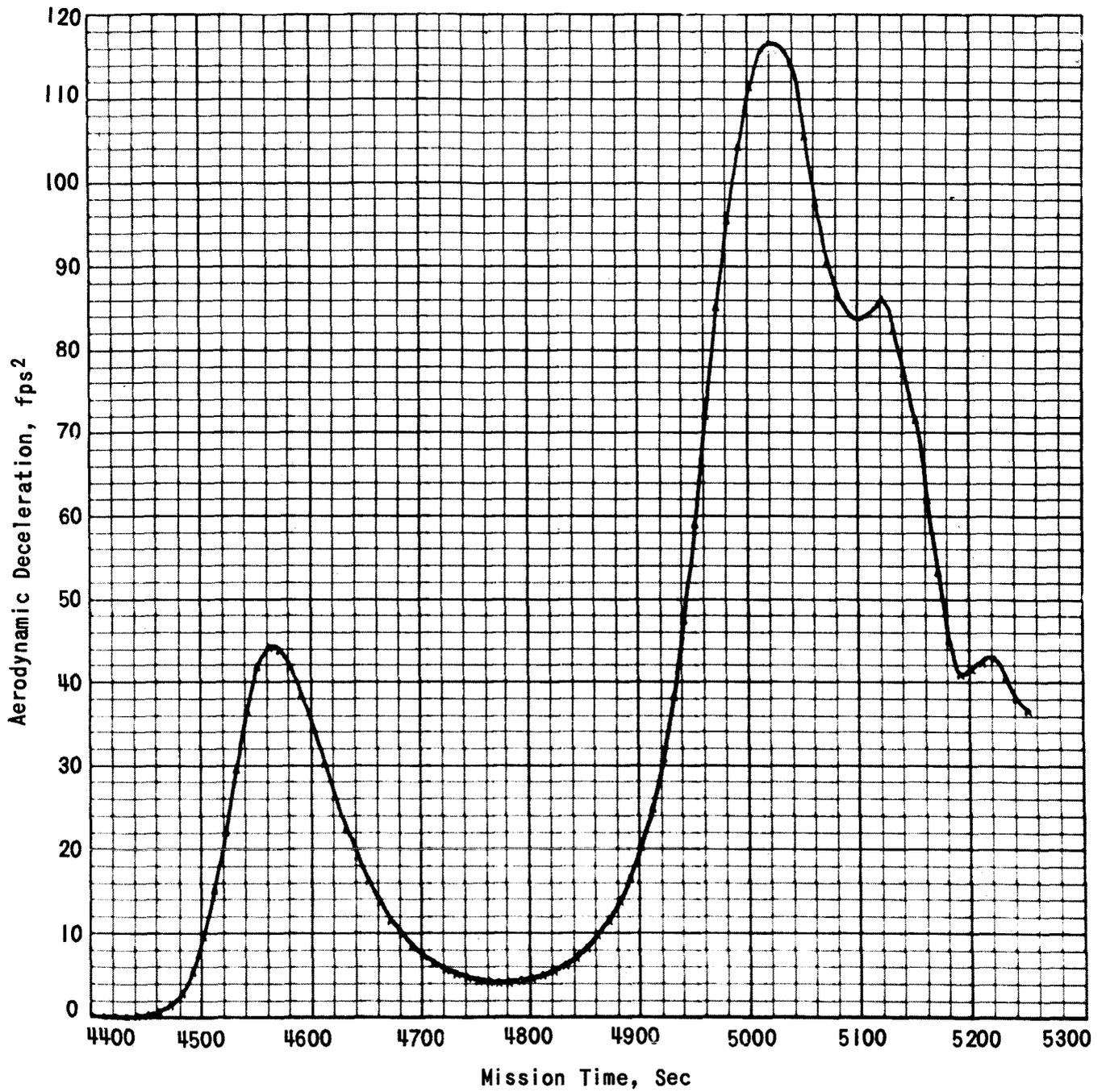


FIGURE 5 - AERODYNAMIC DECELERATION VS. TIME, NOMINAL MISSION

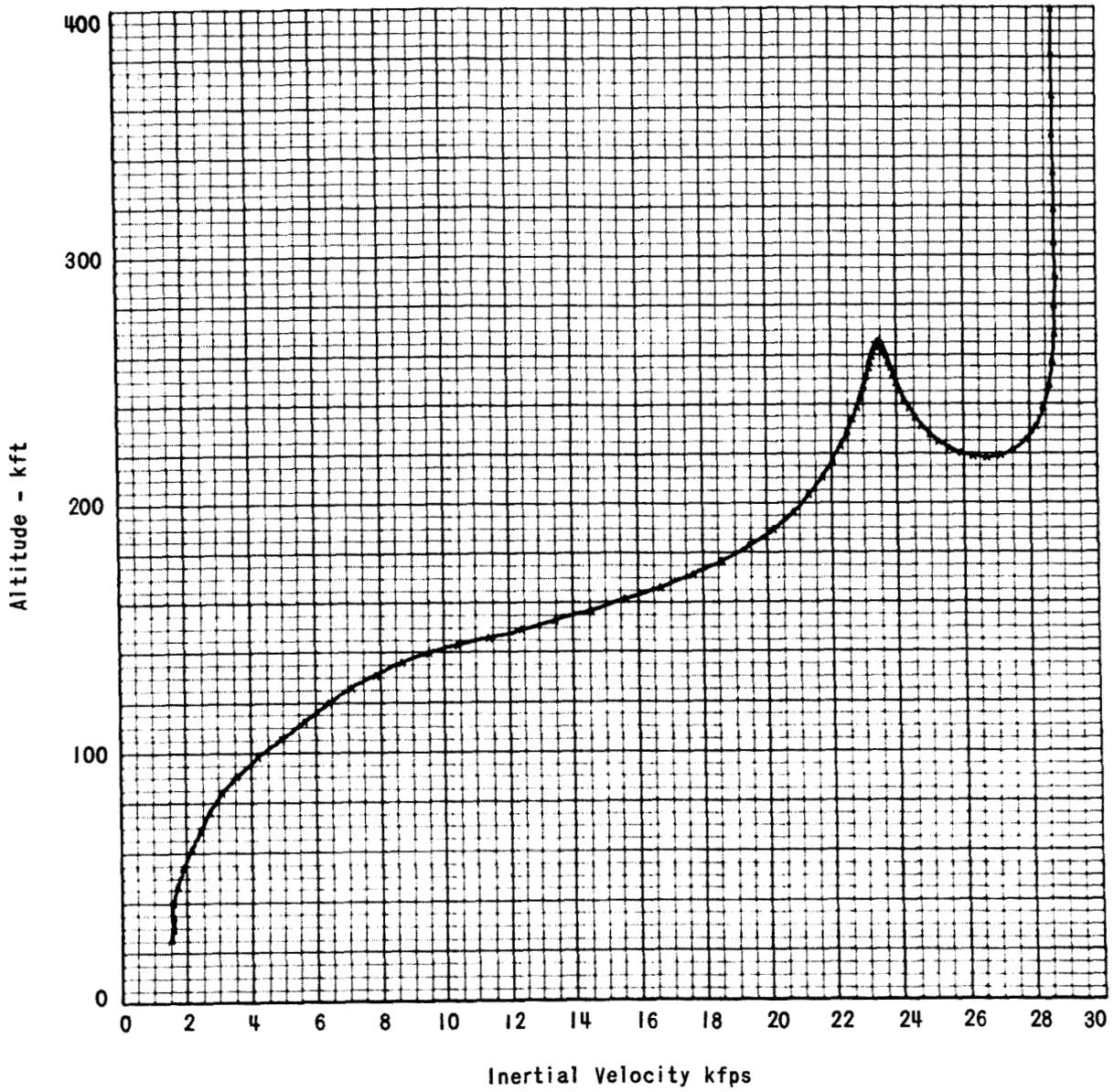


FIGURE 6 - ALTITUDE VS. INERTIAL VELOCITY, NOMINAL MISSION

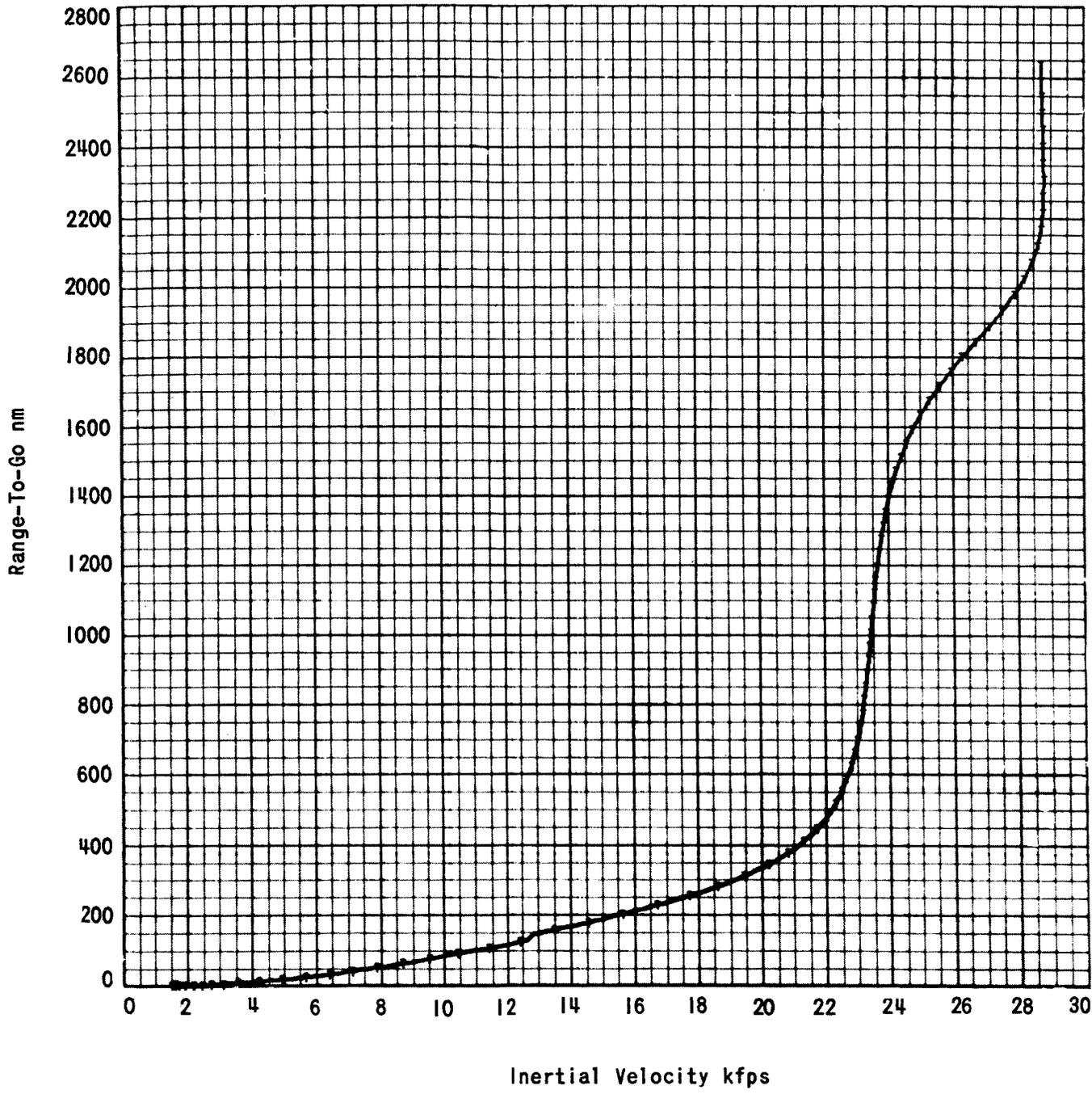


FIGURE 7 - RANGE-TO-GO VS. INERTIAL VELOCITY, NOMINAL MISSION

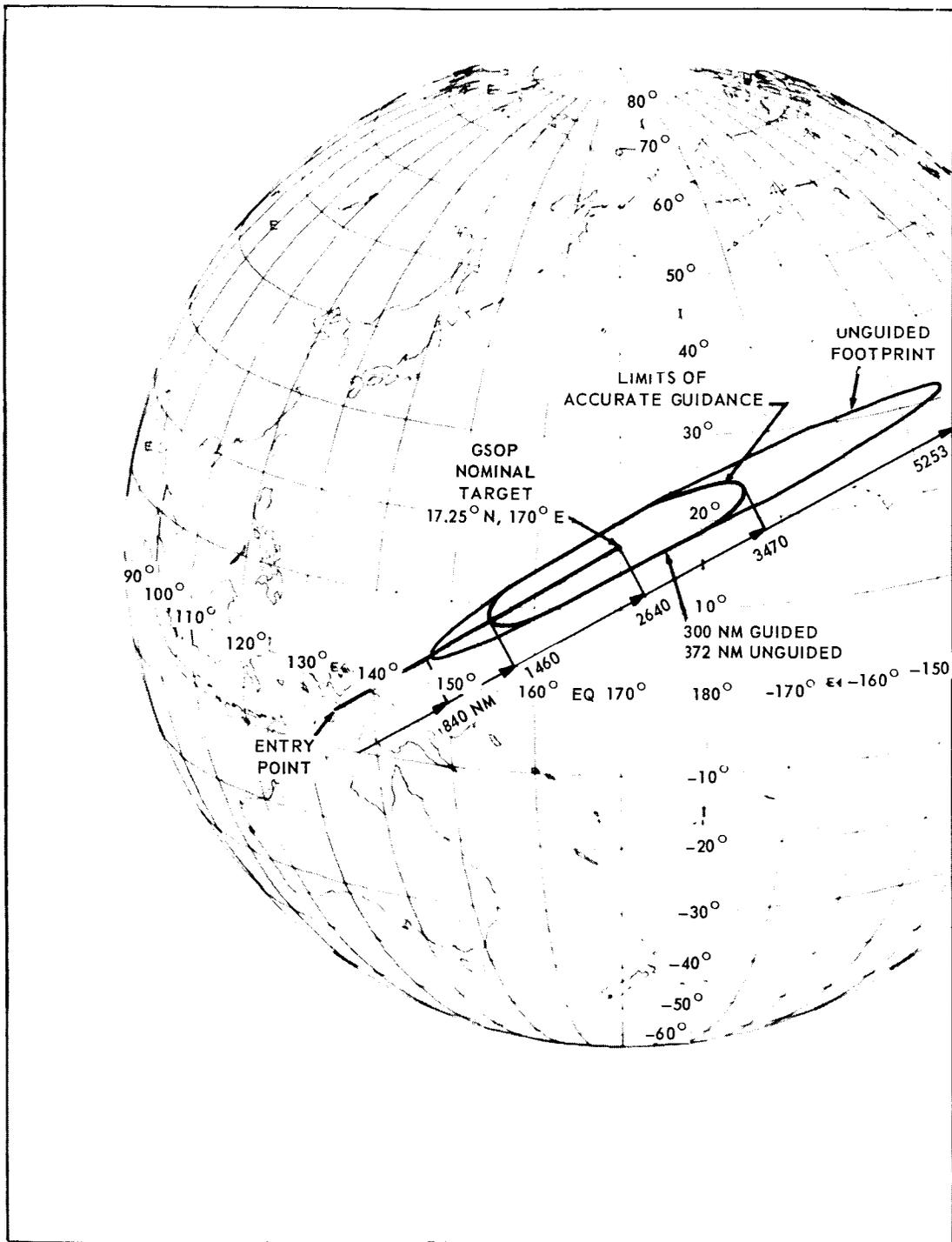


FIGURE 8 - AS-202 GUIDED AND UNGUIDED FOOTPRINTS

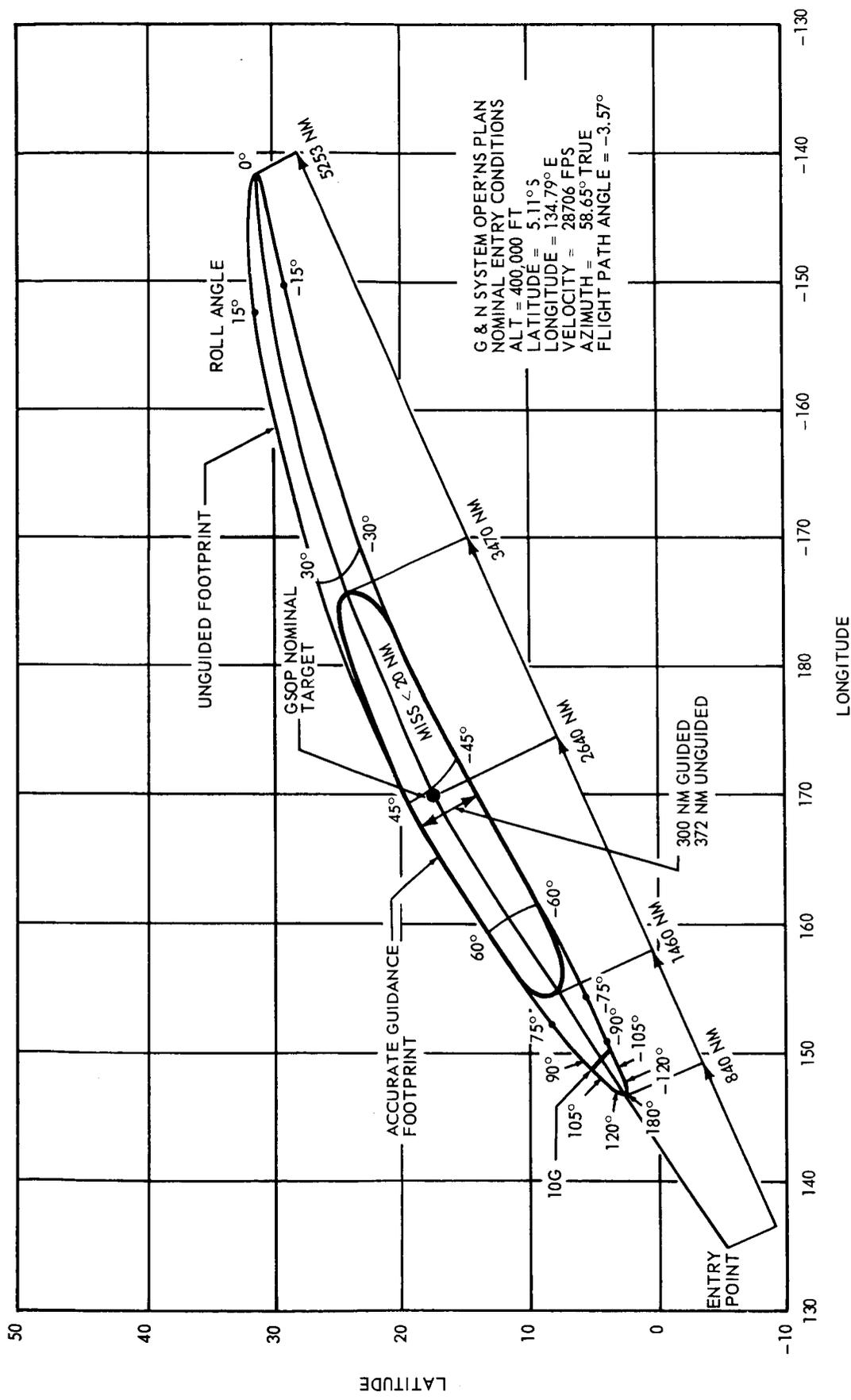


FIGURE 9 - AS-202 GUIDED AND UNGUIDED FOOTPRINTS

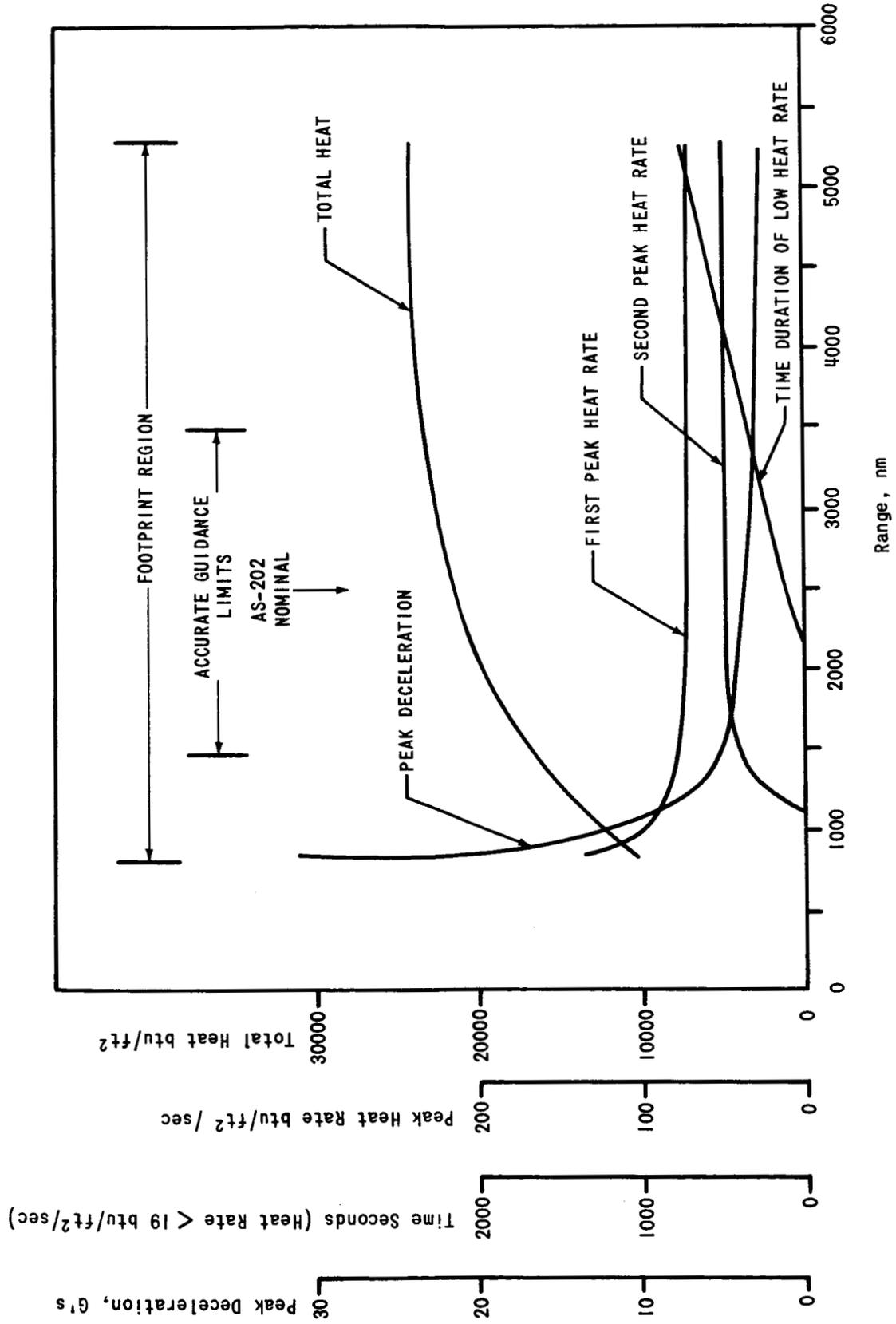


FIGURE 10 - EFFECT OF RANGE ON HEATING AND PEAK DECELERATION

APOLLO FLIGHT REGIME (TESTS)

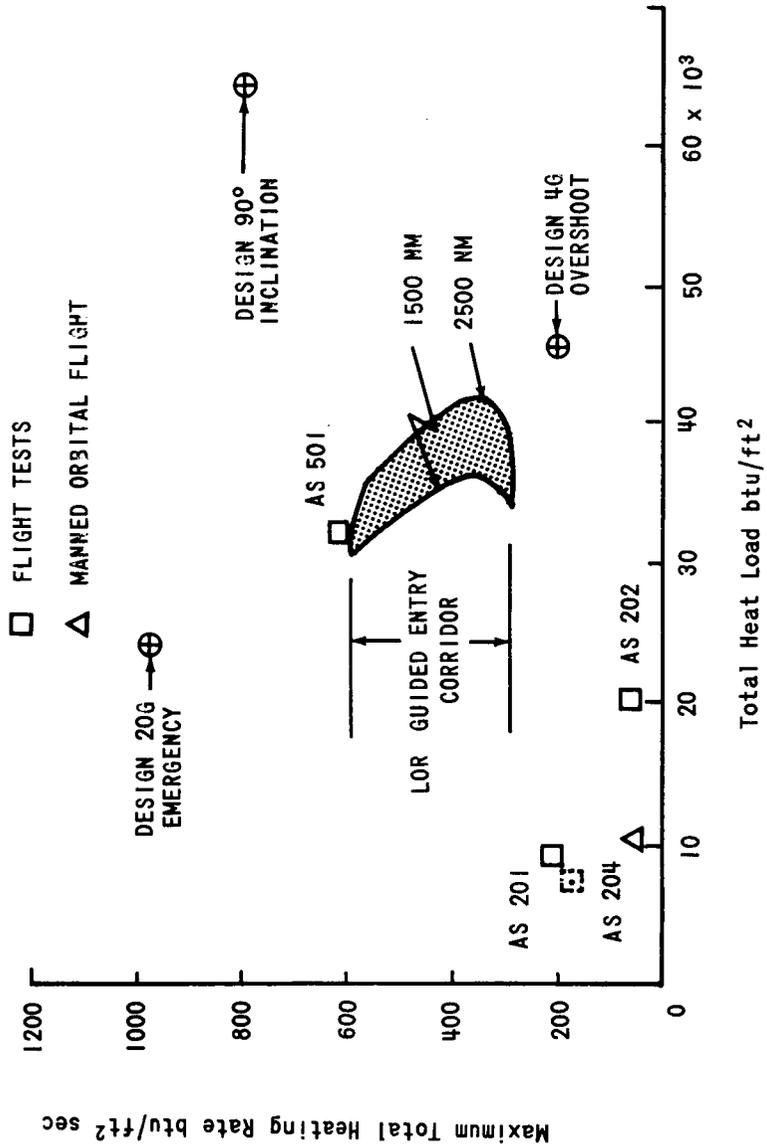


FIGURE 11 HEAT RATE AND TOTAL HEAT FOR APOLLO HEAT SHIELDS

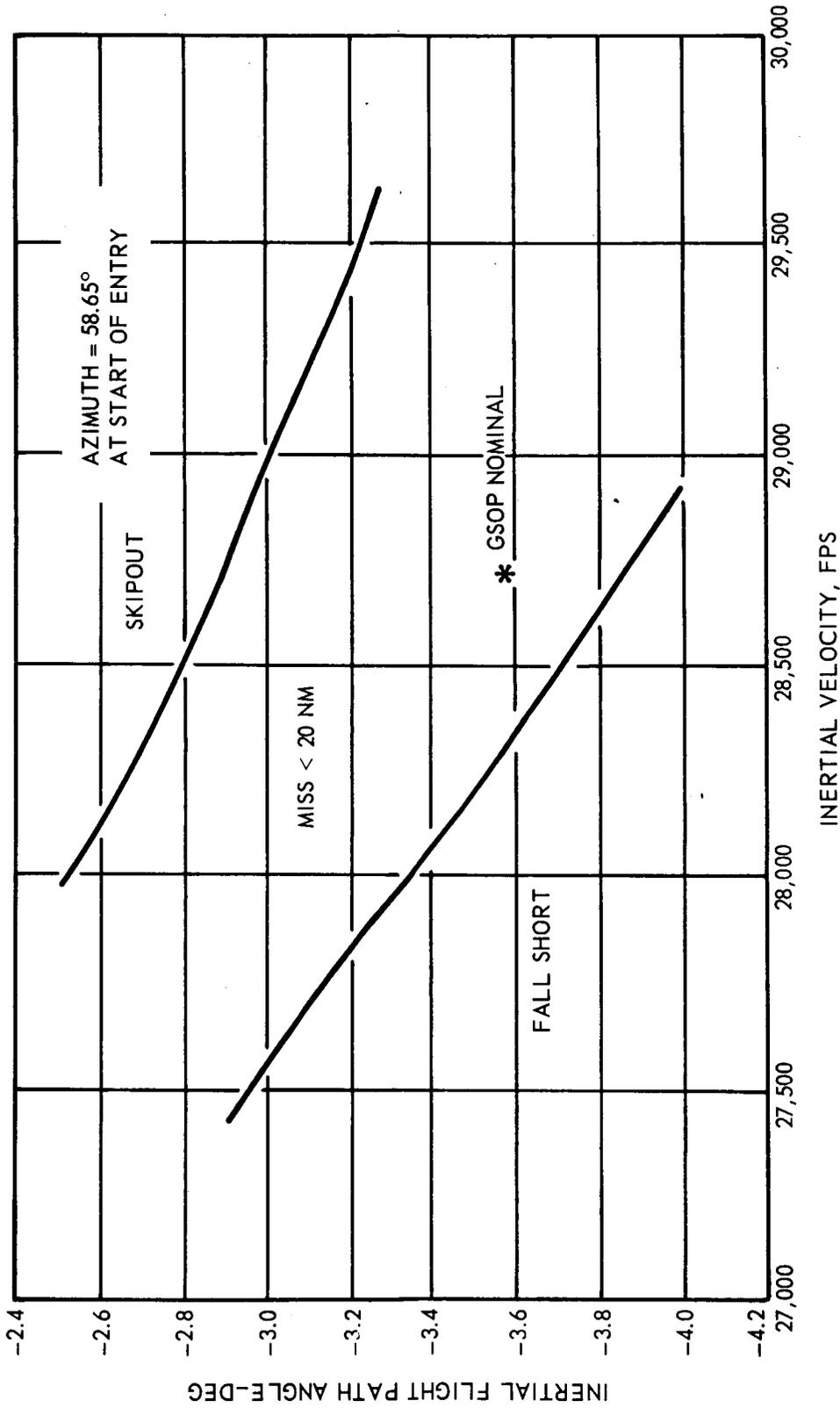


FIGURE 12 - EFFECT OF ENTRY VELOCITY AND FLIGHT PATH ANGLE ON GUIDANCE ACCURACY

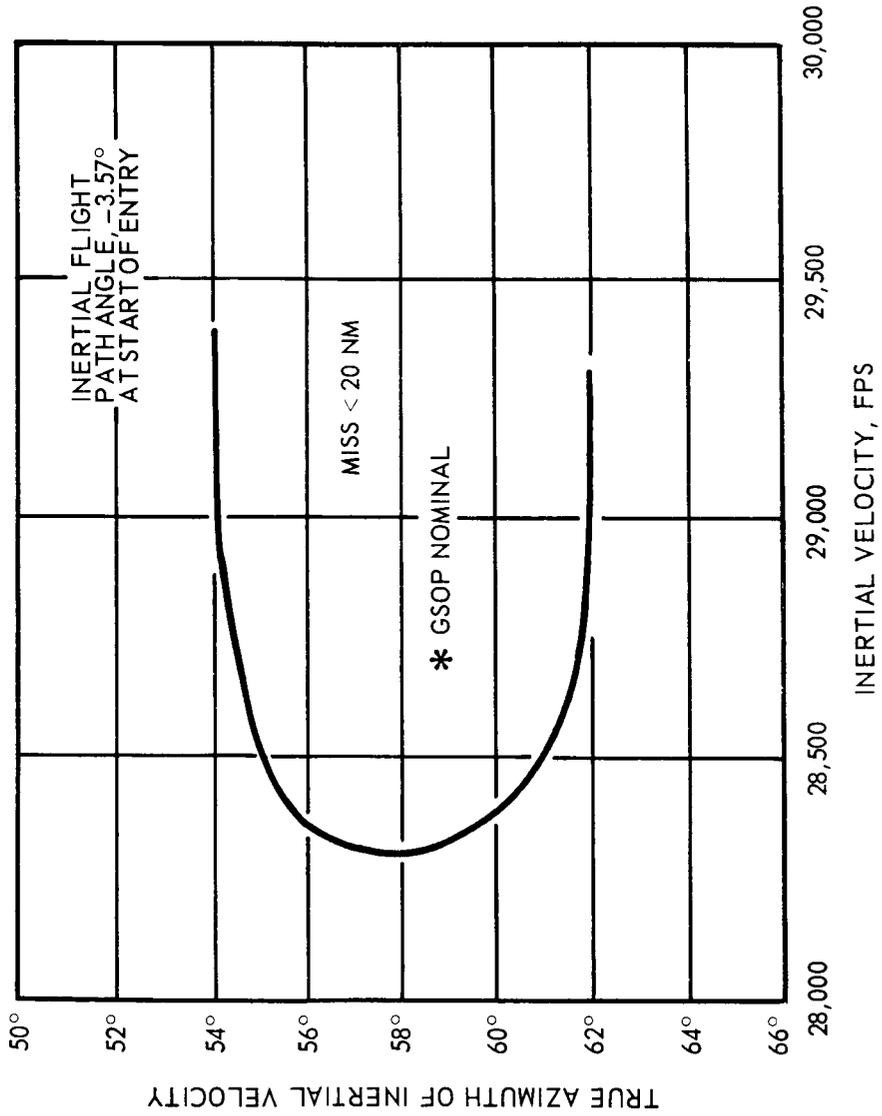


FIGURE 13 - EFFECT OF ENTRY VELOCITY AND AZIMUTH ON GUIDANCE ACCURACY

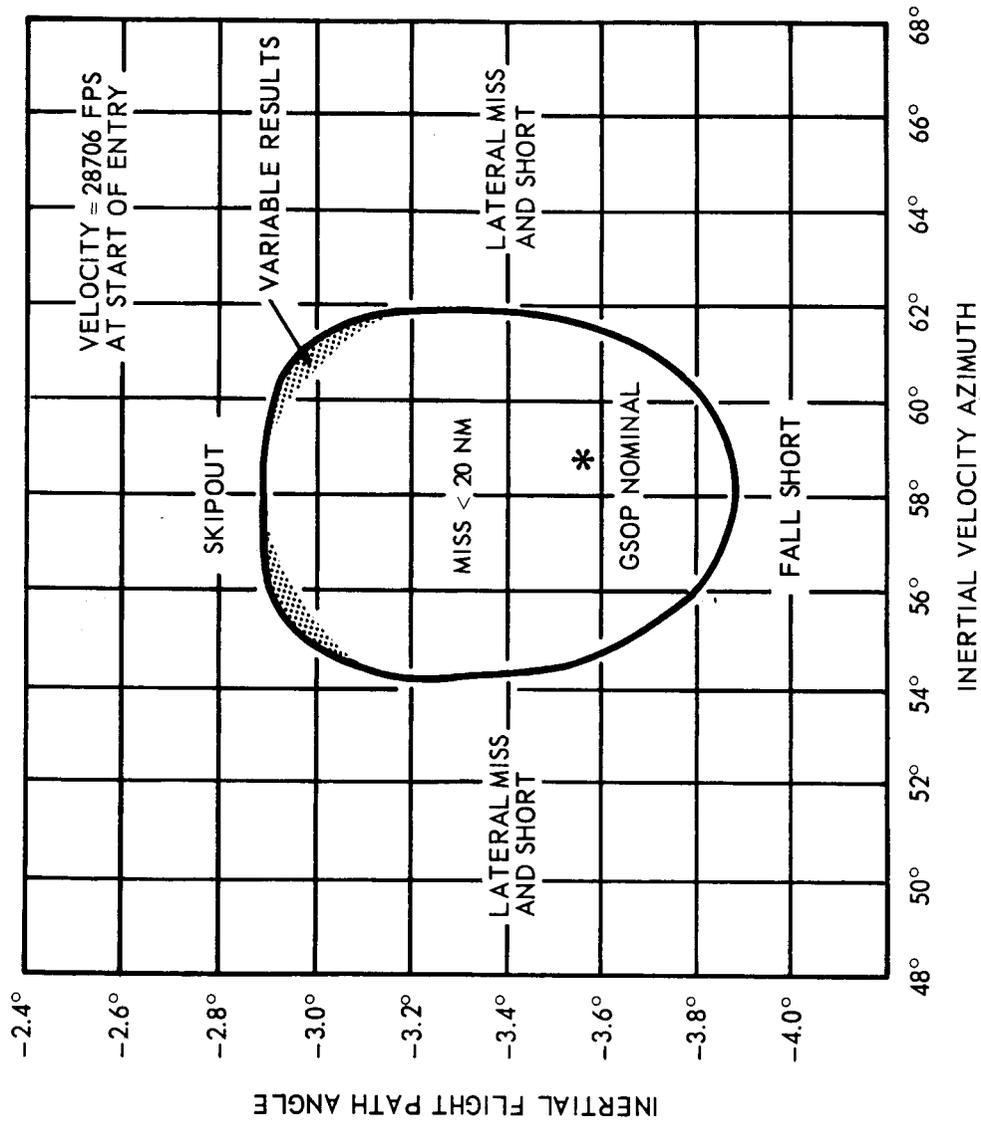


FIGURE 14 - EFFECT OF ENTRY VELOCITY AZIMUTH AND FLIGHT PATH ANGLE ON GUIDANCE ACCURACY

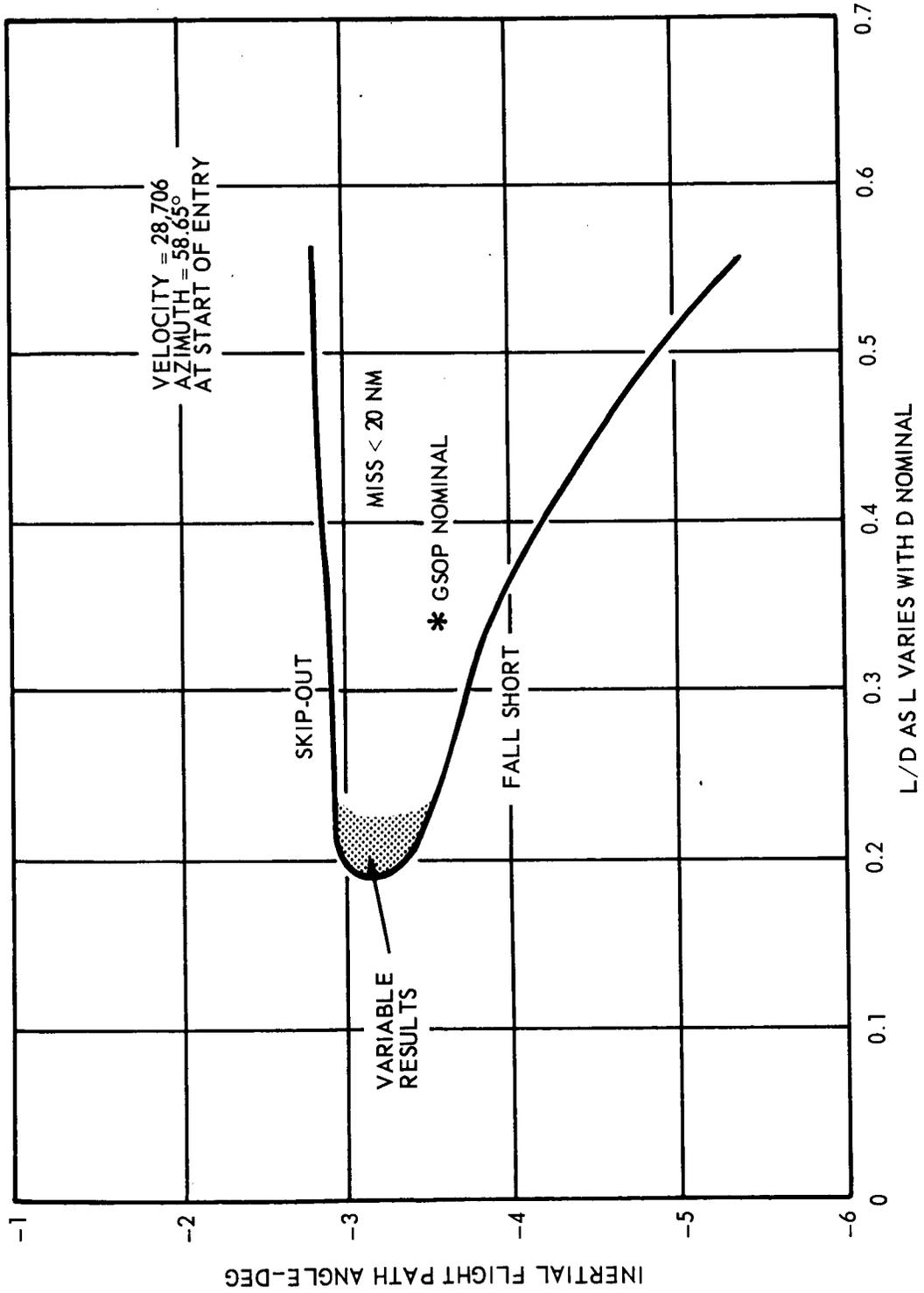


FIGURE 15 - EFFECT OF L/D AND FLIGHT PATH ANGLE ON GUIDANCE ACCURACY

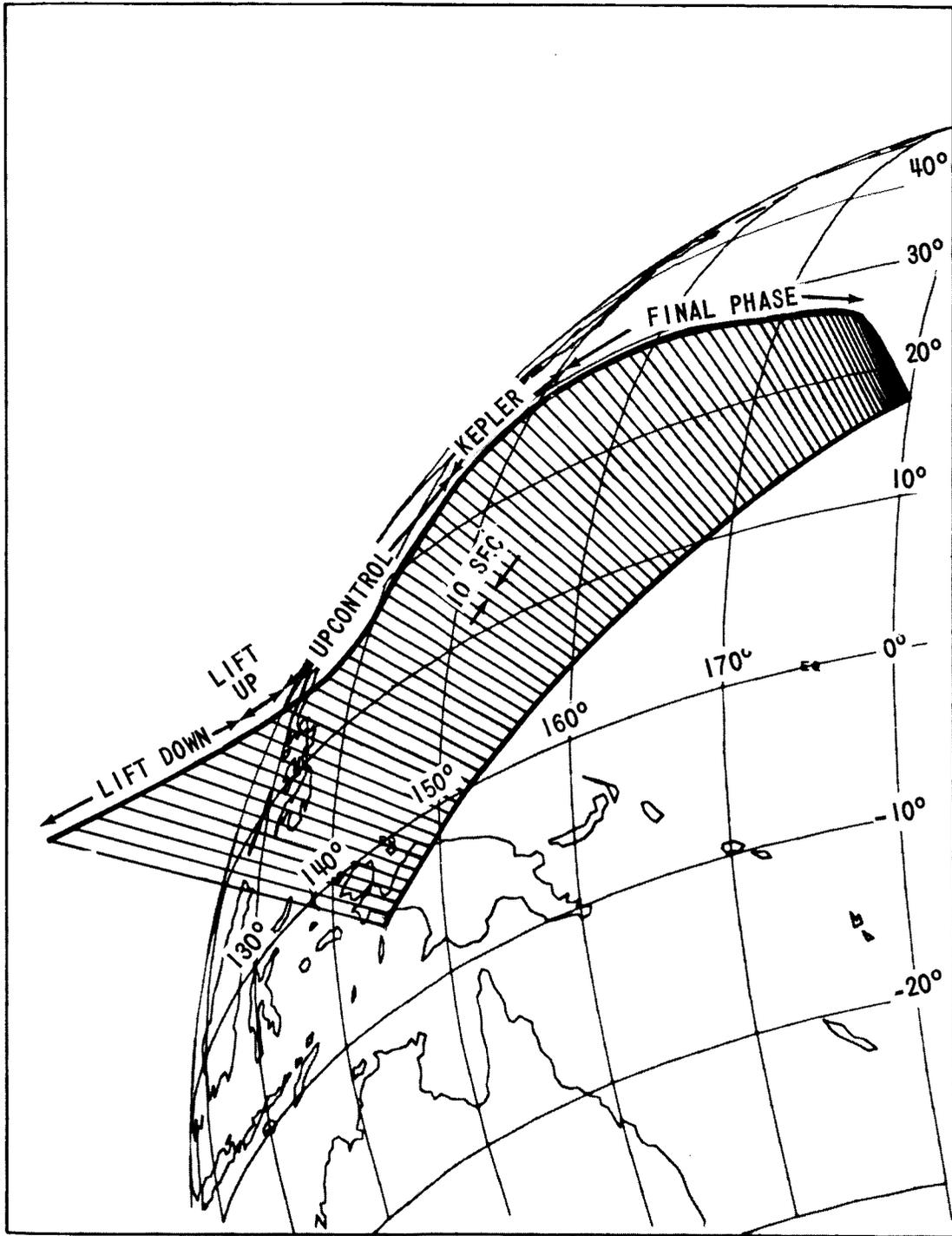
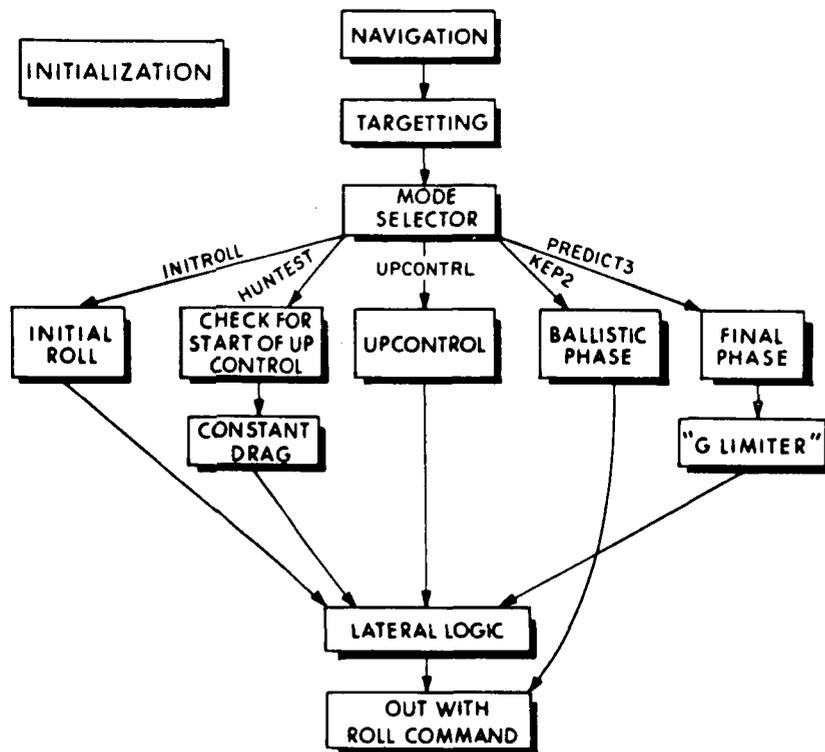


FIGURE 16 - AS-202 NOMINAL ENTRY TRAJECTORY
(ALTITUDE EXAGGERATED 20X)



TRAJECTORY CONDITIONS AT BEGINNING OF PHASES

PHASE	"ENTRY" ALTITUDE	INITIAL ROLL	CONSTANT DRAG	UPCONTROL		BALLISTIC	FINAL PHASE	25000 FT ALTITUDE
				PART 1	PART 2			
Time (relative), sec.	0	94.	NOT USED	128.	170.	318.	436.	850.3
Range to go nm	2639.	2194.		2034.	1843.	1243	791.	-0.53
Lateral miss nm	6.	1.		-3.	-14.	-4.4	-1.	0.01
Altitude Feet	399,188	263,870.		232407.	218337.	259,192	255,339.	25000.
Inertial velocity, fps	28,706.	28,755.		28262.	26709.	23,705.	23,165.	1496.
Flight path angle, deg	-3.57	-2.19		-1.41	0.01	0.51	-0.78	-16.4
Roll angle, deg	-165.	-15.		0.	-94.3	48.0	15.	121.7

FIGURE 17 - MIT ENTRY GUIDANCE - TRAJECTORY PHASES FLOW CHART

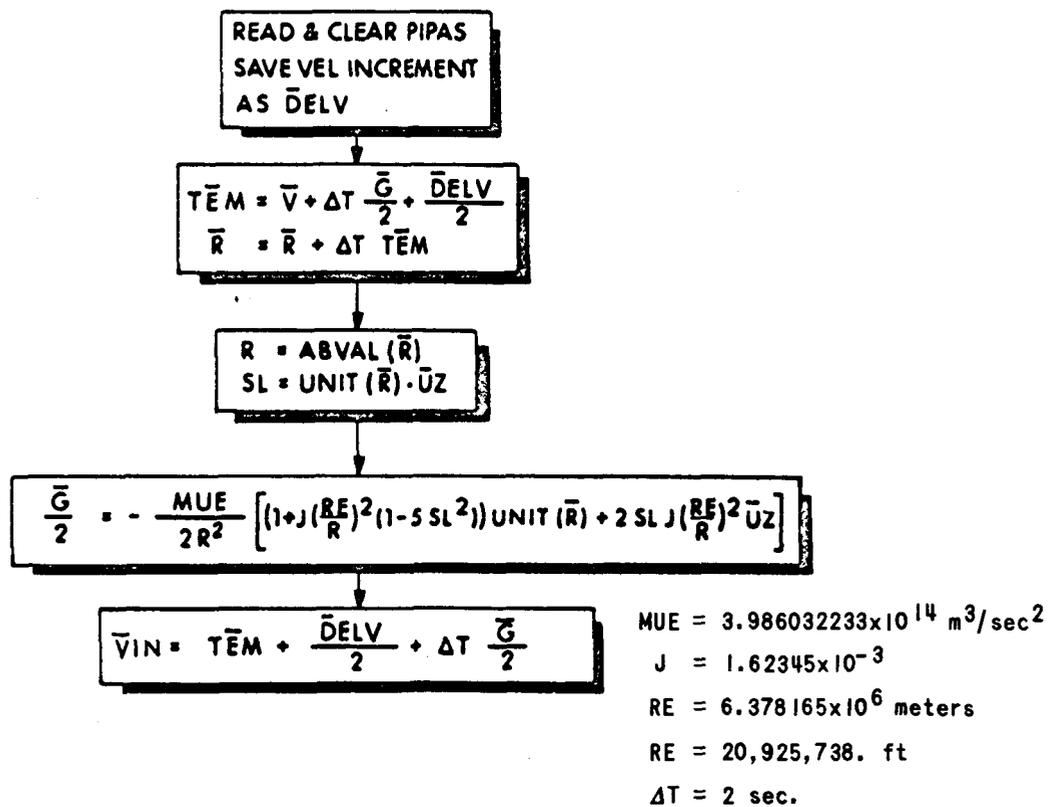


FIGURE 18 - MIT AVERAGE G NAVIGATION FLOW CHART.

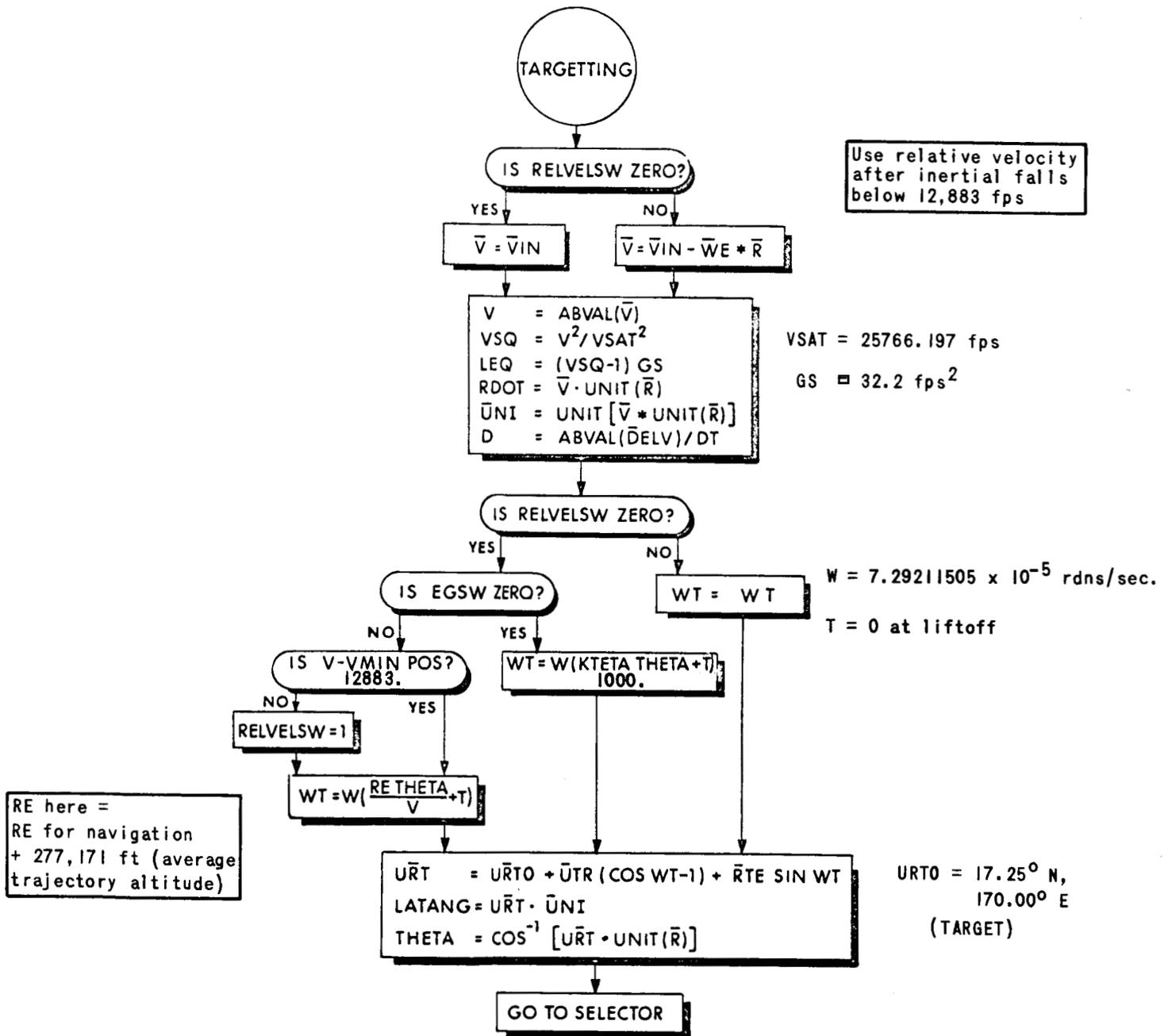


FIGURE 19 - MIT FLOW CHART - TARGETTING COMPUTATIONS

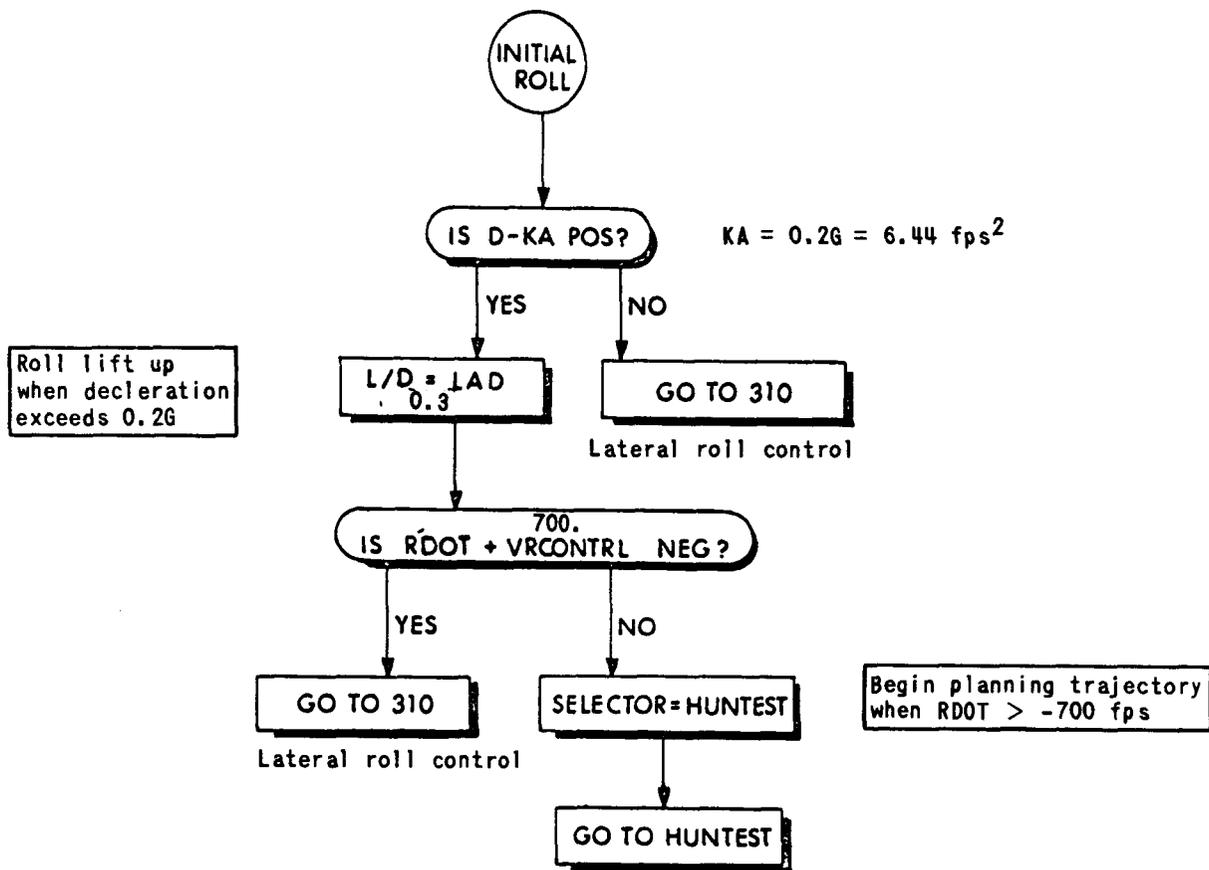


FIGURE 20 - MIT FLOW CHART - INITIAL ROLL PHASE

Lift up to get RDOT = 0 is not required. Plan trajectory using present V and D

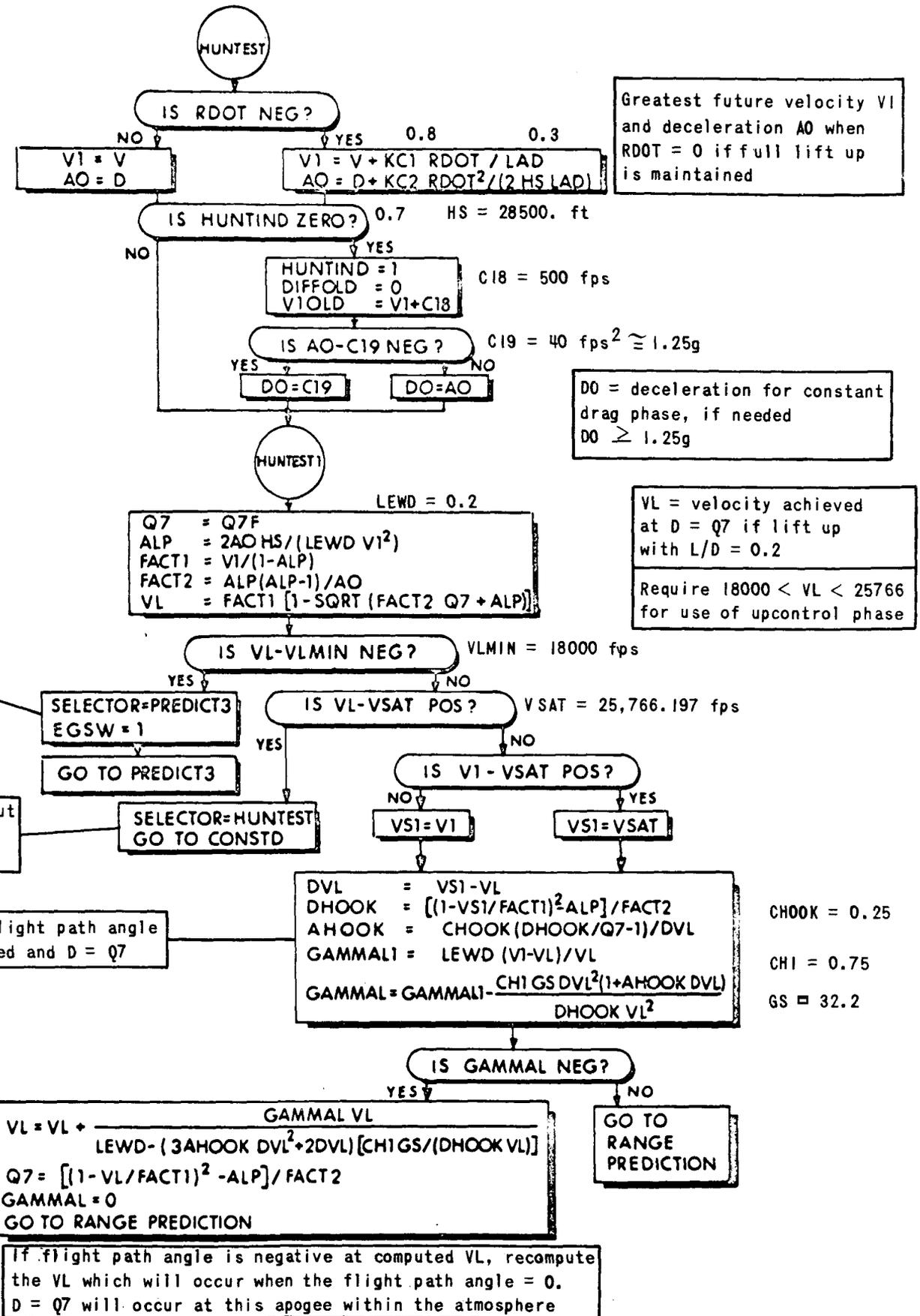
Computations give:
 $V1 = 26871 \text{ fps}$
 $A0 = 47.1 \text{ fps}^2$
 $D0 = 47.1 \text{ fps}^2$
 $Q7 = 6.0 \text{ fps}^2$
 $VL = 23887.$
 $GAMMAL = 0.910^\circ$

Trajectory already in final phase

VL too great: skip out likely so use CONSTD to slow down

GAMMAL = predicted flight path angle when $V = VL$ is reached and $D = Q7$

LEWD = 0.2
 CHI = 0.75
 GS = 32.2



Greatest future velocity V1 and deceleration A0 when RDOT = 0 if full lift up is maintained

D0 = deceleration for constant drag phase, if needed
 $D0 \geq 1.25g$

VL = velocity achieved at $D = Q7$ if lift up with $L/D = 0.2$
 Require $18000 < VL < 25766$ for use of upcontrol phase

CHOOK = 0.25
 CHI = 0.75
 GS = 32.2

FIGURE 21 - MIT FLOW CHART - HUNTEST COMPUTATIONS

RANGE PREDICTION

$$\begin{aligned}
 \text{VBARS} &= \text{VL}^2 / \text{VSAT}^2 \\
 \text{COSG} &= 1 - \text{GAMMAL}^2 / 2 \\
 \text{E} &= \text{SQRT} [1 + (\text{VBARS} - 2) \text{COSG}^2 \text{VBARS}]
 \end{aligned}$$

V SAT = 25766.197 fps

Actual sequence is
 ASPDWN to RDOT = 0,
 ASPUP TO V = VL,
 ASKEP TO D = Q7 +
 KDMIN AND ASPI
 And ASP3 for final
 phase with gamma
 correction

$$\begin{aligned}
 \text{ASKEP} &= 2\text{ATK} \sin^{-1}(\text{VBAR} \text{COSG} \text{GAMMAL}/\text{E}) && \text{BALLISTIC RANGE} \\
 \text{ASPI} &= \text{Q2} + \text{Q3 VL} && \text{FINAL PHASE RANGE} \\
 \text{ASPUP} &= \frac{\text{ATK}}{\text{RE}} (\text{HS}/\text{GAMMAL}) \log [\text{AOVL}^2 / (\text{Q7 V1}^2)] && \text{UPPHASE RANGE} \\
 \text{ASP3} &= \text{Q5} (\text{Q6} - \text{GAMMAL}) && \text{GAMMA CORRECTION} \\
 \text{ASPDWN} &= -\text{RDOT V ATK} / (\text{AO LAD RE}) && \text{RANGE TO PULLOUT} \\
 \text{ASP} &= \text{ASKEP} + \text{ASPI} + \text{ASPUP} + \text{ASP3} + \text{ASPDWN} && \text{TOTAL RANGE} \\
 \text{DIFF} &= \text{THETNM} - \text{ASP}
 \end{aligned}$$

ATK = 3437.7468 nm/rad
 Q2 = -1002 nm
 Q3 = 0.07 nm/fps
 RE = 21202909 ft
 HS = 28500 ft
 Q5 = 7050 nm
 Q6 = 0.0349 rad
 LAD = 0.3

IS ABS (DIFF) -25 NM NEG?

YES

SELECTOR=UPCONTROL

GO TO UPCONTROL

Begin upcontrol if
 miss < 25 nm and
 18000 < VL < 25766

Computations give:

E = 0.1414
 ASKEP = 664.6 nm
 ASPI = 668.1 nm
 ASPUP = 337.7 nm
 ASP3 = 134.0 nm
 ASPDWN = 225.05 nm
 ASP = 2029.5 nm
 THETNM = 2034.1 nm
 VCORR = -39. fps

NO

IS HIND=ZERO?

NO

IS DIFF NEG?

NO

VCORR=V1-VIOLD

VCORR = $\frac{\text{VCORR DIFF}}{(\text{DIFFOLD} - \text{DIFF})}$

YES

DIFFOLD=DIFF
 VIOLD=V1
 GO TO CONSTD

If an overshoot
 > 25 mi is predicted,
 slow down before
 beginning upcontrol

IS VCORR-VCORLIM POS?

VCORLIM = 1000.

YES

VCORR = VCORLIM

NO

IS VSAT-VL-VCORR NEG?

VSAT = 25766.197 fps

NO

VCORR = VCORR/2

V1 = V1 + VCORR
 HIND = 1
 DIFFOLD = DIFF

GO TO HUNTEST1

Refine trajectory
 prediction

FIGURE 22 - MIT FLOW CHART - RANGE PREDICTION COMPUTATIONS

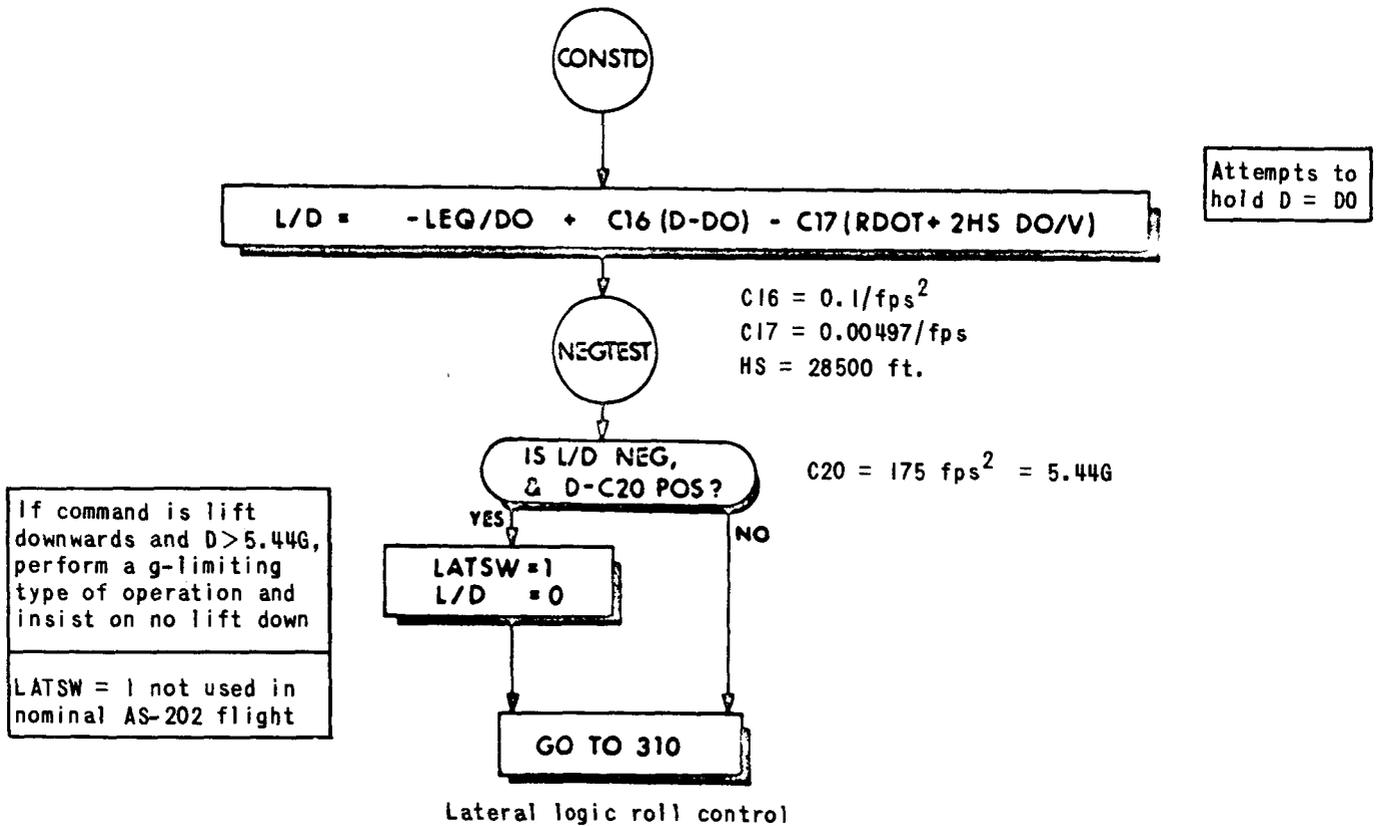


FIGURE 23 - MIT FLOW CHART-CONSTANT DRAG PHASE,

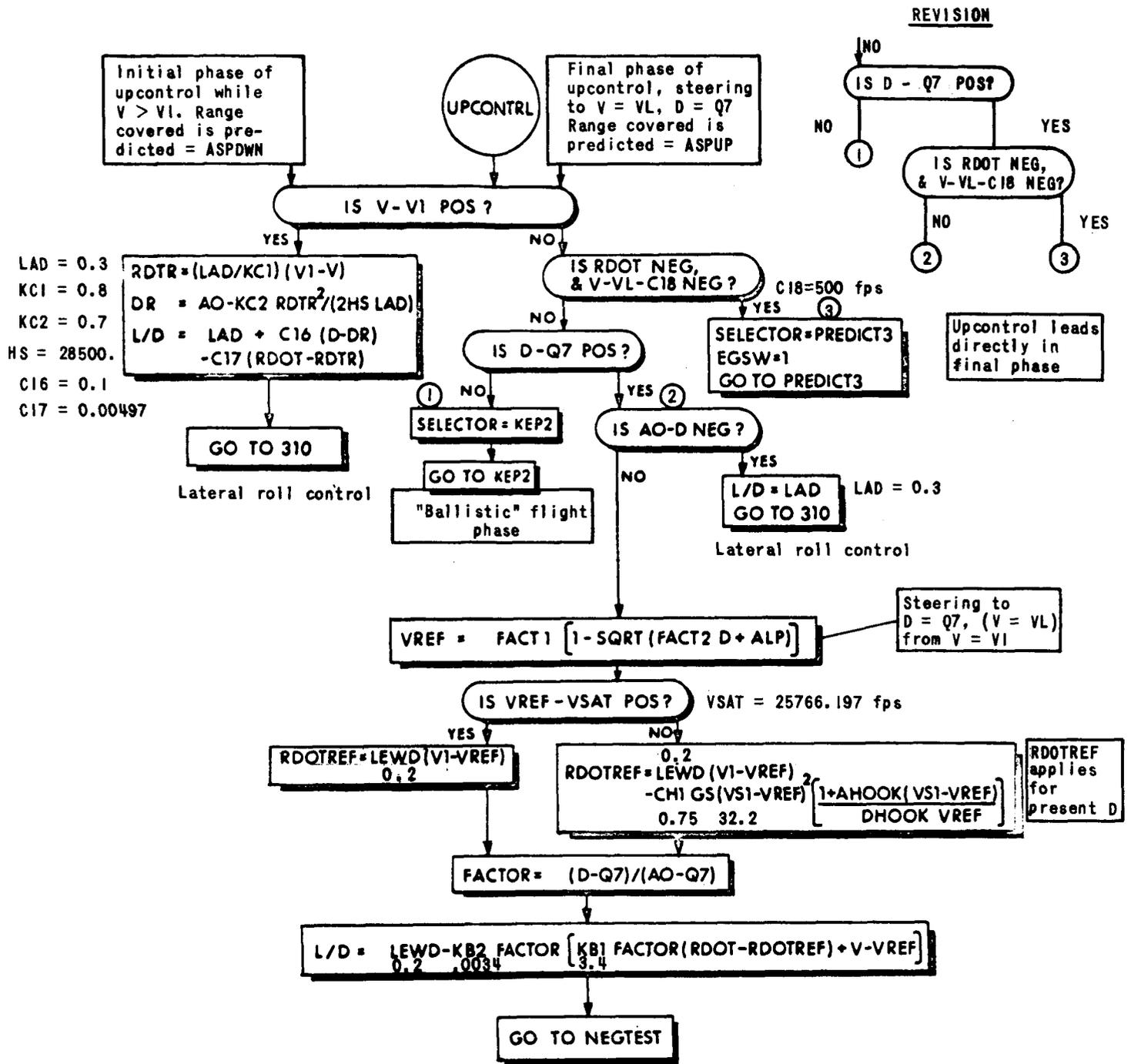


FIGURE 24 - MIT FLOW CHART - UPCONTROL PHASE

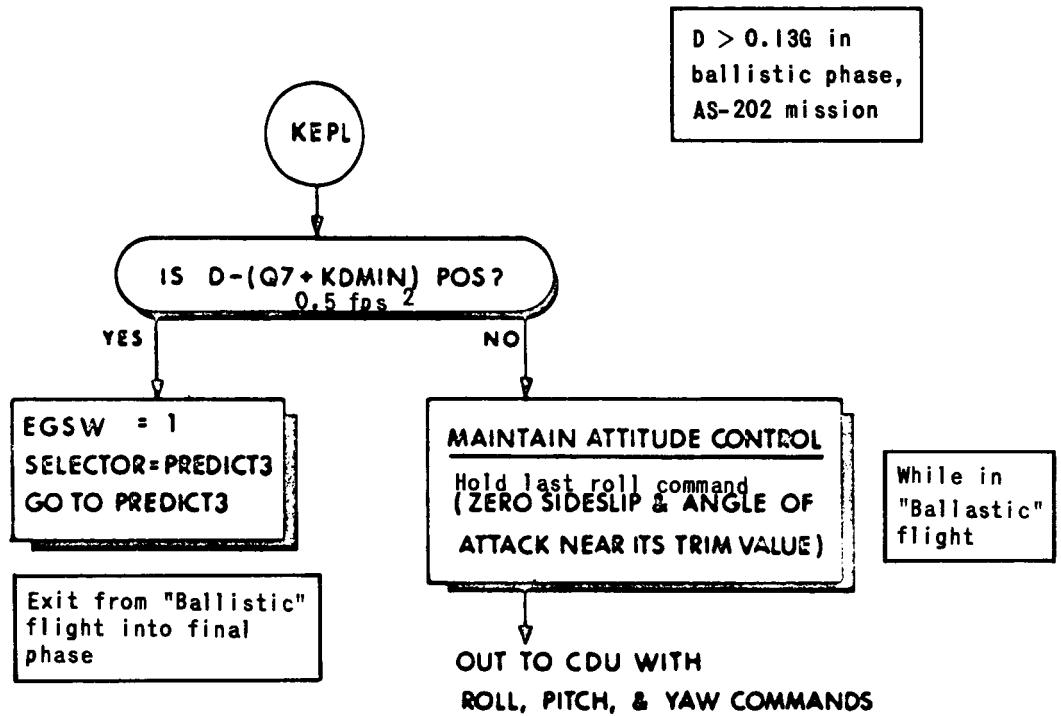


FIGURE 25 - MIT FLOW CHART - BALLISTIC FLIGHT PHASE

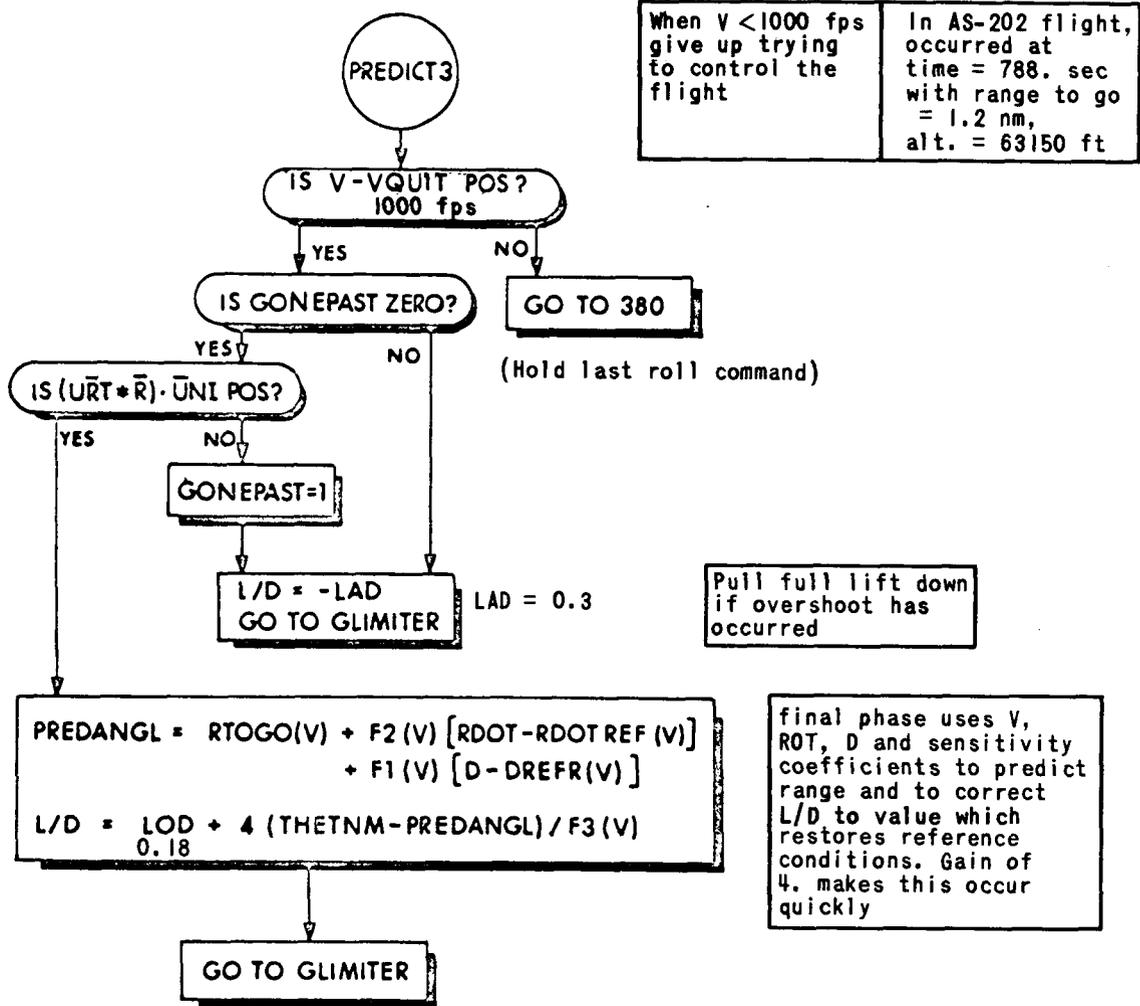


FIGURE 26 - MIT FLOW CHART - FINAL PHASE

VREF	RDOTREF	DREFR	DR/DRDOT	DR/DA	RTOGO	DR/DL/D
FPS	FPS	FPSS	F2 NM/FPSS	F1 NM/FPSS	NM	F3 NM
0	-331	34.1	0	-.02695	0	1
337	-331	34.1	0	-.02695	0	1
1080	-693	42.6	.002591	-.03629	2.7	6.44 x 2
2103	-719	60.	.003582	-.05551	8.9	10.91 x 2
3922	-694	81.5	.007039	-.09034	22.1	21.64 x 2
6295	-609	93.9	.01446	-.1410	46.3	48.35 x 2
8531	-493	98.5	.02479	-.1978	75.4	93.72 x 2
10101	-416	102.3	.03391	-.2372	99.9	141.1 x 2
14014	-352	118.7	.06139	-.3305	170.9	329.4
15951	-416	125.2	.07683	-.3605	210.3	465.5
18357	-566	120.4	.09982	-.4956	266.8	682.7
20829	-781	95.4	.1335	-.6483	344.3	980.5
23090	-927	28.1	.2175	-2.021	504.8	1385
23500	-820	6.4	.3046	-7.569	643.0	1508
35000	-820	6.4	.3046	-7.569	643.0	1508

FIGURE 27 - TABULAR DATA FOR FINAL PHASE COMPUTATIONS

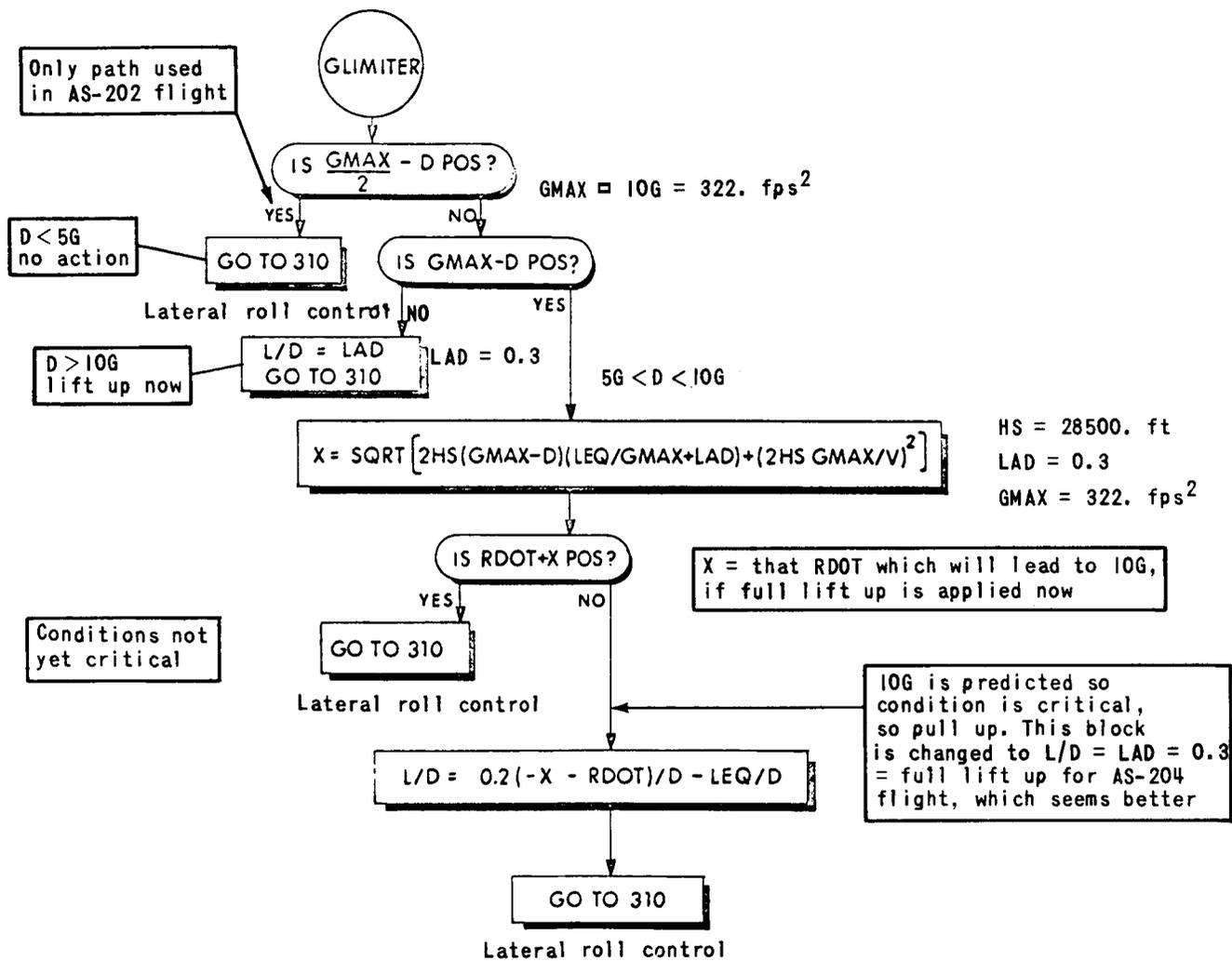


FIGURE 28 - MIT FLOW CHART - G LIMITER CONTROL

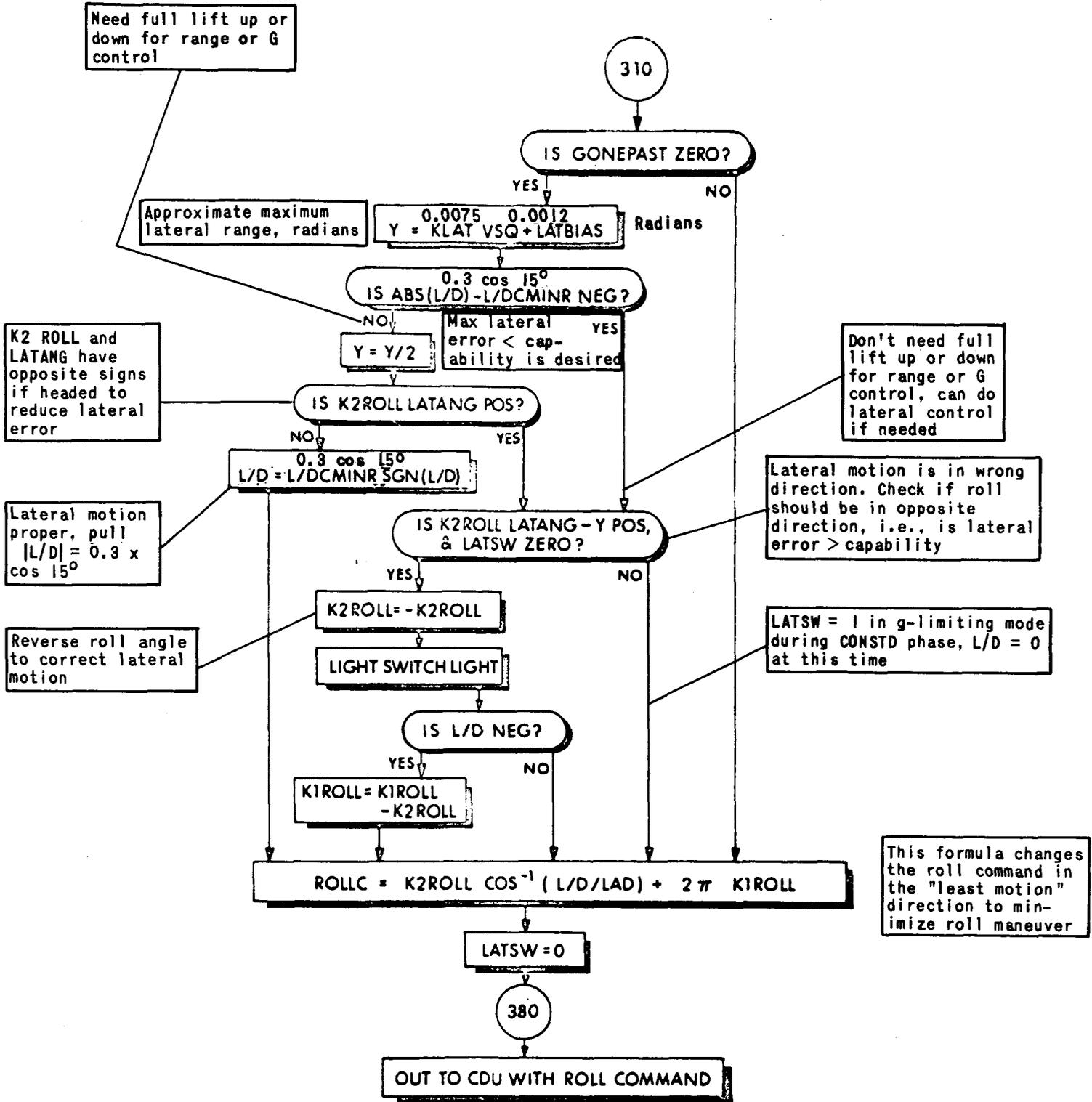


FIGURE 29 - MIT FLOW CHART - LATERAL LOGIC

FIGURE 30 - DEFINITIONS OF VARIABLES USED IN FLOW CHARTS

VARIABLES FOR RE-ENTRY CONTROL

$\bar{U}RTO$	INITIAL TARGET VECTOR (UNIT)
$\bar{U}Z$	UNIT VECTOR NORTH (NORTH POLE)
\bar{V}	VELOCITY VECTOR
\bar{R}	POSITION VECTOR
$\bar{R}TE$	VECTOR EAST AT INITIAL TARGET = $\bar{U}Z * \bar{R}$ (NOT UNIT)
$\bar{U}TR$	NORMAL TO $\bar{R}TE$ AND $\bar{U}Z = \bar{R}TE * \bar{U}Z$ (NOT UNIT)
$\bar{W}E$	EARTH RATE VECTOR
$\bar{U}RT$	TARGET VECTOR
$\bar{U}NI$	UNIT NORMAL TO TRAJECTORY PLANE
$\bar{D}ELV$	INTEGRATED ACCELERATION VECTOR
\bar{G}	GRAVITY VECTOR
AO	INITIAL DRAG FOR UPCONTRL
AHOOK	TERM IN GAMMAL COMPUTATION
ALP	CONST FOR UPCONTRL
ASKEP	KEPLER RANGE
ASP1	FINAL PHASE RANGE
ASPUP	UPRANGE
ASP3	GAMMA CORRECTION
ASPDWN	RANGE DOWN TO PULL-UP
ASP	PREDICTED RANGE = ASKEP+ASP1+ASPUP+ASP3+ASPDWN
COSG	COSINE (GAMMAL)
D	TOTAL ACCELERATION
DO	CONTROLLED CONST DRAG
DHOOK	TERM IN GAMMAL COMPUTATION
DIFF	THETNM-ASP (RANGE DIFFERENCE)
DIFFOLD	PREVIOUS VALUE OF DIFF
DR	REFERENCE DRAG FOR DOWNCONTROL
DREF	REFERENCE DRAG
DVL	VS1 -VL

FIGURE 30 - (Cont'd)

E	ECCENTRICITY	
F1	DRANGE/D DRAG	(FINAL PHASE)
F2	DRANGE/DRDOT	(FINAL PHASE)
F3	DRANGE/D(L/D)	(FINAL PHASE)
FACT1	CONST FOR UPCONTRL	
FACT2	CONST FOR UPCONTRL	
FACTOR	USED IN UPCONTRL	
GAMMAL	FLIGHT PATH ANGLE AT VL	
GAMMAL1	SIMPLE FORM OF GAMMAL	
KA	ACCELERATION LEVEL TO ROLL LIFT UP	
K1ROLL	INDICATOR FOR ROLL SWITCH	
K2ROLL	INDICATOR FOR ROLL SWITCH	
LATANG	LATERAL RANGE	
LEQ	EXCESS C. F. OVER GRAV = (VSQ-1) GS	
L/D	DESIRED LIFT TO DRAG RATIO (VERTICAL PLANE)	
PREDANGL	PREDICTED RANGE	(FINAL PHASE)
Q7	MINIMUM DRAG FOR UPCONTROL	
RDOT	ALTITUDE RATE	
RDOTREF	REFERENCE RDOT FOR UPCONTRL	
RDTR	REFERENCE RDOT FOR DOWNCONTRL	
ROLLC	ROLL COMMAND	
RTOGO	RANGE TO GO	(FINAL PHASE)
SL	SIN OF LATITUDE	
T	TIME	
THETA	DESIRED RANGE (RADIANS)	
THETNM	DESIRED RANGE (NM)	
V	VELOCITY MAGNITUDE	
V1	INITIAL VELOCITY FOR UPCONTRL	
VIOLD	PREVIOUS VALUE OF V1	

FIGURE 30 - (Cont'd)

VCORR	VELOCITY CORRECTION FOR UPCONTRL
VL	EXIT VELOCITY FOR UPCONTRL
VS1	VSAT OR V1, WHICHEVER IS SMALLER
VBAR S	$VL^2/VSAT^2$
VSQ	NORMALIZED VELOCITY SQUARED = $V^2/VSAT^2$
WT	EARTH RATE X TIME
X	INTERMEDIATE VARIABLE USED IN G LIMITER
Y	LATERAL MISS LIMIT

<u>SWITCHES</u>		<u>INITIAL STATE</u>
RELVELSW	RELATIVE VELOCITY SWITCH	(0)
EGSW	FINAL PHASE SWITCH	(0)
HUNTIND	INITIAL PASS THRU HUNTEST	(0)
HIND	INDICATES INTERATION IN HUNTEST	(0)
LATSW	NO LATERAL CONTROL WHEN ON	(0)
GONEPAST	INDICATES OVERSHOOT OF TARGET	(0)

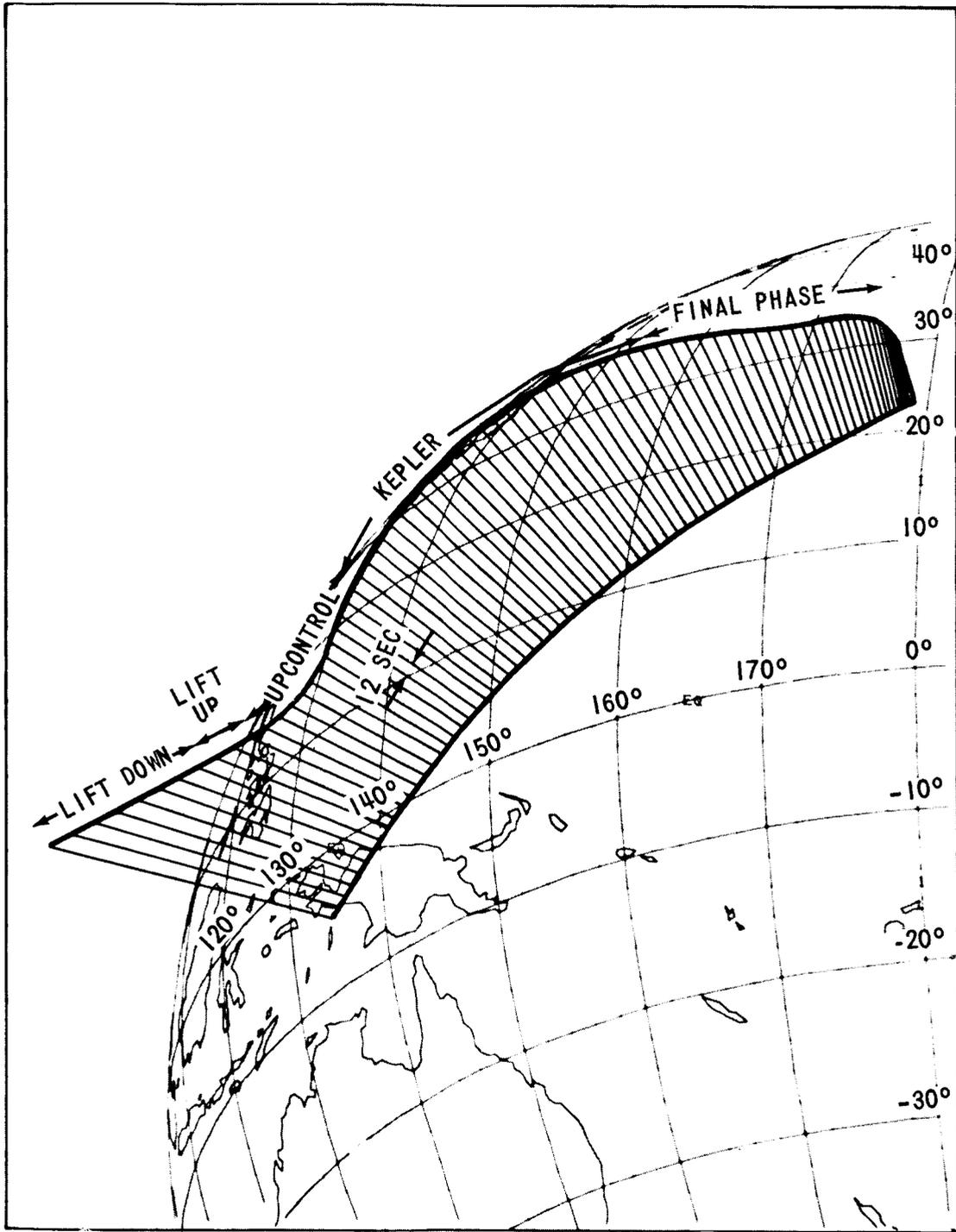


FIGURE 31 • A 3282 NM ENTRY TRAJECTORY
(ALTITUDE EXAGGERATED 20X)

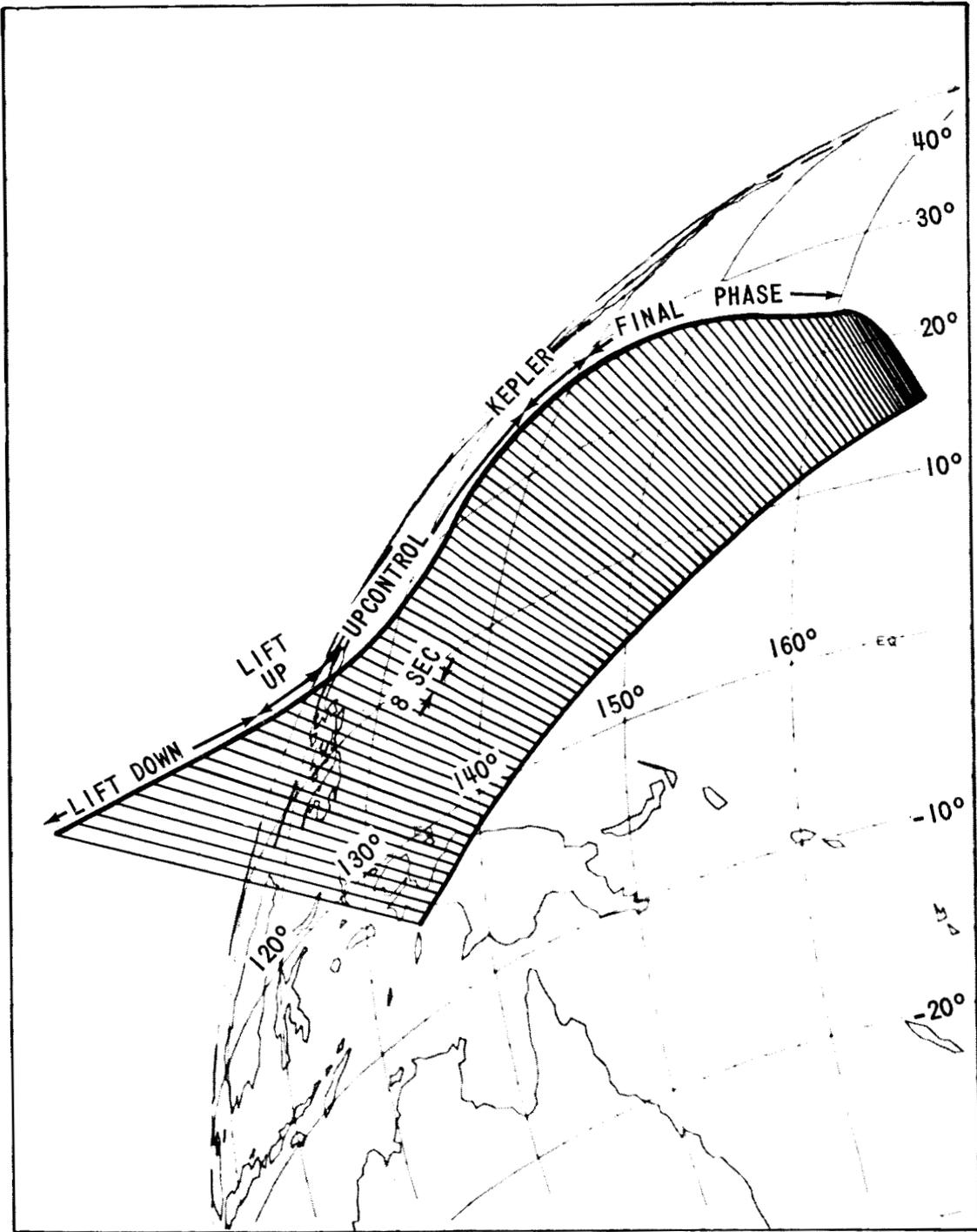


FIGURE 32 - A 2380 NM ENTRY TRAJECTORY
(ALTITUDE EXAGGERATED 20X)

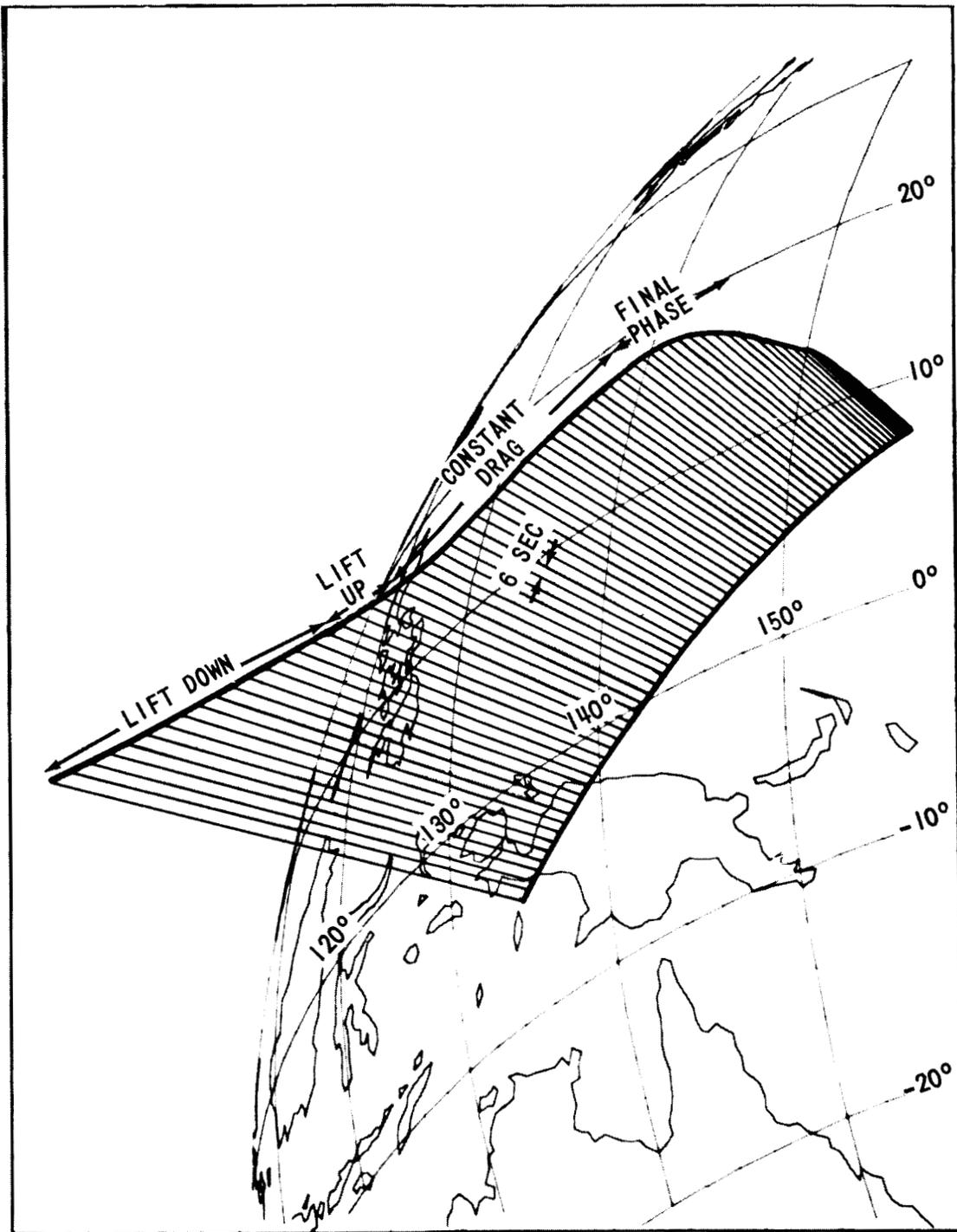


FIGURE 33 - A 1553 NM ENTRY TRAJECTORY
(ALTITUDE EXAGGERATED 20X)

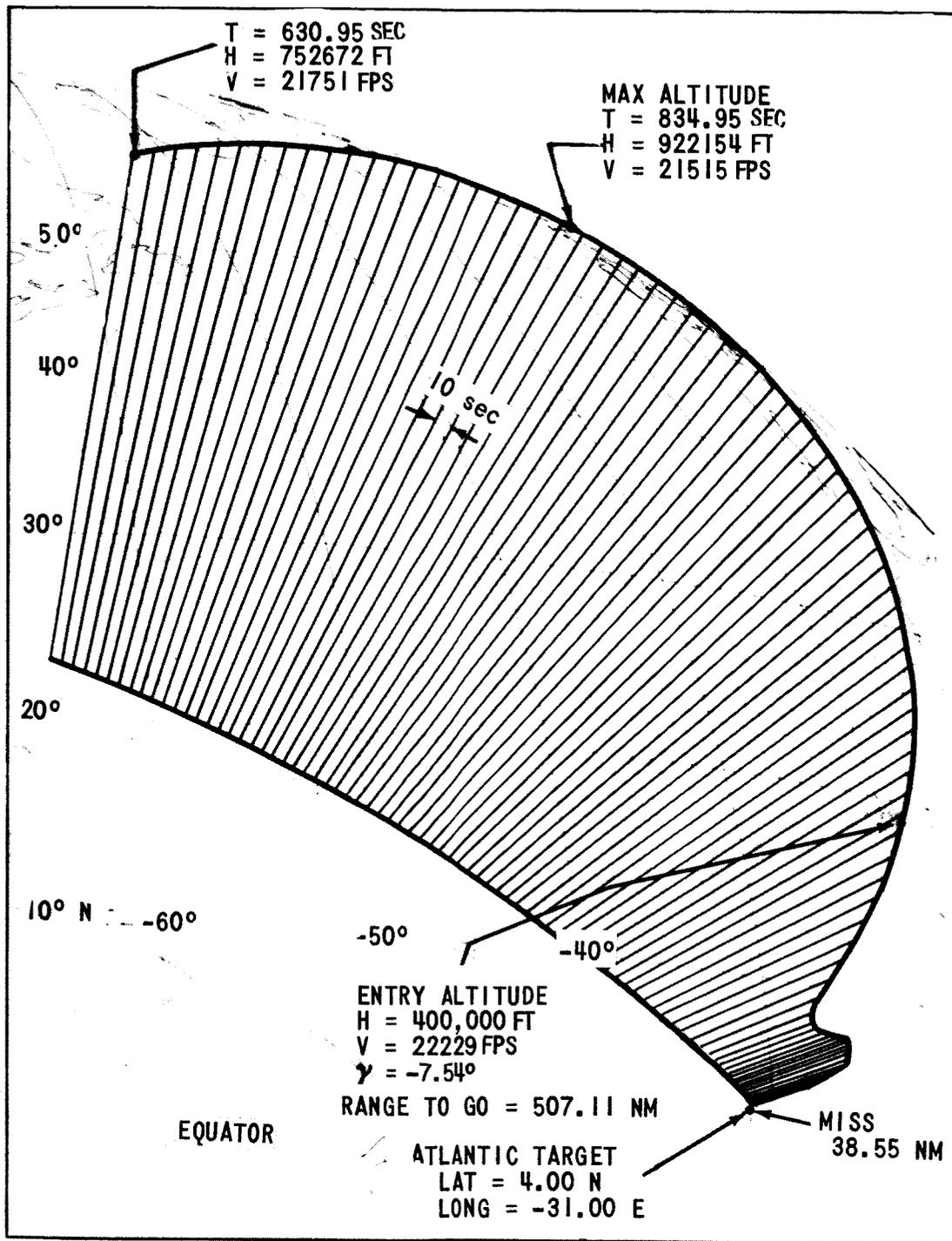


FIGURE 34 - TRAJECTORY AFTER FIRST SPS BURN FAILS
 (ALTITUDE EXAGGERATED 20X)

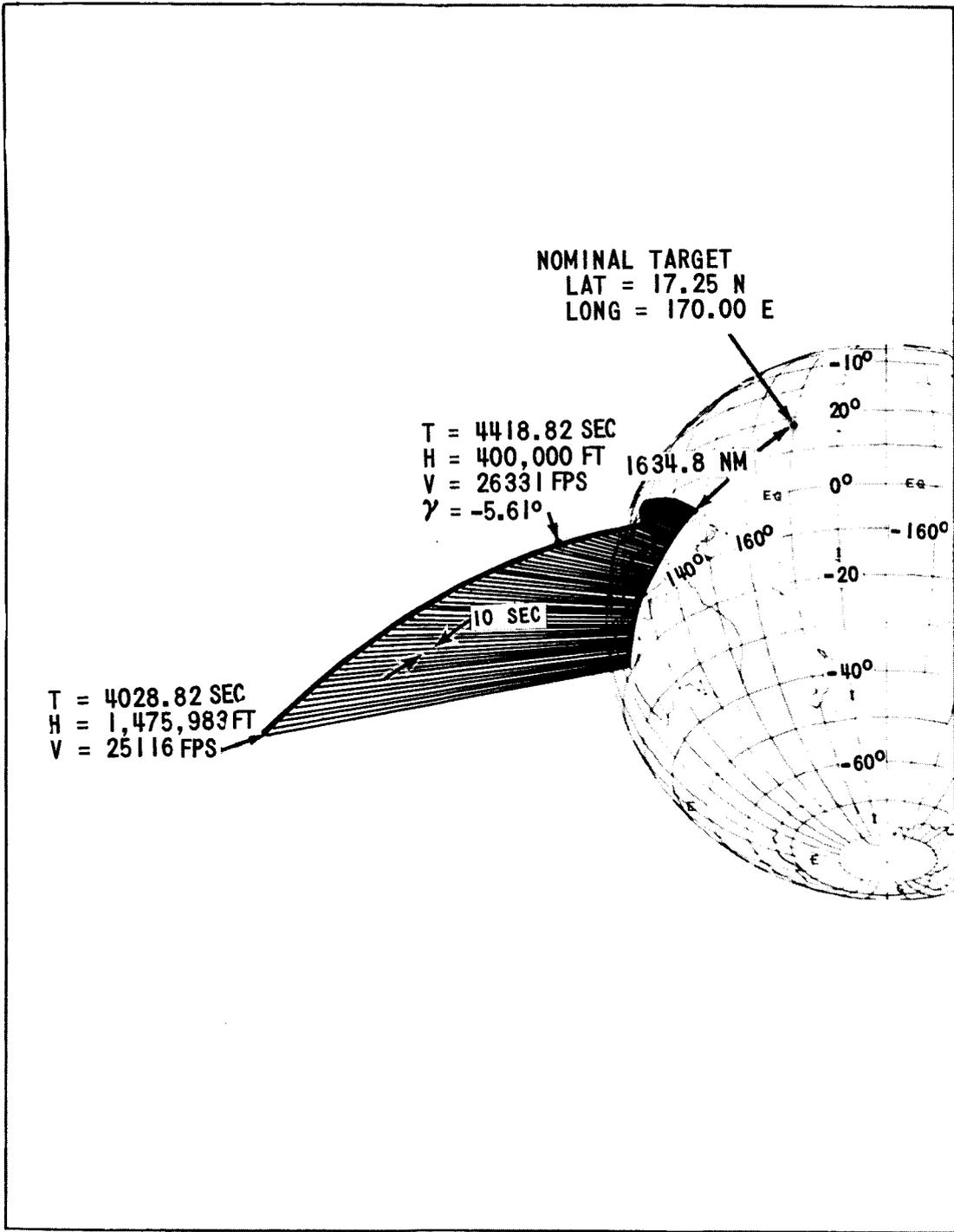


FIGURE 35 - TRAJECTORY AFTER SECOND SPS BURN FAILS
 (ALTITUDE EXAGGERATED 20X)