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A MANUAL LEM BACK-UP GUIDANCE SYSTEM

by

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INSTRUMENTATION LABORATORY

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COPY # 1

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ABSTRACT

This report outlines a manual back-up guidance system for LEM abort to rendezvous from any point in the powered descent or ascent phase, from subsequent transfer and rendezvous phases, or from the lunar surface.

Powered ascent maneuvers are implemented with reference to on-board steering displays. Subsequent transfer and rendezvous maneuvers require steering data obtained through use of the tracking radar and primary G&N system on the CSM.

A clear pericynthion is not obtained until after the transfer maneuver. Relinquishing the requirement for an initially clear pericynthion allows implementation of a more efficient ascent profile, thus permitting low accuracy systems to live within the LEM characteristic velocity budget.

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INTRODUCTION

A manual back-up guidance system for the LEM is outlined. Its function is to provide guidance for a LEM abort from any point in the powered descent or ascent phase, from subsequent transfer and rendezvous phases, or from the lunar surface in the event of primary LEM G&N failure. Such abort involves initial powered ascent to non-clear pericynthion cut-off conditions, transfer (including midcourse corrections) on a clear pericynthion trajectory to LEM/CSM intersection within the first orbit, and rendezvous, canceling relative vehicle motion.

The suggested system offers simplicity, light weight, high reliability, identical procedure regardless of time of abort, complete independence from the LEM primary G&N system, and low development risks. The latter may be important in light of relatively tight development schedules. The low system accuracy accompanying the above advantages results in large position and velocity uncertainties at ascent cut-off. Uncertainties in altitude rate (flight path angle) and total velocity are most critical and require adherence to the following ground rules:

> The nominal flight path angle at ascent cut-off must be biased outward, assuring that the non-clear pericynthion is well ahead of the LEM.
>
> Aborts occurring while the LEM and CSM can maintain line-of-sight contact will depend on tracking by the CSM after ascent cut-off for velocity correction information to obtain a clear pericynthion. The same correction will place the LEM on the proper trajectory for transfer to the CSM orbit. Rendezvous will subsequently be com-

pleted with the aid of tracking by the CSM.

For aborts when CSM tracking is not immediately available, a clear pericynthion will be guaranteed by applying a predetermined velocity correction soon after ascent cut-off. Transfer and rendezvous will subsequently be completed with the aid of tracking by the CSM.

The nominal ascent cut-off velocity may safely be sub-orbital because of the positive flight path angle at cut-off and the required subsequent velocity correction.

SYSTEM DESCRIPTION

Powered ascent is accomplished by thrusting at two predetermined inertially fixed angles. The same two angles are always used regardless of the time of abort. The only variable steering parameters are the time at which the vehicle is reoriented to the second thrust vector angle and time of engine cut-off. A Timing Display indicates these two variable "reference times" along with the clock time from initiation of powered descent. The remainder of the system consists of an Attitude Reference to obtain proper thrust vector angles.

Transfer and Rendezvous maneuvers rely on steering parameters obtained from the CSM (Reference I). These velocity corrections would be implemented by the clock and attitude reference mentioned above.

Timing Displays

A description of the operation of the Timing Display illustrates the abort procedure. At the beginning of the powered descent phase the "Time From Event" digital display is started (Fig. 1). Adjacent to this clock and driven directly by its shaft are two sets of mechanical counter time displays representing two sets of variable reference times. The first set would be utilized if both the descent and ascent stages were operable, while the second set would be referred to if it were necessary to abort with only the ascent stage (the latter would be necessary during descent if the descent stage failed or during ascent). Each set has its own associated thrust vector angles. When an abort signal is received, the "Time From Event" clock continues to operate while the rest of the display is disengaged, thus fixing the two sets of reference times. The astronaut's first actions would be to refer to the proper display set, and to orient the vehicle to the first of the two predetermined thrust vector angles. This angle is maintained until the first reference time is read on the "Time From Event" clock. The vehicle is then reoriented to the second thrust vector angle, which is maintained until cut-off at the second reference time. Aborts occurring within a few seconds of initiation of powered descent would require only the transfer and rendezvous phases. Further down the powered descent trajectory a short period exists where aborts would utilize only the first thrust vector angle during the ascent phase. Figure 3 illustrates these early abort situations.

Design of the Timing Display requires that two acceptable thrust vector angles be determined along with the relationships governing the variable reference times. Such an analysis was performed for the case where only the ascent stage is operable. A particular position and velocity vector must be obtained at ascent cut-off. The planar case was considered and range was relinquished as a constraint at cut-off. Three constraints remained; altitude, tangential velocity, and radial velocity. Four degrees of freedom are available to satisfy these constraints (two thrust vector angles and two reference times), suggesting a family of possible solutions. A family of solutions was obtained for the fuel-critical abort from hover case. The solution requiring least fuel provided a specification of the two constant thrust vector angles. Having fixed these two degrees of freedom, only the two reference times remain as variables for obtaining satisfactory ascent cut-off conditions for aborts from earlier portions of the descent phase or from the ensuing ascent phase. One of the three end condition constraints, altitude, was then relinquished in order to allow solution of the problem. Fortunately, the resultant altitude variations, as a function of time of abort, were not excessive. Steering, therefore, involves controlling the cut-off velocity vector by the two reference times. These times were found to be piecewise linear with time from initiation of powered descent (Fig. 3). The intersection of the two linear portions occurs at the time in the descent phase that vehicle reorientation and engine throttling are required, suggesting that these actions caused the linear interruption. A third linear portion is expected to be necessary for aborts during nominal ascent, though the analysis was not extended to this region.

The above has yielded the relationships necessary for design of one set of mechanical counters on the Timing Display. A similar analysis for the case where both ascent and descent stages are operable will yield the design information for the second set. The digital computer program written for the above analysis can be utilized for the remaining analyses.

A more accurate scheme, requiring the addition of an integrating accelerometer along the thrust axis, would utilize the accelerometer outputs rather than time in this display. (System #2, Table IV).

Attitude Reference

A variety of attitude references could be utilized with the above Timing Display. The simplicity and expected light weight of the Timing Display suggest that overall system weight and simplicity will depend on the choice of the attitude reference. Therefore, a crude two degree of freedom gyro with the spin axis aligned near the nominal ascent cut-off velocity vector was selected. Investigation of commercial availability indicates that a limited development program could provide an instrument with the following characteristics:

1) Less than 10^{o}/ hour equivalent total drift rate
2) Less than 5 pounds total weight
3) 360^{o} outer gimbal, $\pm$ 80^{o} inner gimbal freedom
4) Direct mechanical display of outer and inner gimbal angle, though indirect displays may be more convenient.
5) High reliability figures quoted by industry require further investigation.

(References having accuracies greater than 3^o to 4^o/hr equivalent total drift rate deserve an automatic steering loop rather than the manual technique described above. With such a loop, and the more accurate attitude reference, the simplified steering law suggested above could provide a clear pericynthion.)

Lack of spin axis attitude, though least important, requires some additional reference. This could be a simple reticule on the windows for alignment with the lunar horizon ("wings level"). The instrument could be initially aligned prior to powered descent with respect to the spacecraft. Subsequent re-alignments for transfer and rendezvous maneuvers would rely on celestial references, perhaps utilizing the Optical Alignment Telescope (OAT) and/or window reticules. A quick caging and re-alignment might be necessary if the primary and back-up references reached gimbal lock simultaneously. This can happen early in powered descent, though sufficient fuel is available at that time to compensate for the inaccurate ascent cut-off that might result. At hover the possibility of the two systems gimbal locking simultaneously is remote.

A more complicated "all attitude" scheme could employ two of these gyros. The spin axes of each would be offset sufficiently to reduce the possibility of both reaching gimbal lock simultaneously. In addition, this dual gyro arrangement could, with proper resolution, provide three degrees of attitude indication. Moreover, the LEM would have available three independent attitude references allowing majority voting for system monitoring.

A less accurate reference might be a simple annular reticule pattern scribed on the LEM window. The reticule could be utilized as follows: two or more stars, located within a few degrees of the pole of the descent/ascent plane, would be preselected as references. Maintenance of the correct stars in their respective annuli constrains the vehicle to pitch motion in the proper plane. Further positioning of the stars along the circumference of their respective annuli provides pitch attitude. These reference stars could be located during unpowered descent from the CSM orbit. During powered descent the stars could be more easily detected in that they should appear within their proper annuli if descent is correct. Moreover, their position along the annuli should properly vary with time. This suggests that the annular reticule pattern might also serve as a simple visual system monitor. Regardless of the attitude reference selected, it could be utilized in the same way for systems monitoring. Both the attitude (pitch) and time at which the vehicle is nominally supposed to be at that attitude could be keyed together on the same display. Monitoring simply involves glancing at the "Time From Event" clock and the "keyed" time indication simultaneously. They should agree (Fig. 2).

SYSTEM PERFORMANCE

Tables I and II summarize the system uncertainties at ascent cut-off for an abort after sixty seconds of hovering and an abort from the surface. The large uncertainty in flight path angle at ascent cut-off precludes the possibility of obtaining a clear pericynthion without a subsequent velocity correction. (For an abort from hover Table I indicates an altitude rate uncertainty of 183 ft/sec, corresponding to a flight path angle uncertainty of two degrees.) The nominal flight path angle at ascent cut-off must be biased outward, assuring that the non-clear pericynthion is well ahead of the LEM. Fortunately, for an abort from hover where rendezvous within one orbit is desired, an ascent profile with a positive flight path angle at cut-off is <u>more</u> efficient than the corresponding clear pericynthion profile. It "lofts" the vehicle to an apolune where the transfer velocity correction can be more efficiently applied, and allows the nominal ascent cut-off velocity to be sub-orbital. The increased efficiency of this flight profile permits this low accuracy system to live within the LEM characteristic velocity budget. (See Figs. 4 through 7).

The requirement for a velocity correction subsequent to ascent cut-off to obtain a clear pericynthion imposes certain operational restrictions. Figures 4 through 7 summarize these considerations for cases where the velocity corrections (transfer maneuver) take place ten, twenty, and thirty minutes after ascent cut-off. Ten minutes is a lower limit because of the requirement for smoothing the CSM tracking radar data, while time intervals greater than thirty minutes require more accurate ascent cut-off control and/or more characteristic velocity. (A clear pericynthion represents the upper limit.)

Table III delineates the envelope of ascent cut-off conditions selected for study and indicates the "worst case" situations for two suggested systems. System acceptability is gaged on ability to live within the LEM characteristic velocity budget for these "worst case" conditions. A breakdown of the characteristic velocity required by each system is given on Table IV. In all cases characteristic velocity figures reflect correction for 2° out of plane error at ascent cut-off. Both systems utilize an attitude reference having a drift rate of 10 deg/hr. System #1 steers on the basis of the two reference times previously described, while System #2 replaces time with integrated thrust acceleration.

CONCLUSIONS

The study results indicate that a simple manual abort guidance system could provide LEM back-up capability within the present LEM fuel budget. Its light weight and low development risk are particularly attractive and suggest that it might serve as a third tier back-up system.

Relinquishing the requirement for an initially clear pericynthion allows implementation of a more efficient ascent profile, thus permitting this low accuracy system to live within the LEM characteristic velocity budget.

The study also indicates that the steering law suggested could be utilized along with a more accurate attitude reference and automatic steering loop to obtain a clear pericynthion.

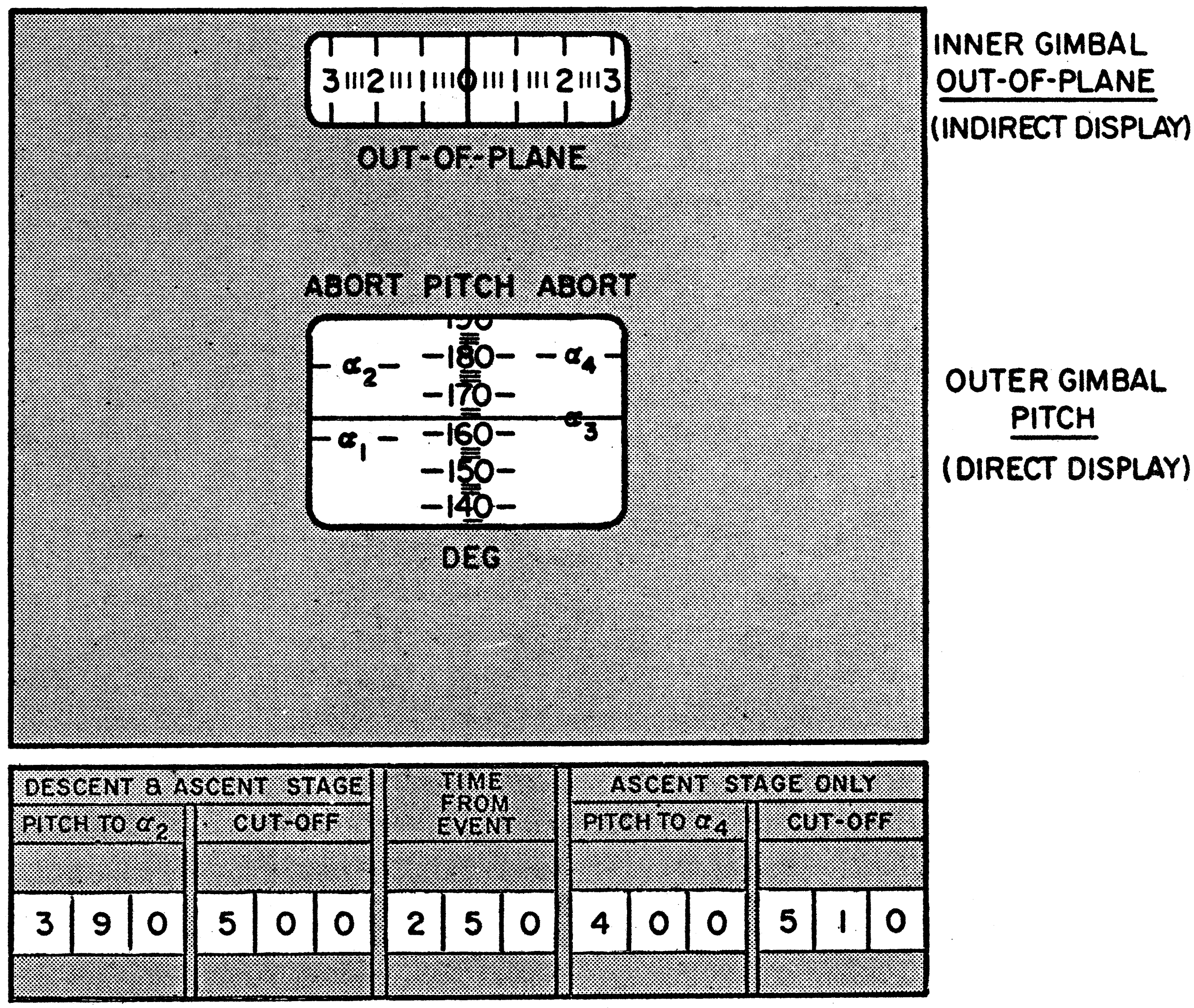

Fig. 1 System display.

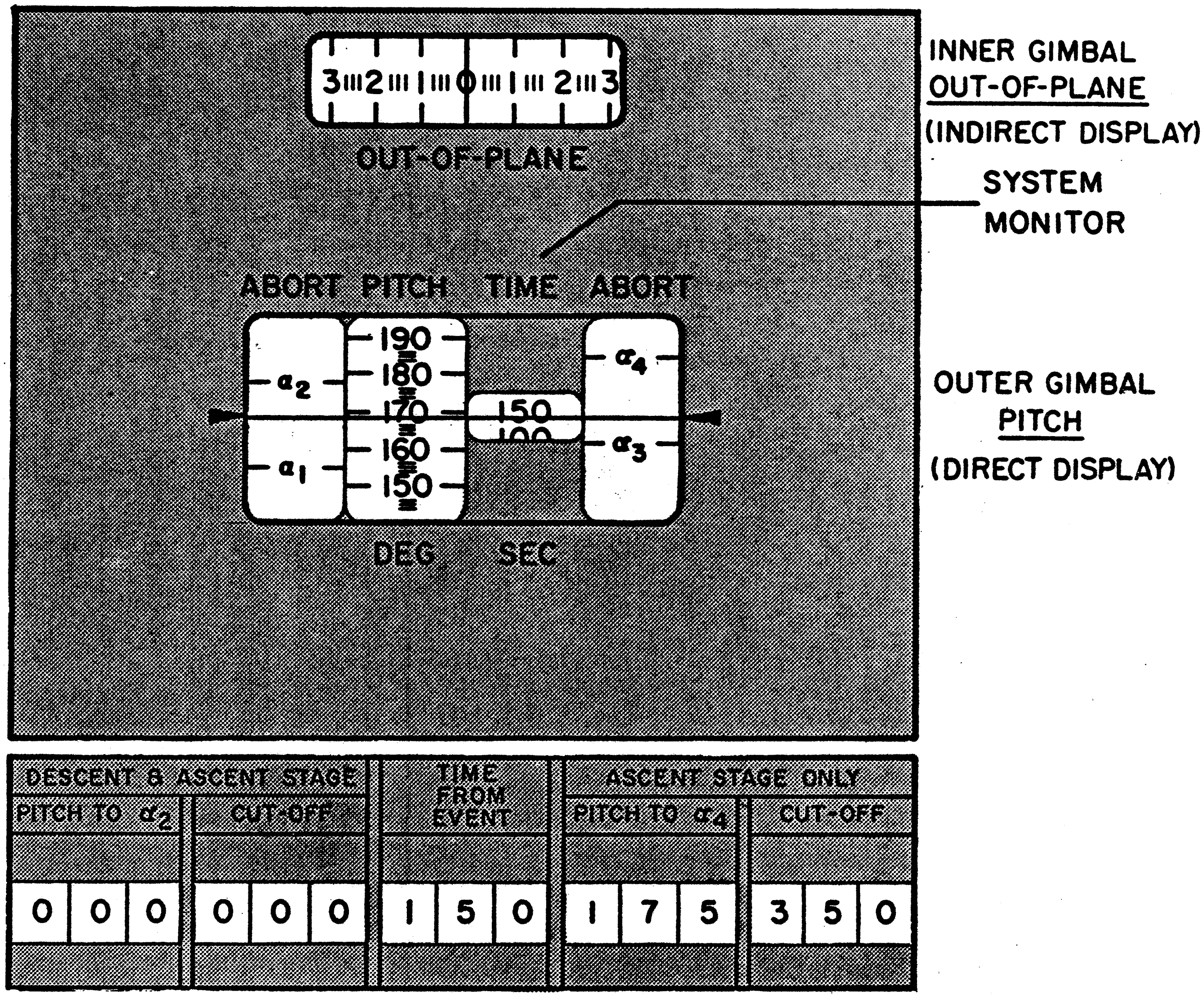

Fig. 2 System display.

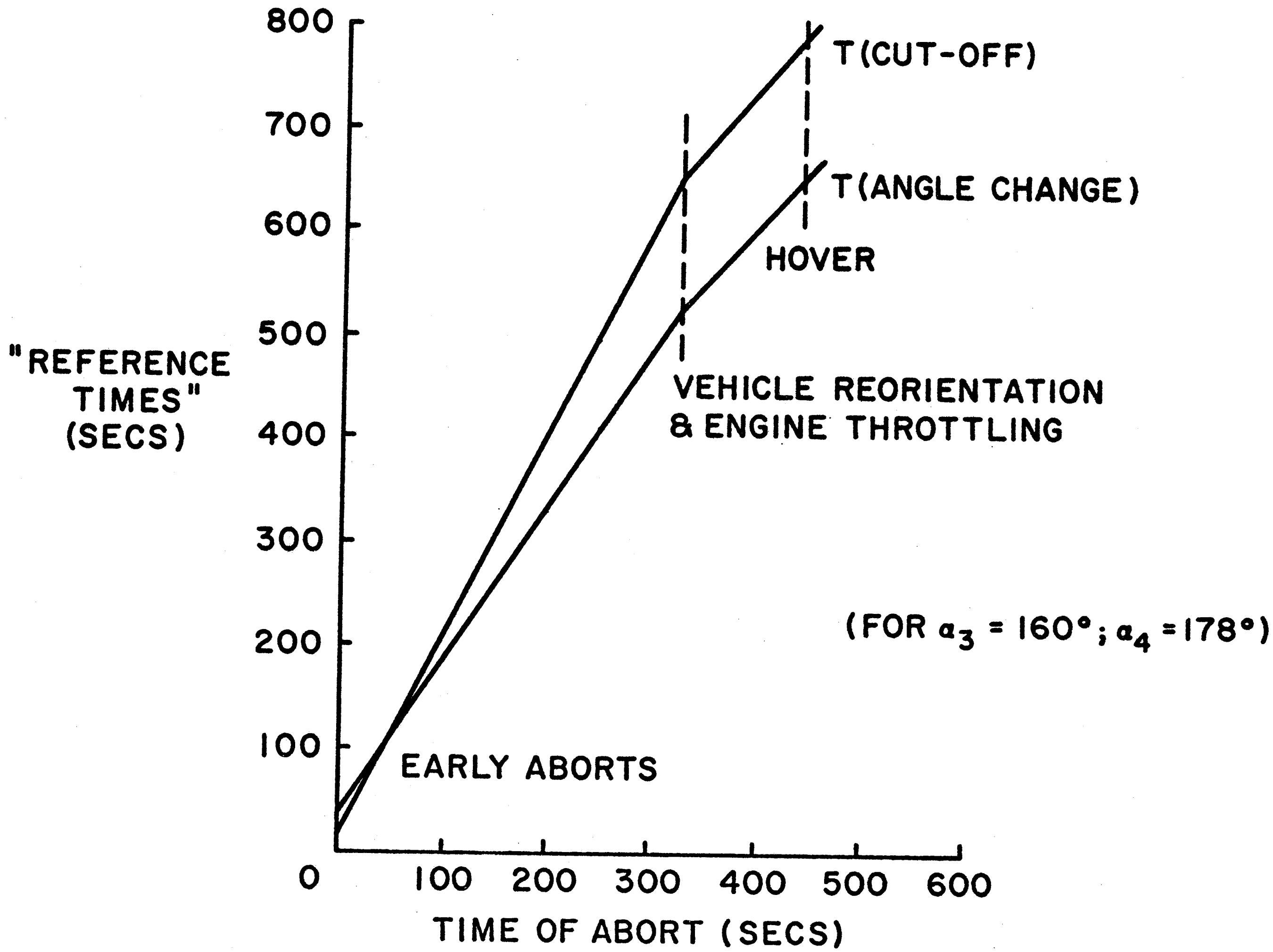

Fig. 3 Steering parameter time relationships.

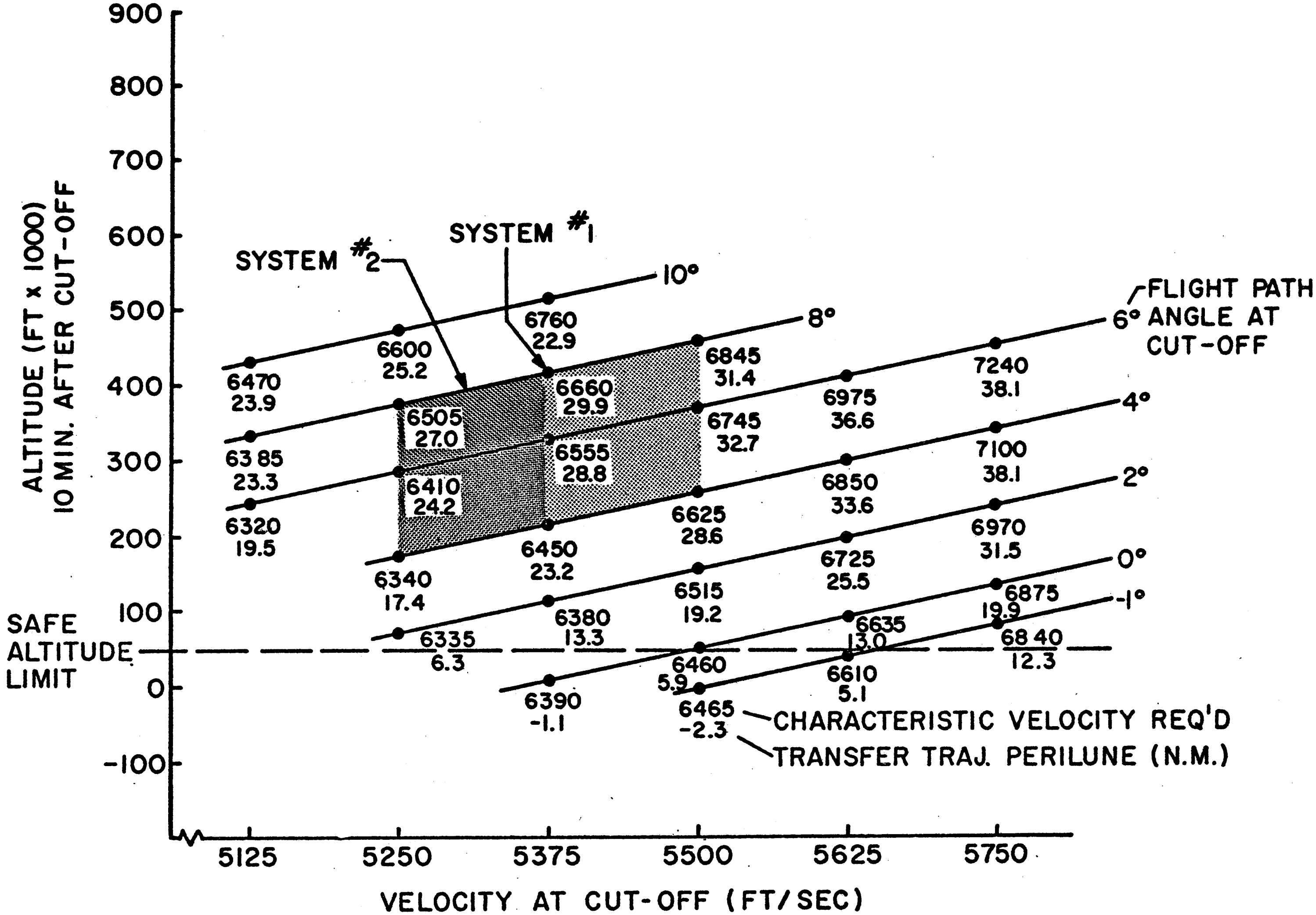

Fig. 4 Operational envelope-abort from hover-transfer velocity correction occurs 10 min after cutoff.

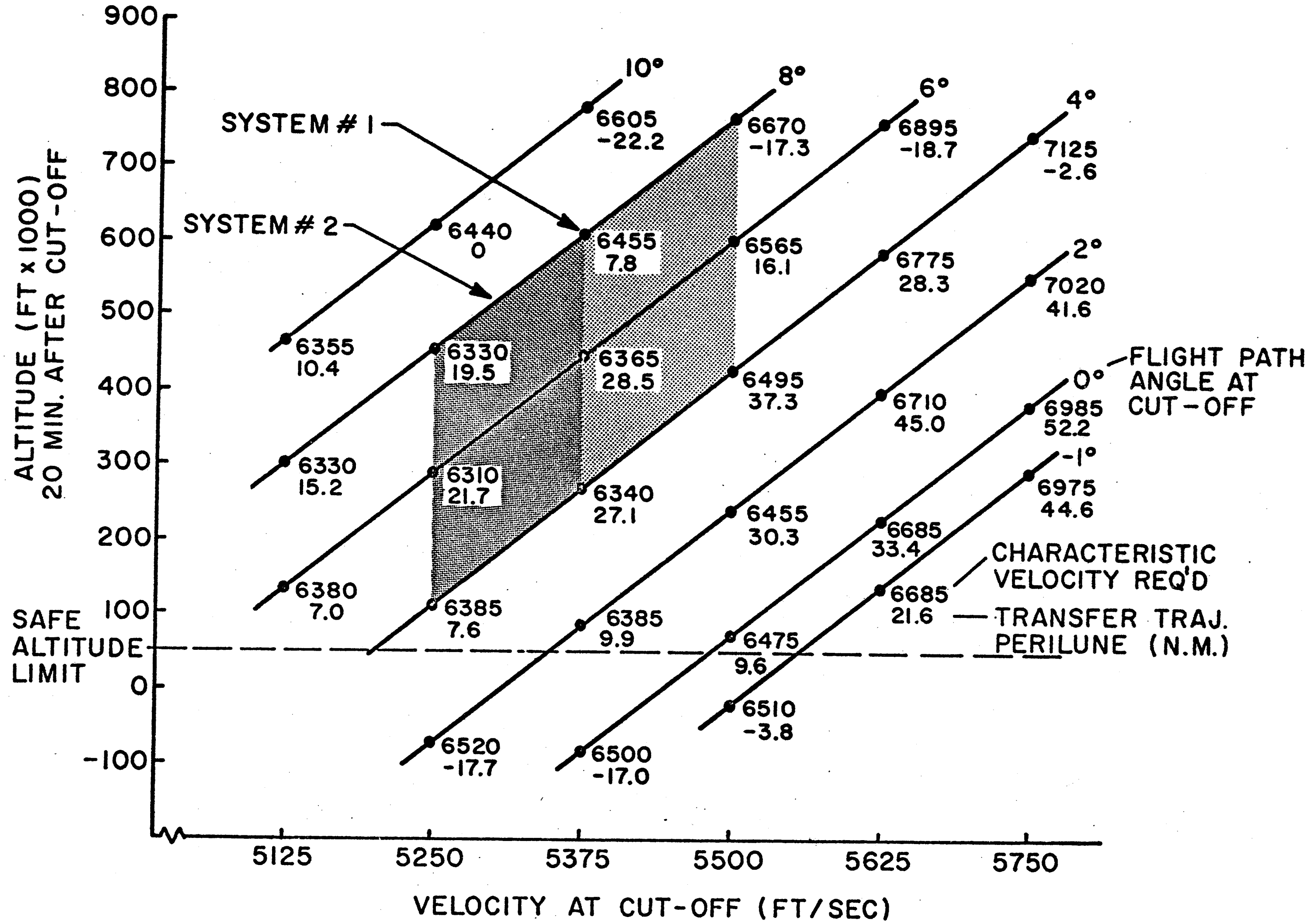

Fig. 5 Operational envelope-abort from hover-transfer velocity correction occurs 20 min after cutoff.

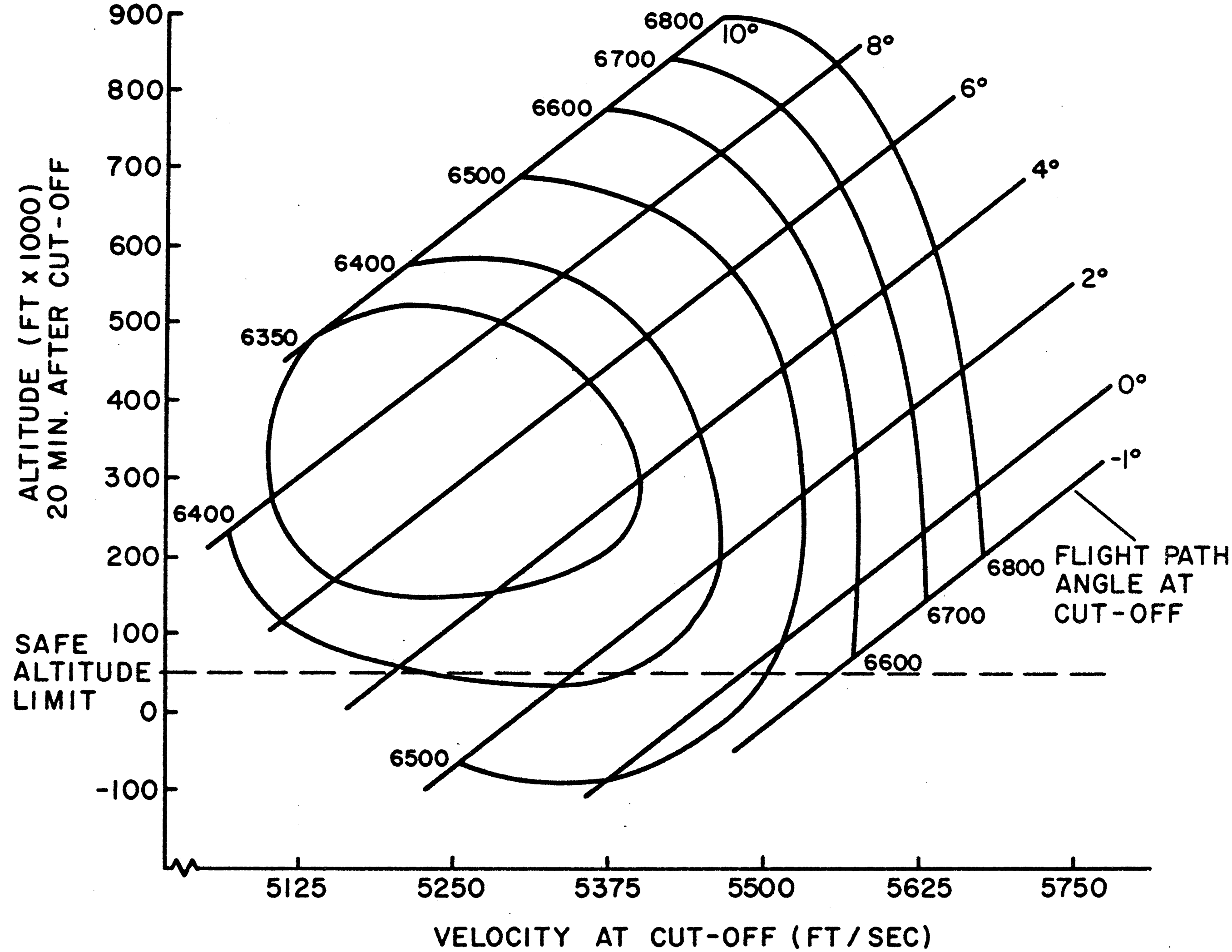

Fig. 6 Operational envelope-abort from hover-transfer velocity correction occurs 20 min after cutoff.

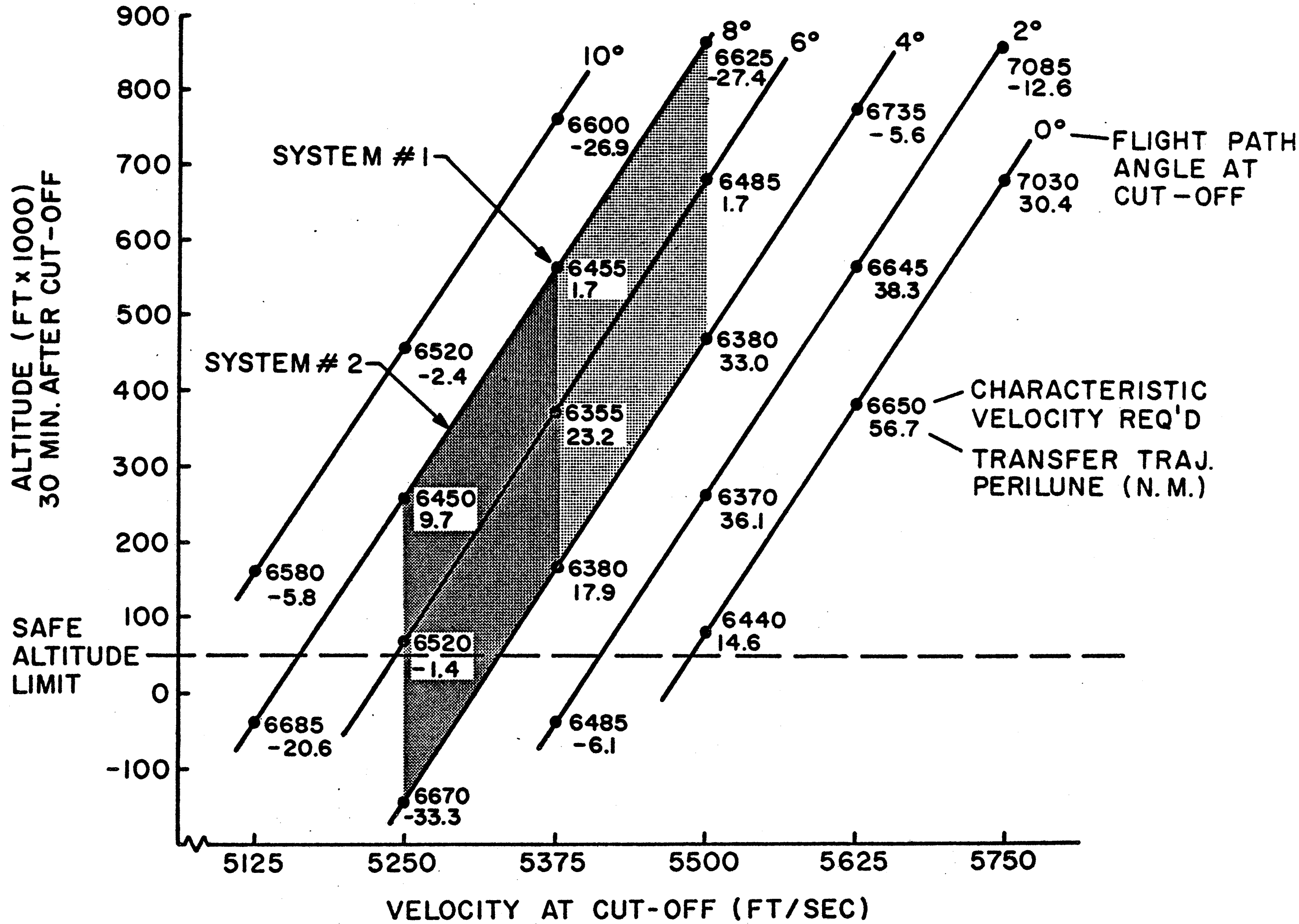

Fig. 7 Operational envelope-abort from hover-transfer velocity correction occurs 30 min after cutoff.

TABLE I
ABORT FROM HOVER
Uncertainties* at Cut-off

Perturbations	H (altitude)	$\dot{H}$	R (range)	$\dot{R}$	T (track)	$\dot{T}$
Initial Conditions:						
H (5000 ft)	5525	3	800	1	0	0
$\dot{H}$ (50 ft/sec)	18000	53	2600	8	0	0
R (5000 ft)	700	5	4650	2	0	0
$\dot{R}$ (25 ft/sec)	1275	12	8450	23	0	0
T (5000 ft)	0	0	0	0	4800	2
$\dot{T}$ (50 ft/sec)	0	0	0	0	17000	48
RSS	18885	55	10021	25	17665	48
Alignment:						
Outer Gimbal (Pitch)						
1st angle (1.5°)	(15000)	(60)	(7700)	(30)	(0)	(0)
2nd angle (2°)	(7500)	(104)	(2400)	(25)	(0)	(0)
Arithmetic Subtotal	22500	164	10100	55	0	0
Inner Gimbal (1.75°)	0	0	0	0	23420	165
Spin Axis (2°)	0	0	0	0	17500	92
RSS	22500	164	10100	55	29236	189
Operational:						
Pitch Rate (±5°/sec)	3500	5	8000	13	0	0
Angle Change (1 sec)	300	5	275	2	0	0
Cut-off timing (1 sec)	250	4	550	23	0	0
RSS	3880	8	9712	27	0	0
State Variables:						
Thrust (1%)	5100	37	9700	75	0	0
Mass (1%)	5100	37	9700	75	0	0
ISP (1%)	730	8	2000	22	0	0
RSS	7250	53	13863	108	0	0
RSS(Total)	30555	183	22557	127	34158	195

*3σ values resulting from utilizing the ascent stage only and the crudest abort guidance system presently envisioned.

TABLE II
ABORT FROM SURFACE
Uncertainties* at Cut-off

Perturbations	H (altitude)	$\dot{H}$	R (range)	$\dot{R}$	T (track)	$\dot{T}$
Initial Conditions:						
H (5000 ft)	5525	3	800	1	0	0
$\dot{H}$ (0 ft/sec)	0	0	0	0	0	0
R (5000 ft)	700	5	4650	2	0	0
$\dot{R}$ (0 ft/sec)	0	0	0	0	0	0
T (5000 ft)	0	0	0	0	4800	2
$\dot{T}$ (0 ft/sec)	0	0	0	0	0	0
RSS	5569	6	4718	2	4800	2
Alignment:						
Outer Gimbal (Pitch)						
1st angle (1.5°)	(15000)	(60)	(7700)	(30)	(0)	(0)
2nd angle (2°)	(7500)	(104)	(2400)	(25)	(0)	(0)
Arithmetic Subtotal	22500	164	10100	55	0	0
Inner Gimbal (1.75°)	0	0	0	0	23420	165
Spin Axis (2°)	0	0	0	0	17500	92
RSS	22500	164	10100	55	29236	189
Operational:						
Pitch Rate (±5°/sec)	3500	5	8000	13	0	0
Angle Change (1 sec)	800	5	275	2	0	0
Cut-off timing (1 sec)	250	4	5500	23	0	0
RSS	3600	8	9712	27	0	0
State Variables:						
Thrust (1%)	5100	37	9700	75	0	0
Mass (1%)	5100	37	9700	75	0	0
ISP (1%)	730	8	2000	22	0	0
RSS	7250	53	13863	108	0	0
RSS (Total)	24657	175	20752	124	29627	189

*3σ values resulting from utilizing the ascent stage only and the crudest abort guidance system presently envisioned.

TABLE III

Ascent Cut-Off Conditions	Total Ascent, Transfer, and Rendezvous Velocity Requirements****		
Flight Path Angle (deg)	4^{o}	6^{o}*	8^{o}
Out of Plane Angle (deg)	2^{o}	2^{o}	2^{o}
Total Velocity (ft/sec)			
5250	6385	6310	6330
5375*	6340	6365	6455***
5500	6495	6565	6670**

* Nominal ascent cut-off conditions for System #1

** "Worst Case" for System #1

*** "Worst Case" for System #2

**** In all cases application of the transfer ΔV took place twenty minutes after ascent cut-off.

TABLE IV
CHARACTERISTIC VELOCITY BUDGET (ft/sec)

	System #1 (Time)	System #2 (Acceleration)
Ascent*	5915	5790
1% Analysis Uncertainty**	60	60
Transfer & Rendezvous	670	580
Docking**	25	25
Total Required	6670	6455
Total Available	7101	7101
LEM = 6646		
CM = 455		
Excess Fuel	+431 ft/sec	+646 ft/sec

* Includes a 10 second vertical rise initially and cut-off conditions biased to account for guidance uncertainties summarized in Table I.

** As per NASA budget

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