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OPERATIONAL ABORT PLAN FOR APOLLO 10 (MISSION F)

Flight Analysis Branch

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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PROJECT APOLLO

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By Contingency Analysis Section
Flight Analysis Branch

April 9, 1969

MISSION PLANNING AND ANALYSIS DIVISION
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OPERATIONAL ABORT PLAN FOR APOLLO 10 (MISSION F)

By Contingency Analysis Section

1.0 SUMMARY

A continuous method has been defined for a safe return of the flight crew to earth for the Apollo 10 (Mission F) mission, with or without ground control help. The rationale and supporting data are given. The supporting data consist primarily of maneuver monitoring techniques and limits used to protect against known constraints and abort trajectory data produced by computer simulations of the recommended abort procedures.

This document does not include techniques, descriptions, or supporting trajectory data for contingencies that require LM rescue or contingency rendezvous.

2.0 INTRODUCTION

The purpose of this document is to demonstrate that an adequate return to earth abort plan exists for all mission phases of the first manned Apollo CSM/LM flight to the moon, the Apollo 10 (F/CSM-106/LM-5) mission. In addition, the document presents information that could be used by ground controllers and by the crew to provide a safe abort from a mission launched May 17, 1969, on a 72° flight azimuth from launch pad 39A. This document does not contain an analysis of LM rescue or contingency rendezvous.

Of particular importance is the relationship among the various methods of abort described in this document and the capability to abort at any time, which is normally provided by RTCC and ground control procedures. The relationship is presented in figure 2-1, which also presents the level of failure from the nominal mission that is required before a particular abort mode would be used. Because most crew-determined abort circumstances occur during a powered-flight phase of the mission, nominal maneuver monitoring procedures are required to provide the necessary safety constraints to insure abort capability. Detailed ground and crew procedures for all methods of abort required for this mission are presented in references 1 and 2. This abort plan document consists primarily of abort trajectory data that would result from aborts with each of the methods identified in figure 2-1. Also, the abort plan shows that an abort procedure and the required data will be available throughout Apollo 10 (Mission F) if a contingency should arise. Launch phase information and TLI trajectory information were obtained from reference 3, and the nominal spacecraft trajectory characteristics were obtained from reference 4. The information in this document is very similar to information presented in reference 5, the Apollo 8 abort plan. The primary differences result from the additional capabilities provided by the LM.

Input constants common to the analyses of the phases of the mission are presented in appendix A. This document and its relation to other Apollo 10 (Mission F) milestones for the Contingency Analysis Section are shown in appendix B.

This document presents trajectory data that simulate abort-related events prescribed in the Mission F contingency techniques document (ref. 1). Note that the contingency techniques document does not prescribe the use of two-impulse abort maneuvers except in the case of LOI mode II. The use of two-impulse abort maneuvers is not precluded if it is determined in real time that the situation warrants their use. The prime purpose in the selection of the two-impulse mode would be to achieve a much faster return time by a sequence of DPS and SPS engine burns designed to provide maximum ΔV. In appendix C is presented an analysis of fastest return times possible from the free-return trajectory that approximates two-impulse maneuvers from translunar coast.

3.0 ABBREVIATIONS

ACRA	Atlantic continuous recovery area
ADRA	Atlantic discrete recovery area
AGC	abort guidance control
AOL	Atlantic Ocean line (recovery)
CDR	commander
CLA	contingency landing area
CM	command module
CMC	CM computer
CO	cutoff
COAS	crew optical alinement sight
COI	contingency orbit insertion
CSM	command and service modules
c.g.	center of gravity
DPS	descent propulsion system
DSKY	display keyboard
DVM	ΔV magnitude
EMS	entry monitoring system
EOI	earth orbit insertion
EPL	Eastern Pacific line (recovery)
EPO	earth parking orbit
ESS	early S-IVB staging
FCUA	fuel-critical unspecified area

FDAI	flight director attitude indicator
GETL	ground elapsed time of landing
G.m.t.	Greenwich mean time
g	entry load
g.e.t.	ground elapsed time
h	altitude
$\dot{h}$	altitude rate
IGA	inner gimbal angle
IMU	inertial measurement unit
IOL	Indian Ocean line (recovery)
I_{sp}	specific impulse
LET	launch escape tower
LEV	launch escape vehicle
LM	lunar module
LOI	lunar orbit insertion
LOI-1	first LOI burn into a 60- by 170-n. mi. altitude orbit
LOI-2	lunar orbit circularization burn into a 60- by 170-n. mi. altitude orbit
LPO	lunar parking orbit
LV	launch vehicle
L/D	lift-to-drag ratio
MCC	midcourse correction
MCC-H	Mission Control Center - Houston
MGA	middle gimbal angle

MPL	mid-Pacific line (recovery)
MSI	moon's sphere of influence
MSFC	Marshall Space Flight Center
MSFN	Manned Space Flight Network
NR	North American Rockwell
N50E	CMC program to enter NOUN 50
OGA	outer gimbal angle
PC	plane change
PGNCS	primary guidance, navigation, and control system
POSMAX	maximum quantity displayed in a given register
PTC	passive thermal control
P00	CMC program 00
P11	CMC program 11
P37	CMC program 37 (return to earth)
RCS	reaction control system
REFSMMAT	transformation matrix from inertial to stable member (IMU)
RL55	roll left 55°
RL90	roll left 90°
RTCC	Real-Time Computer Complex
R_{ip}	predicted full-lift landing range from the launch pad
SC	spacecraft
SCS	stabilization and control subsystem
S-IVB	launch vehicle third stage
SLA	spacecraft LM adapter

SPLERROR (ΔR)	difference between the onboard predicted landing point and the mode III target point
SM	service module
SPS	service propulsion subsystem
T&D	transposition and docking
TAR	time from abort to reentry
TB_7	time base 7 - initiated at TLI cutoff
TEC	transearth coast
TEI	transearth injection
TFT	total flight time from TLI, LOI, or TEI shutdown to landing
THC	thermal control
TLC	translunar coast
TLI	translunar injection
t	time of lift-off
t_B	burn time
t_D	time of abort
t_{ff}	time of free fall
t_{IG}	time of ignition
USBS	Unified S-band System
V_{EI}	entry velocity
V_i	inertial velocity
V75E	CMC program to enter VERB 75

V82E	CMC program to enter VERB 82
WPL	West Pacific line
ΔR	SPLERROR
ΔV	total sensed velocity change

Subscripts:

a	apogee
min	minimum
p	perigee

4.0 GUIDELINES AND CONSTRAINTS

This document is based on a number of fundamental guidelines and constraints; the most important are listed below.

1. An abort is defined as the recognition and performance of those conditions necessary to terminate the current mission and return the flight crew to earth.

2. An alternate mission is defined as the continuation of the flight, usually with less ambitious objectives than were originally planned.

3. Return to earth abort maneuvers normally are targeted to CLA's. The CLA's for Apollo 10 (Mission F) are shown in appendix A.

4. Aborted mission return times are consistent with known system constraints and generally are optimized to provide the fastest return for the least ΔV.

5. When it is practicable, the LM DPS engine will be used before the CSM SPS engine is used.

6. Return to earth inclinations will not exceed 40°.

7. The inertial velocity at entry will not exceed 36 333 fps, except for time-critical returns (37 500 fps).

5.0 LAUNCH PHASE

5.1 Introduction

The launch abort trajectory data provide information about abort monitoring, about abort maneuver requirements, and about abort results. It is assumed that the launch vehicle performance can vary over a wide range of conditions during launch. Therefore, the conditions must be bounded by limits that would allow sufficient reaction time for the crew and the spacecraft systems operations to perform a safe abort. If the launch vehicle violates these limits, abort action would be initiated to prevent a flight with unsafe conditions. To avoid the abort of a successful launch, the limit lines are defined for the least restrictive conditions that will allow a safe abort.

During launch, the initial abort conditions that could be expected change drastically (figs. 5-1, 5-2, and 5-3). Because of these changes, several abort modes are required, each adapted to a portion of the launch trajectory.

1. Mode I aborts protect the SC and crew while the LV is on the pad and in atmospheric flight. The launch escape system is used for safe separation, and the aborts result in a suborbital trajectory with landings in the ACRA.

2. Mode II abort capability begins when the LET has been jettisoned (197 sec g.e.t.) and continues until the COI capability begins or until the resultant landings threaten the African coast (R_{ip} = 3200 n. mi.). Mode II aborts consist of a manual CSM separation from the LV, CM/SM separation, an entry orientation maneuver, and an open-loop, full-lift entry. Mode II aborts also result in a suborbital trajectory with landings in the ACRA.

3. The mode III abort capability begins at the end of mode II and continues until the maneuver violates free-fall time. The mode III aborts consist of a manual CSM separation, a fixed-attitude SPS retrograde burn, CM/SM separation, an entry orientation maneuver, and an open-loop, bank-left 55° entry. The abort maneuvers result in a suborbital trajectory with landings at the ADRA approximately 3350 n. mi. down range of the launch pad, just south of the flight azimuth.

4. The S-IVB orbital capability begins when the S-IVB can be used to insert the S-IVB/LM/CSM configuration into a nominal orbit (approximately 6 min g.e.t.). The early S-IVB staging consists of a manual upstage of the S-IVB from the malfunctioning S-II stage. The upstage maneuver will result in a nominal EOI from which the nominal mission or an alternate mission can be planned.

5. The SPS COI capability begins when the SPS can be used to insert the CSM into a safe orbit ($V_i \approx 22\ 000$ fps or g.e.t. = 517 sec) and continues until the LV has obtained a safe orbit. The COI maneuver consists of a manual CSM separation and a fixed-attitude, posigrade SPS burn by 90 seconds after abort which results in a 200-n. mi. apogee altitude or a 75-n. mi. perigee altitude, whichever occurs first. If the 200-n. mi. apogee altitude is achieved first, the burn is terminated; then a second fixed-attitude, posigrade SPS burn must be performed at apogee to raise perigee altitude to 75 n. mi. or greater. If the 75-n. mi. perigee altitude is achieved first, the burn is terminated at a perigee altitude greater than 75 n. mi. or whenever perigee altitude starts to decrease. The maneuvers result in a safe orbital trajectory from which an alternate mission or an immediate deorbit can be planned.

The launch abort mode capabilities are summarized on a bar chart (fig. 5-4) for the nominal 72° launch azimuth time line. Also shown in figure 5-4 is the ESS capability region, which defines the time at which the S-IVB can stage directly from the S-II (360 sec g.e.t.) and still achieve a parking orbit.

The initial state vectors used for the abort initiation for this study were based on the AS-505 operational launch trajectory (ref. 3). A list of the key events during launch is presented in table 5-I. Pertinent SC constants and performance characteristics for CSM-106 used for this study are presented in appendix A. Other constants that are peculiar to the launch phase are presented in table 5-II.

The launch abort data shown here are consistent with the latest Apollo 10 (Mission F) characteristics and are for aborts from the nominal 72° azimuth launch trajectory. A detailed analysis of CSM aborts (modes II, III, and IV) from a typical Saturn V launch trajectory is presented in reference 6; the analysis shows the effects of variable launch azimuths. The data are directly applicable to Apollo 10 (Mission F) and can be used to estimate the effects of variable azimuths on the launch abort modes. The sensitivities of the various launch abort parameters for variations in weight, in altitude, in burn attitude, and in other parameters are discussed in reference 7. Another document that should be used to supplement the launch abort information presented is reference 2. This reference presents the launch phase abort techniques and data flow for the Saturn V Apollo launches and contains the flow charts and rationale for the abort cues, decisions, and data flow for each of the abort modes. Note that the reference trajectory assummed launch would occur from launch pad 39A.

5.2 Launch Trajectory Monitoring

5.2.1 Ground monitoring.- The ground (MCC-H) flight controllers have the primary responsibility to monitor the trajectory during the launch phase. The ground control is prime to determine abort trajectory limit violations, abort mode decisions, and the GO/NO-GO orbit insertion status. The flight dynamics displays help ground control to monitor trajectories. These displays consist of the launch digitals and the projection plotters displayed on cathode ray tubes and on analog plotboards. The displays are driven by real-time computer computations based on the actual flight data received from the MSFN. The flight dynamics displays currently used in Apollo 10 (Mission F) simulations are presented in reference 8. The displays will be similar for all the planned Saturn V launches and are defined in reference 6.

The launch abort trajectory limits are summarized in figure 5.5. The crew safety limits include a structural breakup limit, a 16g limit, a 100-second free-fall time limit, and an exit heating limit (ref. 9). The limits define a launch corridor that is acceptable for safe SC abort capability. Another corridor which could be used lies between the previously described corridor and the nominal trajectory and is composed of the bounds of the malfunctioning LV trajectories which make orbit. In other words, when the latter corridor is violated by the launch trajectory trace, the launch vehicle is beyond all probable orbital capability and will eventually arrive at abort conditions. Therefore, the actual abort decision can be made when the launch trajectory trace pierces this envelope, but the abort action must be requested prior to the time that the abort trajectory limits are exceeded. The use of the malfunctioning LV capability to orbit envelope has two advantages: it can be used as an advanced abort call to eliminate additional flight with a malfunctioning LV; or it can be used to provide additional abort flexibility, to choose a suitable abort through improvement of entry or landing conditions or both prior to crew abort limit violation. The envelope was defined based on a sophisticated analysis of the more probable malfunctioning LV trajectories for AS-503 (ref. 10). A similar analysis is currently being conducted by MSFC for AS-505. The eventual operational use of this envelope for AS-505 would require MSFC authorization. In addition to the limits and the nominal trajectory, the S-IVB early staging (ref. 11) and the SPS COI capability lines are shown in figure 5-5. The latter two lines define, respectively, when the S-II has progressed sufficiently for the S-IVB to stage directly into a parking orbit (100 n. mi., circular) and when the S-II or S-IVB has progressed enough for the SPS (mode IV) to insert the SC into contingency orbit (h_p > 75 n. mi.).

The abort mode overlap can be determined by a comparison of the COI capability with the suborbital capability for the near-insertion region (fig. 5-6). The mode IV COI capability is shown in figure 5-6 to overlap the end of mode II and all of mode III along the nominal trajectory. The dispersed S-IVB cutoff conditions that would require a mode III or apogee kick maneuver are also shown in figure 5-6. The 75-n. mi. perigee altitude line indicates when the S-IVB has achieved a GO orbit, and the 200-n. mi. apogee line indicates an S-IVB overspeed condition. Note that mode III capability is limited by a 100-second t_{ff} constraint and that increased insertion ranges would further restrict the mode III capability. Therefore, large mode III SPS burns could be terminated at the 100-second t_{ff} limit prior to achievement of the landing target. Zero lift (roll left 90°) is recommended for those cases that require premature termination. The apogee kick COI capability is bounded by two things: the physical capability of the SPS to achieve a 75-n. mi. perigee altitude at apogee, which is the apogee kick boundary; and the assured crew reaction time for SPS ignition of 90 seconds from abort to apogee. The apogee kick COI maneuver will not be performed until the LV cutoff conditions are beyond the 5-minutes-to-apogee line. The SPS COI capability is divided into three regions (fig. 5-7). The first region requires two SPS burns, one burn as soon as possible within 90 seconds after abort to raise apogee altitude to 200 n. mi. (h_p < 75 n. mi.) and the other burn at apogee to raise perigee altitude to at least 75 n. mi. The second region requires only one SPS burn as soon as possible within 90 seconds after abort to raise perigee altitude to at least 75 n. mi. (h_a < 200 n. mi.). The third region requires only one SPS burn at apogee to raise perigee altitude to at least 75 n. mi.

The trajectory lines shown in figures 5-5, 5-6, and 5-7 are analogous to the plotboard information displayed to the flight controllers in real time. Comparison of the actual launch trace with the trajectory information will help the flight controllers to determine the trajectory status during launch and to determine the appropriate abort mode, if necessary. Because the S-IVB early staging and SPS capabilities depend on altitude, additional plotted information (figs. 5-8 and 5-9) will be required to determine the capabilities for nonnominal altitude histories. The ground control will keep the crew informed of the trajectory status by voice communications and, upon abort confirmation, will request abort action by voice and with the abort light.

5.2.2 Onboard monitoring.- To facilitate trajectory monitoring during launch, the crew uses the FDAI displays and CMC program P11 and its corresponding DSKY displays. The P11 is initiated automatically at lift-off (or manually by V75E) and is available until the ground control or the crew commands P00. Normally, the ground control will inform the crew of the status of the spacecraft trajectory. However, if voice communications were lost during launch, the crew would have to depend on the displays to determine the status of the trajectory. The values of the DSKY parameters for a nominal launch are shown in table 5-III; the values were computed with the COLOSSUS guidance equations (ref. 12) for Mission F. The nominal FDAI attitudes during the launch are shown in figure 5-3. The DSKY displays are updated every 2 seconds and are displayed to the crew. If the ground control should rule NO-GO for the SC guidance computer, the computer would be commanded to P00, and the DSKY displays would no longer be available.

In conjunction with the DSKY displays associated with P11 (fig. 5-10), an onboard chart (fig. 5-11) is proposed for use in the event of voice communications loss during the launch. The basic DSKY displays used to monitor the launch are the inertial velocity V_i, altitude rate $\dot{h}$, and altitude h parameters. Therefore, these parameters are used to govern the charts. The charts are used with the DSKY displays to help determine when and what abort action is necessary. This function normally would be conducted by the ground control when voice communications exist. After the abort decision has been made, the crew would use the DSKY parameters to monitor the abort burn. The crew is prime (with voice) to perform the COI maneuver with the SPS. The procedure is to manually separate the CSM from the LV, aline the SC heads down posigrade with the window scribe on the horizon so that a 31.7° angle exists between the line of sight to the horizon and the X-body axis, and then ignite the SPS as soon as possible in the SCS automatic mode. The SPS burn will be terminated at a 200-n. mi. apogee altitude if this altitude would preclude a 75-n. mi. perigee altitude (in the two-impulse region only); a second SPS burn would then be initiated at apogee ($\dot{h} = 0$) with the window scribe and far horizon used for attitude reference. The second burn would be analogous to the apogee kick maneuver and would be used to raise perigee altitude to at least 75 n. mi. Note that a check on $\dot{h}$ should be made prior to termination of the first burn at $h_a = 200$ n. mi. ($h_p < 75$ n. mi.); if $\dot{h} < 0$ fps, the first burn should be continued until $h_p \geq 75$ n. mi. or until perigee altitude starts to decrease. If the LV cutoff conditions are within the apogee kick region, the SPS burn should be delayed until apogee, at which point only one burn will be required to raise h_p to 75 n. mi. If t_{ff} decreases to 100 seconds and continues to decrease during a burn, the burn must be terminated and immediate entry preparation must be initiated. Caution should be used during the mode IV burn. If at any time during the burn perigee altitude starts to decrease, the burn should be terminated. If the burn is terminated with $h_p < 75$ n. mi., a mode III abort should be initiated when $\dot{h} \leq 0$ or when $h_a < 75$ n. mi., and an apogee kick should be initiated when $\dot{h} > 0$.

The first part of the onboard chart (fig. 5-11) shows the comparison of nominal altitude rate and velocity and shows the current abort trajectory limits. An abort would be required if the actual flight trace violated the booster breakup line, the exit heating limit line, or the maximum entry load limit line (16g). If the trace approaches the t_{ff} limit line, V82E and N50E should be called; abort action is taken when t_{ff} equals 100 seconds and is decreasing. Note that, even if voice communications were lost, the ground control still might be able to command abort action by use of the abort light. Because of the sensitivity to altitude of the 16g limit line, this limit is shown for several different altitudes; and the current altitude displayed on the DSKY would govern which abort limit to use. In addition to the abort trajectory limits, the malfunctioning LV capability to orbit envelope is presented. This envelope can be used to make the abort decision, with final safe abort conditions defined by the abort trajectory limits. The ground control is prime to request aborts for trajectory violations when voice communications exist. A discussion of the envelope is presented in section 5.2.1. The envelope is insensitive to altitude deviations and would be much simpler and safer to use for abort action in the no-voice case than the 16g limit for the different altitudes. The S-IVB early staging orbital capability line is also shown in figure 5-11.

The second part of the onboard chart shows the comparison of nominal altitude rate and velocity for approximately the last 3 minutes of the launch. The chart expands the region where abort capability starts to vary rapidly. The primary purpose of this chart is to show for which LV cutoff conditions COI capability exists. The three regions of SPS COI capability are shown. The first region (two-impulse) requires two SPS burns; one burn as soon as possible within 90 seconds after abort to raise h_a to 200 n. mi. (h_p < 75 n. mi.) and the other burn at apogee to raise h_p to $\geq$ 75 n. mi. The second region (single impulse) requires one SPS burn to raise h_p to 75 n. mi. as soon as possible after abort. The third region (apogee kick) requires one burn at apogee ($\dot{h}$ = 0) to raise h_p to $\geq$ 75 n. mi. The initial SPS COI boundary (two-impulse) is defined for different altitudes. Because the altitude is fairly static near insertion, the crew could choose the appropriate COI boundary and could determine when the LV trace crosses into the COI capability region. The suborbital abort capabilities can be determined directly from the DSKY. After tower jettison has occurred, mode II capability extends until ΔR becomes greater than -400 n. mi., which corresponds to a full-lift landing at approximately 3200 n. mi. A no-burn, half-lift entry (RL55) abort procedure is required when S-IVB cutoff conditions result in a ΔR of between -400 and 0 n. mi. when a suborbital abort is required. For $\Delta R > 0$, a mode III burn is required. A GO orbit is achieved when perigee altitude is greater than or equal to 75 n. mi.

Whenever the t_{ff} is 59^m59^s, the ΔR computation is invalid, which occurs when the perigee altitude becomes greater than 300 000 feet. If a mode III burn is required in this region, ΔR will become valid when the burn has progressed enough to decrease perigee altitude to below 300 000 feet.

The effects of launch azimuth variations in the ΔR computation have been investigated (ref. 13). Because the ΔR computation is based on the prelaunch loading of the mode III target (ADRA), the computation would be erroneous for launches on other than the planned launch azimuth. The study in reference 13 indicated that the mode III target for the 72° launch azimuth would result in Atlantic landings when $\Delta R = 0$ for all launch azimuths between 72° and 108° and could be used for these launch azimuths for the no-voice cases.

5.3 Suborbital Aborts

5.3.1 Mode I LEV aborts.- The possibility of mode I LEV aborts from the Saturn V vehicle launched from complex 39A exists from the time the LEV is armed until the launch escape tower is jettisoned.

The LEV is designed to accelerate the CM away from the LV to a safe separation distance and far enough down range from the launch pad for a safe water landing. Mode I aborts are divided into three categories: mode IA (low altitude), mode IB (medium altitude), and mode IC (high altitude).

The mode I (LEV) abort data presented in this document were generated for the Apollo 9 mission (ref. 14). The input data necessary to generate mode I LEV aborts for the F mission are not available at this time. However, the Apollo 9 mission mode I abort data are considered to be representative of the F mission mode I abort trajectory characteristics and can be used until the final data are published.

A summary of the nominal mode I (LEV) abort trajectories (no winds) for a nominal launch trajectory of 72° flight azimuth is presented in table 5-IV. Mode I LEV abort landing points for pad aborts through 195.9 seconds g.e.t. are presented in figures 5-12 and 5-13. Safe water landings can be made at landing points from near the pad to approximately 519 n. mi. down range.

5.3.2 Mode II aborts.- The mode II abort procedures are designed for contingencies that occur during the time after the LET jettison (197 sec g.e.t.) until a safe orbit can be achieved or until the resultant landings threaten the west coast of Africa (R_{ip} = 3200 n. mi.).

Because the aborts initiated in this region can result in high entry loads or time-critical entries or both, no range control maneuvers are considered. A full-lift entry is used to minimize the entry load, and a simple separation technique is established for rapid entry orientation. To orient the spacecraft to the proper atmospheric capture attitude, the mode II procedure requires at least a 100-second t_{ff} from S-IVB cutoff to the time that a 300 000-foot altitude is achieved. For this condition to be fulfilled for low launch trajectories, there is sometimes a requirement for an extension of the mode I region by delay of the tower jettison until sufficient free-fall time is available to perform the mode II abort.

The sequence of events simulated for a mode II abort are listed below.

t plus 0 second	LV shutdown; beginning of tailoff
t plus 3 seconds	LV/CSM separation; +X SM RCS ON (four-jet)
t plus 24 seconds	+X SM RCS OFF, CM/SM separation sequence begun; CM oriented to entry attitude
g = 0.5	CM oriented for full-lift entry
h = 23 500 feet	Drogue parachute deployment

A list of the pertinent trajectory parameters for mode II aborts from the nominal launch trajectory are presented in table 5-V. The resultant t_{ff}, entry load, and landing range that result after a mode II abort are plotted in figure 5-14. The spacecraft IMU gimbal angles that correspond to the proper CM entry orientation attitude for mode II aborts are compared with time of abort and landing range in figure 5-15. A more detailed analysis of the mode II aborts for the Saturn V launches is presented in reference 6.

5.3.3 Mode III aborts.- The mode III abort procedures are required for contingencies that occur beyond mode II (R_{ip} > 3200 n. mi.) when a safe orbit cannot be achieved or when SC systems malfunctions dictate immediate landings. The first mode III requirement is unlikely because of the large COI region and because the LV cutoff conditions would have to be greatly dispersed from the nominal launch trajectory. The second requirement is unlikely because if such a malfunction had occurred during launch the abort would more probably be initiated before mode III was entered, and failures that occur after mode III is entered would be almost impossible to confirm in sufficient time to recommend a mode III abort. These types of failures are undefined at present.

The sequence of events simulated for a mode III abort are listed below.

t plus 0 second	LV shutdown; beginning of tailoff
t plus 3 seconds	S-IVB/CSM separation; +X SM RCS ON (four-jet)
t plus 24 seconds	+X SM RCS OFF; orientation begun to SPS retrograde attitude if burn required
t plus 125 seconds	Retrograde attitude obtained (fig. 5-16); SPS engine ignition (SCS automatic)
Half-lift landing range = 3350 n. mi.	CM/SM separation sequence begun; CM oriented to entry attitude; SPS burn terminated
g = 0.05	CM oriented for full-lift entry; capture attitude (fig. 5-17)
g = 0.2	CM oriented for half-lift entry; RL55
h = 23 500 feet	Drogue parachute deployment

Mode III abort capability begins at the end of mode II when R_{ip} exceeds 3200 n. mi. (581 sec g.e.t.). Because mode III entries are half lift (RL55) and because the SPS retrograde burn is required only to achieve a landing range of 3350 n. mi., a period exists between 581 and 604 seconds g.e.t. for which the no-burn landing would land west of the 3350-n. mi. landing target (ADRA). The mode III capability ends when the required SPS burn violates the 100-second t_{ff} constraint. The violation of the constraint occurs prior to the nominal orbit insertion [table 5-VI(a)], and suborbital aborts required after that time would require termination of the burn at t_{ff} = 100 seconds followed by a zero lift (RL90) entry to avoid a land landing.

A list of the pertinent trajectory parameters for mode III aborts from the nominal launbh trajectory are presented in table 5-VI. The SC IMU gimbal angles that correspond to the horizon-monitored (31.7° window scribe mark) retrograde SPS burn attitude are presented in figure 5-16 for mode III aborts from the nominal trajectory. The required ΔV and the resultant t_{ff} and entry load for the mode III aborts are plotted in figure 5-18. The mode III ΔV requirements to achieve landings at the ADRA are shown in figure 5-19 for deviations from the nominal flight-path angle. Note from these figures that the mode III region is bounded by the end of mode II, by the 16g entry load limit, and by the 100-second t_{ff} limit. The proper capture angles are shown in figure 5-19 for the mode III aborts from the nominal LV trajectory. A more detailed analysis of the mode III aborts for the Saturn V launches is presented in reference 6.

5.4 Contingency Orbit Insertion

The mode IV COI procedure is selected for launch contingencies when the SPS can insert the SC into a safe orbit (perigee altitude $\geq$ 75 n. mi.) and can deorbit the SC from any place in the resultant orbit. The capability begins at 517 seconds ($V_i \approx$ 22 000 fps) and ends when the S-IVB has achieved a safe perigee, approximately 672 seconds g.e.t., or 2 seconds prior to nominal S-IVB cutoff signal. The COI maneuver (S-IVB or SPS) is the prime selection whenever the capability exists because it is the safest and has potential alternate mission capability. The COI maneuver allows the ground control and the crew ample time in earth orbit to determine the SC trajectory and systems status, and the ground control can compute a precise deorbit maneuver for a planned landing area.

Three different COI procedures are recommended (fig. 5-7) for use on Apollo 10 (Mission F).

1. The first procedure, a mode IV two-impulse abort, requires two SPS burns to achieve a contingency orbit. The first burn is performed as soon as possible and is terminated when h_a = 200 n. mi. (h_p < 75 n. mi.). The burn shapes the trajectory so that a second burn can be performed at apogee to raise perigee altitude. Consequently, the second burn is performed at apogee, and it is terminated when $h_p \geq$ 75 n. mi.

The sequence of events simulated for the two-impulse mode IV maneuver is listed on the next page.

t plus 0 second	Abort-initiated LV shutdown; beginning of tailoff
t plus 3 seconds	LV/CSM separation; +X SM RCS ON (four-jet)
t plus 24 seconds	+X SM RCS OFF; orientation begun to SPS posigrade attitude
t plus 90 seconds	Posigrade attitude obtained (fig. 5-20); SPS engine ignition (SCS automatic)
h_a = 200 n. mi. or $h_p \geq 75$ n. mi.	Termination of SPS burn
If burn terminated on h_a = 200 n. mi. with $h_p < 75$ n. mi. and $\dot{h} = 0$	Beginning of coast to apogee
$\dot{h} = 0$ (apogee)	Posigrade attitude obtained (fig. 5-20); SPS engine reignition (SCS automatic)
$h_p \geq 75$ n. mi.	Termination of SPS burn

2. The second procedure, a mode IV single-impulse abort, requires only one SPS burn to achieve a contingency orbit. The burn is initiated as soon as possible and is terminated when $h_p \geq 75$ n. mi. ($h_a \leq 200$ n. mi.). The sequence of events simulated for this maneuver is the same as the sequence for the first procedure except that the second burn is not required.

3. The third procedure, the apogee kick, also requires only one SPS burn to achieve orbit. However, the burn is delayed until apogee ($\dot{h} = 0$) and is terminated when $h_p \geq 75$ n. mi. The sequence of events simulated for this maneuver is the same as the sequence for the first maneuver except that the first burn time is deleted and the burn is performed at apogee ($\dot{h} = 0$).

A list of the pertinent trajectory parameters for mode IV maneuvers performed from the nominal trajectory is presented in table 5-VII. The SC IMU gimbal angles that correspond to the horizon-monitored (31.7° window scribe mark) posigrade SPS burn attitudes are presented in figure 5-20 for burns from the nominal trajectory. The SPS ΔV's and associated burn times for mode IV burns from the nominal launch trajectory are presented in figure 5-21; the resultant orbital parameters are presented in figure 5-22. Some typical cases of aborts from the nominal mission are presented in figures 5-23 and 5-24 which show free-fall time, altitude, perigee altitude, apogee altitude, and true anomaly histories during the SPS burns.

The initial mode IV capability is not dependent upon the amount of SPS propellant loaded for the mission but is based on the SPS capability to achieve orbital velocity with the fixed burn attitude prior to premature entry. This capability is extremely sensitive to pitch errors during the maneuver (refs. 6 and 7). Therefore, the capability is defined for a ±2° pitch error bias during the burn. However, yaw errors of up to 15° have a negligible effect on the maneuver and are not included in the bias. The constraints limit the maximum combined ΔV for COI to less than 7000 fps for the nominal insertion attitude (figs. 5-25 through 5-28).

Constant ΔV lines required for the COI maneuvers are presented in figures 5-25, 5-26, and 5-27. The SPS ΔV required to achieve either h_a = 200 n. mi. or h_p = 75 n. mi. is presented in figure 5-25 for the first burn. Note that in the two-impulse region the first SPS burn will enable the spacecraft to achieve h_a = 200 n. mi. before it reaches h_p = 75 n. mi. The recommended procedure is to terminate the burn at h_a = 200 n. mi. (h_p < 75 n. mi.); if $\dot{h} = 0$, the spacecraft would coast to apogee ($\dot{h} = 0$) and the SPS would be reignited to raise h_p to 75 n. mi. The corresponding ΔV and ignition times for the second SPS burn based on the initial abort conditions (LV cutoff) are shown in figure 5-26. It is important to note that there is a region in the single-impulse area where $h_a \geq$ 200 n. mi. and h_p < 75 n. mi. with $\dot{h} < 0$ (approaching perigee); for these conditions, the first burn must be continued until $h_p \geq$ 75 n. mi. or until $\dot{h} > 0$ (approaching apogee). If h_p reaches a maximum prior to reaching 75 n. mi. with $\dot{h} < 0$, then a deorbit maneuver would be required. The constant ΔV lines and the SPS ignition times for the apogee kick maneuver are presented in figure 5-27.

The SPS ignition time for the first burn is as soon as possible. However, for the studies included in this document, an ignition time of 90 seconds was used. The time was based on crew reaction as the earliest time for which the crew could always assure an SPS ignition from time of abort. The RTCC is configured to compute a mode IV COI solution for a 125-second SPS ignition time for a single burn to raise h_p to 75 n. mi. Therefore, for delayed ignition times or for loss of the onboard computer, the ground control would have the mode IV COI solution available to pass to the crew. The constant SPS ΔV lines that correspond to the RTCC solution are presented in figure 5-28.

6.0 EARTH PARKING ORBIT

Preflight computations for aborts from EPO are provided for the crew and are targeted so that the landing occurs in one of three possible areas. Should it become necessary to abort while the crew is out of communication with the ground control, a solution would be available. After orbit is reached, the ground control updates the solutions so that the crew always has one solution for a revolution beyond the time it would be used. Because this type of abort is well documented (ref. 15), no further information is required of this document.

7.0 TRANSLUNAR INJECTION AND TRANSLUNAR COAST PHASE

7.1 Translunar Injection Monitoring

The primary objective after a problem develops during TLI, as well as during all other mission phases, is to perform an alternate mission (fig. 2-1). With this objective as the next most desirable one after crew safety, the extent of the deviated flight conditions must be known in advance to insure that the desired alternate mission capability will exist. Also, consideration has been given to provision of reasonable initial conditions for performance of an abort maneuver. These things have been done by the development of a crew monitoring procedure which includes appropriate S-IVB shutdown limits.

The crew must be able to monitor and evaluate TLI without ground support because the maneuver can occur off the MSFN tracking range. Generally, TLI occurs at various locations over the west Pacific Ocean (figs. 7-1 and 7-2). A schematic of the basic crew monitoring technique is shown in figure 7-3. An abort can be performed for S-IVB attitude rate problems and for attitude deviation problems as well as for SC system problems. Because S-IVB problems normally would result in a SC alternate mission, only a critical SC system problem is likely to require an abort.

Several significant items can be noted about the TLI monitoring technique.

1. The TLI maneuver will be inhibited if the launch vehicle attitude before ignition is more than 10° from nominal as determined by horizon reference.

2. The TLI maneuver will be terminated by the crew for S-IVB initiated rates of 10 deg/sec.

3. The TLI maneuver will be terminated by the crew with the abort handle for attitude deviations of 45° from the nominal attitude, which are determined by onboard charts of the nominal pitch and yaw gimbal angle histories.

4. A backup to the S-IVB guidance cutoff signal will be performed by the crew if the S-IVB has not shut down at the end of the predicted burn time plus a 2σ dispersion of 6.0 seconds and if the nominal inertial velocity displayed by the spacecraft computer has been achieved.

The rationale for the monitoring procedures and for the determination of the limits discussed above are documented in references 16, 17, and 18. Because attitude excursions can reduce perigee to as low as 75 n. mi., item 3 has the largest impact on possible abort maneuvers. However, delay of the shutdown to 45° optimizes chances for adequate time for an entry midcourse maneuver after a TLI abort.

The crew charts mentioned in item 3 are shown in figures 7-4, 7-5, and 7-6. The double scale on the pitch chart (fig. 7-4) indicates the TLI ignition gimbal angle for a 72° launch azimuth. For any other day or launch azimuth, the crew will renumber the scale by changing the zero point to the ignition pitch gimbal angle uplinked by the ground control during EPO.

If a tumbled S-IVB inertial platform exists during TLI, the crew may assume manual control of the burn with the hand controller. In this case, the IMU would be used to obtain reference information, and the crew charts would be used to determine required attitudes. Because a ground rule for manual takeover requires illumination of the guidance failure lights when the S-IVB platform is tumbled, the 45° attitude deviation limits are required for protection against other S-IVB malfunctions.

Crew confirmation of a drifting trajectory with any two SC inertial references could cause manual takeover on a smaller limit of approximately 10° if the capability were available.

7.2 Aborts from the Translunar Injection and Translunar Coast Phases

7.2.1 Introduction.- In this section, trajectory analyses are presented for aborts initiated during the second S-IVB burn, for aborts initiated during the period immediately after this burn, and for aborts initiated on the translunar coast leg of Apollo 10 (Mission F). Analyses of abort maneuver dispersions also are presented for aborts performed during and immediately after the second S-IVB burn. Analyses of abort maneuvers performed from dispersed TLI burns have been performed and are available in references 19, 20, 21, and 22.

The trajectory data included in this section represent the results of digital computer simulations of the abort techniques defined in reference 1. Input used to generate the enclosed abort trajectory data for the TLI and TLC aborts are listed in appendix A. The program used to generate the enclosed data is described in reference 23.

7.2.2 The 10-minute abort.- The contingencies with which this section is concerned are the spacecraft subsystems problems which can be isolated during TLI and which can result in catastrophe if immediate action is not taken. At this time, there are no known single point failures which would require the crew to shut down the S-IVB manually and to execute an abort maneuver immediately.

It has been recommended in reference 1 and in numerous meetings with Apollo crew members that, if the situation permits, the crew should allow the S-IVB to complete TLI, at which time the ground control and crew can perform a malfunction analysis to determine the necessity of an abort.

If a critical subsystems failure occurs during TLI and necessitates the shutdown of the S-IVB and the immediate return of the crew to earth, the following sequence will occur to lead to the 10-minute abort. The 10-minute abort is a fixed attitude abort (attitude established preflight, fig. 7-7) performed 10 minutes after S-IVB shutdown and targeted to the contingency entry target line.

Time from S-IVB cutoff, min:sec, g.e.t.	Event
00:00	S-IVB burn time is recorded; THC is turn counterclockwise to initiate S-IVB shutdown; inertial velocity (V_i) is recorded from the DSKY; the four +X RCS jets are turned on
00:03	CMS/S-IVB separation occurs
00:13	The four +X RCS jets are turned off, and the crew begins pitching up ($+X_b$ down) to -r (down the radius vector), with the earth used as the visual reference to determine -r
01:00	The four -X RCS jets are turned on to initiate an evasive maneuver to provide clearance between the CSM and S-IVB for the abort maneuver
01:08	The four -X RCS jets are turned off, and the crew begins maneuvers to the abort maneuver thrusting attitude (fig. 7-7) after having driven to the following IMU gimbal angles OGA = 180° MGA = 0.0° IGA = ground computed in EPO
04:00	The crew selects the abort ΔV from a chart of ΔV versus V_i and S-IVB t_B and enters this value in the ΔV counter; the crew begins preparations for an SCS automatic maneuver
05:00	The COAS elevation angle is reset to 0°; the CDR adjusts his position in the couch to view the horizon through the COAS reticle image
09:30	The spacecraft is alined to the required horizon referenced attitude (fig. 7-7).
10:00	The SPS is ignited and the burn is controlled by SCS automatic

Although this time line has been recommended, the actual time line will be presented in the Apollo Abort Summary for Mission F to be prepared by the Crew Safety Section, Crew Safety and Procedures Branch, Flight Crew Support Division.

The charts that the crew will need onboard on launch day are presented in figures 7-8 and 7-9 which show abort ΔV measured along the X-body axis, SPS abort burn time, and time from SPS abort (SPS off) to entry as functions of inertial velocity at the time of abort. The figures are double scaled at the top and bottom to show S-IVB burn time, EMS ΔV during the Saturn burn, and inertial velocity. The S-IVB burn time and the EMS ΔV are required as the backup independent variables for determination of the abort ΔV.

The landing point loci are shown in figure 7-10 as a function of S-IVB burn time for three TLI's on the nominal day of launch when the abort ΔV's that are shown in figure 7-8 were applied at S-IVB cutoff plus 10 minutes. The ground elapsed time of continuous USBS tracking is shown in figure 7-11 as a function of inertial velocity at S-IVB cutoff. The altitude at which the CSM would be at initiation of the abort maneuver is shown in figure 7-12 as a function of the inertial velocity at S-IVB cutoff.

The ground control will provide the crew with the pitch gimbal angle (referenced to the launch pad REFSMMAT) for the crew to use for the initial attitude maneuver for the fixed attitude abort. The IGA at 10 minutes remains constant for the full range of TLI velocities. The IGA at the abort point is shown in figure 7-13 as a function of the launch azimuth for the planned day of launch.

The primary purpose of the fixed attitude abort is as stated previously: to return the crew to earth as rapidly as possible without regard to landing location. For the abort to be as insensitive as possible to execution errors, the maneuver is targeted to achieve the midcorridor or contingency entry target line, which is the same as the entry target line that is stored in the CMC. Therefore, subsequent midcourse corrections determined onboard will be targeted to the entry target line used to determine abort ΔV.

Two possible sources of execution errors, abort ΔV errors and pitch attitude errors, have been considered in this analysis, and their effects have been shown. Ignition time errors have proven to be the least sensitive (i.e., the effect of the errors are more tolerable) of the possible sources of execution errors. Previous studies have shown that ignition time can be off by as much as 1 minute and the maneuver still can achieve the entry corridor (ref. 5). The abort maneuver is very sensitive with respect to attitude errors for aborts performed after about 200 seconds into the TLI burn; however, past this time, sufficient time remains prior to entry to perform a midcourse correction maneuver to return to the entry target line. The tolerable pitch errors and ΔV errors for the abort maneuver execution are shown in figures 7-14 and 7-15 as a function of inertial velocity at S-IVB cutoff. The pitch error can be very large for early shutdowns, but an accuracy to within 1° is required for a fixed attitude abort after a nominal TLI cutoff.

From converstaions with Apollo crew members, it was found that the expected accuracy in pitch for the horizon reference attitude alinement is within $\pm 3^{\circ}$. Based on the expected accuracy, even if the TLI burn is nominal and the maneuver is performed at the correct ignition time and the correct abort ΔV is used, an MCC will be required for aborts that occur after about 200 seconds into TLI (fig. 7-16). Previous studies have shown that the required MCC is less than 150 fps for pitch errors that are less than 3° if the MCC is performed 1.5 hours from abort (ref. 5).

7.2.3 The 90-minute abort.- As stated previously, it has been recommended that, if possible, TLI should always be continued to nominal cutoff, at which time the ground controllers and the crew could perform a malfunction analysis to determine the necessity of an abort. If it is determined that an abort maneuver is required after TLI, the ground controllers and the crew will begin preparations that lead to an abort maneuver that would be performed approximately 90 minutes after TLI cutoff. The 90-minute time is not the time of actual SPS ignition but has been fixed primarily as input time of ignition for P37 (onboard return to earth abort program) if the crew is ever required to calculate the abort maneuver onboard; the ground computers would perform the same calculations to determine the CM landing point. The P37 will be used to enable the crew to return to earth if a critical subsystems failure occurs that would require an abort and if ground-to-air communications are lost. The criteria used to determine the 90-minute abort ΔV magnitude are as follows.

1. The abort trajectory returns to a CLA.

2. Return flight time does not exceed 18 hours from TLI cutoff to landing.

3. Abort ΔV does not exceed 8000 fps.

For the full range of possible abort ΔV's, the earth will always be in view at SPS ignition, but a small portion of the earth will be obscured by the lower right-hand side of the left forward viewing window. This is shown in figure 7-17.

For the nominal spacecraft trajectory, the 90-minute abort will require an abort ΔV of 7415 fps, and the resultant landing point will be in the Atlantic Ocean recovery area. The SPS ignition for this maneuver occurs 85.3 minutes after TLI cutoff or at $04^h02^m04^s$ g.e.t. for the TLI of the May 17 72° launch azimuth, first-opportunity mission.

Maneuver execution errors of less than 1° in pitch attitude for the 90-minute abort can cause the entry vector to lie outside the entry corridor. The MCC ΔV magnitude required to correct for execution errors is a function of three things: the time of the MCC, the magnitude of the error, and the purpose of the MCC. If the MCC is designed to retarget to the original preabort computed landing point, the magnitude grows as a function of delay time from SPS abort cutoff. If the MCC is designed to retarget to the entry corridor only, the optimum orbital position at which to perform the MCC is the apogee of the postabort trajectory. Thus, the optimum time from the MCC to the entry corridor is a function of postabort true anomaly, which in turn is a function of the abort ΔV.

The MCC ΔV required to achieve the entry corridor only is shown in figure 7-18 as a function of delay time from SPS abort cutoff for several pitch errors at the abort point. The MCC ΔV required to achieve the preburn-computed landing point is shown in figure 7-19 as a function of delay time from SPS abort cutoff for several pitch errors at the abort point. A region can be seen in figure 7-19 between approximately 3.5 hours from SPS abort to approximately 5.0 hours from SPS abort where it is impossible to land exactly on the AOL with a return time consistent with the nominal abort return time. Solutions to the AOL exist for return times a day later, but the ΔV requirements for these solutions greatly exceed the ΔV that remains (≈2600 fps). For the unspecified area solutions (fig. 7-18) for the errors considered, landings will occur within 60 n. mi. of the AOL.

7.2.4 Translunar coast aborts.- In EPO, prior to TLI, the ground controllers will pass to the crew two abort solutions based on a nominal TLI burn. The first solution, the 90-minute abort, would be used if a critical subsystem failed and if ground-to-air communications were lost after TLI. The second solution would be used if no critical subsystems failure had occurred but if ground-to-air communications could not be established after TLI and if the crew had determined that T&D could not be performed. In both instances, it is recommended that the crew retarget the abort maneuver onboard by use of P37 to account for any trajectory dispersions which might be induced by the S-IVB during TLI.

After TLI, the ground controllers periodically will provide abort solutions (block data) to the crew to be used in case of a total loss of communications for both the CSM and the LM. In these instances, P37 will be used for both the abort maneuver and for the MCC that follows the abort maneuver.

The block data solutions that are passed to the crew during TLC will be targeted to return the SC to the prime CLA located in the middle of the Pacific Ocean. However, targeting of the block data solutions to the prime CLA does not preclude the targeting of abort solutions to any of the four contingency areas that remain or the return of the crew to an unspecified water landing area if the situation warrants such action.

For abort maneuvers targeted to an unspecified area, the return time is a function of orbital position (delay time from S-IVB cutoff) and of the ΔV expended. This relationship is shown in figure 7-20, which presents the time from abort to entry as a function of the g.e.t. of abort for several abort ΔV's. The GETL is shown in figure 7-21 as a function of entry velocity and of the g.e.t. of abort for several abort ΔV's.

The thrust vector for the 90-minute abort is approximately 6° below the crew line of sight to the horizon or approximately 6° between the radius vector and the thrust vector with the earth in the window at SPS ignition. As the spacecraft moves farther out on the TLC, the angle between the thrust vector and the radius vector decreases. Also, the attitude difference decreases between very small ΔV abort maneuvers and very large ΔV abort maneuvers. After approximately 4 hours on the TLC, the angle between the thrust vector and the radius vector is about 2°, and the attitude difference between small ΔV maneuvers and large ΔV maneuvers is less than 1°. At the last block data abort point on TLC, the thrust vector is alined along the radius vector (fig. 7-22).

Because a small angle exists during TLC between the thrust vector and the radius vector, the earth can be used as a visual reference for the TLC return to earth maneuver. Also, because the attitude difference between the very small ΔV's and very large ΔV's is known to be very small, the abort targeting to contingency landing areas can be explained easily in terms of abort ΔV and return time. Suppose at some time on the TLC an abort solution is found which returns the SC to one of the five CLA's (fixed longitude); that solution will require an abort ΔV of x fps and will provide for a return of the SC in y hours. For the same delay time, several solutions exist that will return the SC to that same contingency area. If more ΔV is applied at nearly the same attitude, the return time is shortened; if less ΔV is applied, the return time is lengthened. To find the other solutions, the ΔV must be increased sufficiently to shorten the return flight time by exactly 24 hours or be decreased sufficiently to lengthen the flight time by 24 hours (fig. 7-23). For any given abort time, several solutions to the same CLA exist with a difference in return time of 24 hours.

In figure 7-24, the latitudes of landing are shown at which abort ΔV's are applied at various times if the solution achieves the contingency area. The RTCC displays data similar to the data presented in figures 7-23 and 7-24 to the flight controllers so that aborts can be planned and so that a first guess to subsequent abort processors can be obtained. After the final abort solution has been selected, the flight controller will generate digital information and a target load for each abort solution.

During TLC, if it is determined by the ground control and the crew that an abort maneuver is required, the maneuver will be performed based on the external ΔV targeting in the CMC. In table 7-I is presented representative information that would be voiced to the crew to support crew activities during the performance of the actual abort maneuver. Postabort groundtracks that would result from the use of three of the abort solutions shown in table 7-I are presented in figure 7-25. Postabort groundtracks from the 14 listed USBS sites are presented in figure 7-26. The contingency techniques defined in reference 1 indicate that planned direct return abort maneuvers during TLC coast would assume the LM would be jettisoned prior to the abort maneuver. It is also indicated in reference 1 that a planned jettison of the LM does not preclude a real-time decision for a direct return by use of the various available propulsion systems in the docked configuration. The enclosed data may be used to determine docked maneuver capability if it is assumed that an abort ΔV of 1800 fps represents the LM DPS capability and that an abort ΔV of 4500 fps represents the SPS docked capability. For both ΔV's, a sufficient ΔV reserve is assumed for subsequent midcourse corrections, although both ΔV's represent only 90 percent of the maximum ΔV capability.

8.0 LUNAR ORBIT INSERTION AND LUNAR ORBIT PHASE

8.1 Lunar Orbit Insertion Monitoring

Because LOI always occurs behind the moon, the crew must be able to evaluate the progress of the maneuver without ground support. Although two LOI burns are required to produce the desired 60-n. mi. altitude circular orbit, the monitoring requirements are defined primarily for the first burn (LOI-1), because the second burn (LOI-2) lasts for only approximately 15 seconds. The recommended LOI crew monitoring technique is generally depicted in figure 7-3 and is the same as for TLI.

For Apollo 10 (Mission F), the preignition SC attitude check (fig. 7-3) is made more difficult by the presence of the LM. However, it has been determined that the horizon and several stars should be visable from the CDR's rendezvous window and may be used as a backup to the optics for the orientation check prior to ignition. If the SC attitude is not within ±5° of nominal, the LOI should be NO-GO.

Although maintenance of crew safety is always the primary objective of monitoring procedures, an important second objective is the assurance that adequate abort capability is provided and is compatible with possible results of the monitoring procedures. The second objective was accomplished for LOI by definition of sound procedures for the four types of problems possible during LOI. Basically, the four problems areas are guidance and control, non-SPS systems, SPS, and inadvertent shutdowns. The recommended action for each problem is presented in table 8-I. A solution for the first type of problem (guidance and control) would be to have the crew take manual control of the PGNCS-controlled maneuver and to complete the LOI at the original ignition attitude (ref. 24). One of the most dangerous possibilities associated with guidance and control problems could occur if the spacecraft IMU drifts during LOI. The crew cannot detect a small drift until an attitude deviation builds up and appears on the secondary inertial attitude reference system. Because the drift could have occurred in the secondary reference system as well as in the IMU, the crew would be unable to distinguish the erroneous system without the SCS attitude error needles (a third inertial reference system) which provides a tie-breaking capability. This detection of the error would make possible a manual takeover and completion of the burn so that the SC could enter LPO. Because uncorrected IMU drifts in pitch can produce impact trajectories, rules were developed to define attitude limits for which a takeover should be initiated.

The rules and limits require a manual takeover with the SCS at an attitude deviation of 10°, exclusive of start transients; the purpose of the 10° deviation is to prevent an undesirable pericynthion. The effects on pericynthion of platform misalinement and constant drifts through LOI-1 are plotted in figure 8-1. Effects of takeover rules and limits are shown in figures 8-2 and 8-3 for various nodal altitudes with 60 n. mi. as the planned nominal altitude. A third inertial reference system is required during LOI to insure that the IMU does not cause an impact trajectory. Although there are three inertial reference systems in the spacecraft that could be used for LOI, an external reference system such as the lunar horizon or the stars may provide an additional reference system (ref. 25).

Note that omission of the fourth translunar midcourse maneuver so that tracking will be improved can produce dispersions in the approach hyperbola that affect the LOI manual takeover limits. Because the orientation of the desired lunar orbit can be rotated to accomodate for the approach hyperbolic dispersions, LOI can occur at different points in the desired lunar orbit. As a result, each drift during the LOI burn produces a different resultant orbit. To maintain the safety provided by the 10° attitude deviation takeover limits, the approach hyperbola dispersion should be removed until effects on the takeover limits are acceptable. It appears that the most meaningful way to limit the effects on the takeover limits of possible dispersions is to limit the nodal altitude to 65 n. mi. at the intersection of the approach hyperbola and the desired lunar orbit. This nodal altitude gives an acceptable pericynthion ($h_p \geq 20$ n. mi.) for the worst case represented by the lower half of the -10° pitch drift curve of figure 8-4. The bottom half of the -10° curve represents the effect that a negative drift has on a trajectory that nominally burns out after pericynthion; the top half of the curve represents the effect of a negative drift on a trajectory that burns out prior to pericynthion. The situation is reversed for the +10° curve. Additional details may be found in reference 26.

In summary, if the altitude at the node of the approach hyperbola and at the desired lunar orbital plane is greater than 65 n. mi. after MCC-3, then MCC-4 should be performed to preserve crew safety that is provided by the manual takeover limits.

As for TLI, the LOI rate limit is 10 deg/sec and results in a crew takeover and manual completion of LOI at ignition attitude.

Non-SPS problems require completion of LOI because it is advantageous to be in the planned lunar orbit rather than in any other orbit.

The SPS problems may dictate the necessity of an immediate abort maneuver which takes place 15 minutes after LOI ignition, after the crew terminates a nominal trajectory. Problems of this type are caused primarily by SPS problems which indicate that the SPS engine could have a limited burn time or maneuver capability. More specifically, serious SPS problems are as follows.

1. Sustained pressure decay in either fuel or oxidizer tank

2. Thrust chamber pressure lower than 70 psi

3. A delta pressure of greater than 20 psi between fuel and oxidizer tanks

Although built-in redundancy may require two failures before the problems are time critical, the desire to complete the large abort maneuver (approximately 3000 fps) as soon as possible to insure lunar sphere escape is the major justification for a 15-minute abort maneuver.

For inadvertent shutdowns, the crew will try a restart but if unsuccessful will prepare for a LM DPS abort.

Backup of the PGNCS LOI cutoff is performed by the crew primarily on a 9-second time bias to the nominal burn time. In summary, guidance and control problems during LOI result in crew takeover and in burn completion to near-nominal LOI conditions, from which an abort could be initiated. The SPS problems result in early shutdown of LOI and abort.

8.2 Aborts During LOI and Lunar Orbit

8.2.1 Introduction.- The LOI burn transfers the CSM/LM from the free-return translunar trajectory to the LPO. The transfer consists of two SPS burns of 352.5- and 15-second duration, respectively. After the LOI-1 burn, the SC coasts in a 60- by 170-n. mi. lunar orbit for two revolutions. The LOI-2 burn is initiated at the third pericynthion to achieve the 60- by 60-n. mi. lunar orbit.

Premature termination of the LOI maneuver places the vehicle in a nonnominal lunar orbit from which either an alternate mission or an abort situation may result. An early shutdown of the SPS engine may occur as a result of two situations:

1. An inadvertent SPS shutdown

2. Manual shutdown by the crew

If an inadvertent SPS shutdown occurs early in the LOI-1 burn, the recommended procedure is to initiate an immediate SPS restart. If the restart is unsuccessful and an abort situation exists, the LM DPS engine is primary for the abort maneuver.

Manual SPS shutdown would occur only if critical SPS systems problems occur which would severely restrict the future performance of the engine (section 8.1). After a manual SPS shutdown, which occurs prior to 3^m00^s into the LOI-1 burn, the SPS is primary for the abort (the DPS is available as a backup). For the remainder of the LOI burn, the DPS is primary. Discussions of the abort modes and detailed procedures are contained in references 1 and 27.

8.2.2 Characteristics of lunar trajectories resulting from premature LOI shutdown.- The lunar orbits that result from premature termination of LOI-1 can be classified in three categories.

1. Hyperbola: LOI-1 burn 0^m00^s - 1^m50^s

2. Unstable ellipse: LOI-1 burn 1^m50^s - 2^m50^s

Trajectories of this type are greatly perturbed by the earth's attraction. The earlier shutdowns result in extremely long orbital periods. Later shutdowns have periods as low as 24 hours but impact the lunar surface prior to pericynthion.

3. Stable ellipses: LOI-1 burn 2^m50^s - end LOI-2

The conic parameters of LOI termination are shown in figure 8-5 as functions of SPS burn time. Based on systems limitations, terminations that result in stable ellipses will result in either lunar alternate missions or abort situations. Unless a corrective maneuver is made to provide a clear pericynthion, terminations prior to 2^m50^s necessitates an abort.

8.2.3 General abort modes.- Lunar phase abort maneuvers for the F mission are of three basic types.

1. Mode I - a one-impulse maneuver that returns the spacecraft directly to earth. The burn is initiated as soon as possible after LOI termination to reduce the necessary ΔV.

2. Mode II - a two-impulse maneuver that necessitates one lunar orbit. The first impulse is directed down the radius vector and is initiated as soon as possible. The burn reduces the orbital period and provides a stable intermediate lunar orbit. The second burn occurs near pericynthion and injects the spacecraft on the transearth trajectory.

Mode III - a one-impulse maneuver initiated near pericynthion after one or more orbits. This burn is similar to the normal TEI burn.

8.2.4 Abort ground rules.- The abort ground rules for LOI aborts are as follows. Burn monitoring is discussed in section 8.1.

1. If an inadvertent LOI-1 termination occurs, an immediate restart will be attempted, and the LOI burn will be continued. If the SPS does not reignite, the DPS will provide return to earth capability through the use of the following RTCC targeted abort modes.

LOI burn duration, min:sec	Mode	t_{IG}
0:00 - 1:45	I (DPS)	LOI_{IG} plus 2 hr
1:45 - 2:50	II (DPS)	LOI_{IG} plus 2 hr
2:50 - 5:52.5 (end of LOI-1)	III (DPS)	Near pericynthion

2. If a manual SPS shutdown is required (propellant pressure problems), the following criteria should be used to determine the abort mode.

a. LOI burns 0^m00^s to 2^m50^s - a mode I abort maneuver should be initiated at LOI_{IG} plus 15 minutes based on data from a crew chart

b. LOI burns 2^m50^s - end LOI-1 - a mode III DPS burn based on an RTCC solution

The 15-minute crew chart is available from 0^m00^s to 4^m00^s into the burn. However, at 2^m50^s into burn, the preabort orbit is a stable ellipse with a period of 15 hours. At this point, a LM abort would not require an immediate LM activation. The crew chart is discussed in section 8.2.9. If the 15-minute abort is not performed, the abort modes would be the modes that are presented in ground rule 1.

8.2.5 Mode I abort requirements.- The mode I abort region is defined as the class of preabort trajectories from which a one-impulse abort maneuver initiated as soon as possible will result in earth return. The constraints are time of ignition and abort ΔV. The abort ΔV available during the LOI burn is shown in figure 8-6 for an SPS burn, CSM only; for an SPS burn, CSM/LM; and for a DPS burn, CSM/LM.

The fuel-critical, unspecified area (FCUA) mode I abort requirements are shown in figure 8-7(a) for several times of abort initiation. The required abort ΔV increases rapidly as the abort is delayed from LOI termination. The g.e.t. of landing for these FCUA aborts is presented in figure 8-7(b). However, rather than return to an FCUA, a return to the prime landing area would be more acceptable if it were available. The mode I abort ΔV for returns to the MPL is shown in figures 8-8(a) through (c) at a GETL of 118 hours, 142 hours, and 166 hours, respectively, for various delay times.

8.2.6 Mode II abort requirements.- The mode II abort occurs in a region in which the required mode I abort ΔV exceeds the available ΔV and in which a stable lunar ellipse has not yet been achieved. The mode II abort consists of an initial DPS burn (corrective maneuver) to provide a stable intermediate ellipse and a second DPS burn to inject the spacecraft on the transearth coast. The corrective maneuver consists of a variable ΔV magnitude (ΔV_1) that is directed down the radius vector. The time of ignition is nominally LOI_{IG} plus 2 hours.

The nominal value of ΔV_1 is shown in figure 8-9(a) as a function of LOI burn time. Two additional lines are included in the figure. One line defines the maximum time that the corrective maneuver could be delayed before a lunar impact would result from the nominal value of ΔV_1. The other line indicates the maximum delay allowed for the corrective maneuver so that the time between burns is kept below 40 hours with the ΔV_1. This time constraint is important because of the LM DPS supercritical helium pressurization limits. The time between burns for a nominal ΔV_1 applied at LOI_{IG} plus 2 hours and LOI_{IG} plus 5 hours is shown in figure 8-9(b) as a function of LOI burn time.

The total ΔV requirements ($\Delta V_1 + \Delta V_2$) are shown in figure 8-9(c) for FCUA and MPL returns based on an initial value of ΔV_1 obtained from figure 8-9(a). The g.e.t. of landing for the FCUA solution is included in figure 8-9(c).

The mode II data presented in this section are the result of current analyses of the F mission operational trajectory performed by the Flight Analysis Branch. Parametric data of a more general nature will be published in the near future.

8.2.7 Mode III abort requirements.- A mode III abort can be initiated if the preabort trajectory is a stable lunar ellipse. The single abort burn is initiated near pericynthion after one or more orbits. The abort ΔV required for FCUA and MPL returns is shown in figure 8-10(a) as a function of LOI-1 burn time. The g.e.t. of landing of the MPL returns is indicated in the figure; the g.e.t. of landing of the FCUA returns is shown in figure 8-10(b).

8.2.8 DPS abort capability as a function of LOI burn duration.- The general ΔV requirements for the three abort modes were presented in sections 8.2.5 through 8.2.7. The purpose of this section is to summarize the abort mode availability throughout the LOI-1 burn, specifically for DPS aborts. The FCUA abort ΔV is shown in figure 8-11(a) as a function of LOI burn time for the three abort modes. Of primary importance is the fact that DPS backup abort capability exists throughout the LOI-1 burn.

Although a complete DPS backup capability exists for FCUA aborts, the available DPS ΔV is not sufficient to permit returns to the MPL after a complete LOI burn. The abort ΔV required for minimum fuel MPL returns is shown in figure 8-11(b). Because of the increased ΔV requirements, a gap of 6 seconds in the LOI burn exists where returns to the MPL are not available. In this small region of LOI burn time, returns would be targeted to another CLA or at least to a water landing.

The time of ignition of the mode I abort and mode II corrective maneuver is $LOI\text{-}1_{IG}$ plus 2 hours, which is considered the earliest possible time of abort. The maximum allowable delay time from LOI-1 ignition to abort initiation for FCUA aborts is shown in figure 8-12. The mode I data are constrained by both maximum LM DPS ΔV available and by the maximum ΔV based on a 5 percent reserve. The mode II data are constrained by the time between burns (section 8.2.6). Although the mode I abort ΔV increases rapidly with delay time, the mode II ΔV is relatively constant. However, the primary constraint on delay time is the time between burns and pericynthion altitude of the intermediate ellipse.

All the abort data shown in section 8.2 have been simulated by impulsive burns and patched-conic trajectories. The mode I/mode II overlap region in figure 8-13 compares patch-conic and integrated transearth coasts for FCUA returns. Although a small difference in required ΔV exists between patch-conic and integrated trajectories, the conic results are sufficiently accurate for parametric analyses. The mode III requirements at the end of the LOI-1 burn and the entire LOI-2 burn based on integrated trajectories are shown in figure 8-14.

8.2.9 LOI crew chart for SPS aborts that follow manual LOI shutdown.- If SPS problems occur during LOI which indicate an approaching SPS failure (e.g., a decay in the propellant tank pressures), the recommended procedure is to shut down the SPS engine manually. For Apollo 8, the subsequent crew action was an SPS restart after a coast of 15 minutes; data were used from a crew chart supplied by the Flight Analysis Branch.

For the F mission, the SPS ΔV available in the docked configuration is such that the 15-minute abort is available only for a portion of the LOI burn. The applicable region of the LOI burn is discussed in the following paragraphs. The F mission 15-minute LOI crew chart is presented in figure 8-15.

Basically, the procedure to be followed for use of the 15-minute crew chart is as follows.

1. After manual SPS shutdown, the crew maneuvers the CSM/LM combination to a set of gimbal angles that are determined from the CM IMU orientation. The gimbal angles are contained on the chart and are the same regardless of LOI burn duration.

2. The abort ΔV magnitude is read from the chart (fig. 8-15) and is a function of the LOI DVM read from the DSKY after shutdown. The burn time can be used as a backup.

3. An SPS SCS burn is initiated at LOI-1_{IG} plus 15 minutes. Therefore, a constant time of ignition results.

4. If the SPS cannot be restarted, the normal DPS abort procedure is followed.

The 15-minute abort is recommended for LOI terminations that occur prior to 2^m50^s although the chart will be extended to 4^m00^s, at which time the abort ΔV is approximately 500 fps less than the available SPS ΔV. At 2^m50^s into the LOI burn, a stable lunar ellipse with a period of 15 hours has been achieved. The preabort periods in this region of the LOI burn are shown on the crew chart. For the remainder of the LOI burn, a mode III DPS abort can be initiated after one revolution, and immediate LM activation is not required. If the chart is used at 2^m50^s into LOI, the total SPS burn time (LOI plus abort) is approximately the same as if the LOI burn had been continued to nominal termination.

By the use of a technique developed for the Apollo 8 mission, the 15-minute crew chart was prepared to provide relatively constant landing locations and entry conditions regardless of LOI burn duration. For all aborts, the entry velocity is 36 140 fps, and the GETL is 150 hours. The landing latitude and longitude are shown in table 8-II. The gimbal angles on the crew chart correspond to the IMU REFSMMAT (table 8-II).

8.2.10 Crew chart midcourse requirements.- The LOI crew chart discussed in the previous section was constructed based on the nominal F mission operational trajectory. If the launch date is changed or if significant trajectory dispersions occur during the transearth coast, it is possible that the crew chart could require larger MCC's than are desirable. For this reason, the crew chart will be verified and updated if necessary during the final hours of the transearth coast prior to LOI (as in the Apollo 8 mission).

However, for the nominal LOI burn, an MCC may be required if certain execution errors occurred during the 15-minute abort. Specifically, errors in time of ignition, pitch, yaw, and abort ΔV magnitude should be considered. The midcourse ΔV required at the lunar sphere of influence is shown in figure 8-16. Generally, the midcourse ΔV is larger for aborts that are initiated later in the LOI burn.

Although the mode III region becomes available at 2^m50^s into the LOI burn so that the 15-minute abort is not required (because the DPS abort is no longer time critical), the chart extends to 4^m00^s into the LOI burn. For this extreme case, the allowable execution errors within the SM RCS midcourse capability (SM RCS ΔV of 100 fps is assumed) are as follows.

1. t_{IG} error = ±44 sec

2. Pitch error = ±3.3°

3. Yaw error = ±3.6°

4. ΔV error = ±60 fps

8.2.11 Sample abort solutions.- Sample abort points and detailed trajectory listings are presented in reference 28.

8.3 Docked DPS Burn Monitoring

Use of the DPS to perform an abort from lunar orbit has been described previously. Burn monitoring procedures for such a maneuver should be the same as described in reference 29 with the following exceptions. For a TEI maneuver, the attitude deviation and the rate limits should be widened to 20° and 10 deg/sec, respectively; thrust transients are excluded because of the critical nature of such a burn.

The MCC ΔV for several attitude drifts is shown in table 8-III through the docked DPS abort burn. Because the LM control systems were designed to perform docked burns with the PGNCS only and because it is not known at present if the CSM/LM can be controlled manually, the burn should be terminated after the attitude deviation and rate limits are reached. These wide limits provide maximum opportunity to complete the maneuver as well as provide time for the PGNCS to aline the thrust through the center of gravity. If a docked manual control capability can be identified, the crew should complete the abort maneuver rather than shut down the engine. The MCC ΔV for several attitude drifts during a mode II lunar abort are shown in table 8-IV. The ΔV cost is reasonably low.

9.0 TRANSEARTH INJECTION AND TRANSEARTH COAST PHASE

9.1 Transearth Injection Monitoring

Both TEI and LOI occur behind the moon, and the monitoring procedures and techniques for both are basically the same. The major difference is that for TEI the guidance, control, and systems problems will all require a continuation of the maneuver; that is, guidance and control problems result in crew takeover and burn completion at the ignition attitude, while SPS or spacecraft systems problems are ignored until the important TEI maneuver is completed. A backup to the PGNCS TEI cutoff will be performed by the crew at 2 seconds past nominal time, and confirmation that the desired cutoff velocity has been achieved will be shown by the EMS ΔV counter. Inadvertent terminations during TEI will be restarted if possible within approximately 30 seconds, or a ground solution will be required for a later abort attempt. Because abort targeting implies severe SPS problems and because a communications failure would be required before an onboard backup is needed, the extensive preflight effort to generate TEI crew charts is unwarranted.

Manual takeover of the TEI maneuver will occur when, as in LOI, the crew confirms a deviation from the fixed inertial burn attitude by two independent references. A rate limit of 10 deg/sec will require immediate takeover, rate damping, and burn completion. The attitude deviation limit was selected with the aid of figure 9-1, which shows the MCC required for maneuvers controlled by a drifting PGNCS platform. It is seen that a drift which produces a 10° attitude change by the end of the 160-second maneuver (fast return) requires an MCC of approximately 255 fps. The RCS capability at this point in the mission is approximately 175 fps, which means that the next MCC would have to be made with the SPS. This alteration is acceptable because the 10° limit can be used easily for both LOI and TEI and because the probability is low that a platform drift will occur. Also, if the slower return time is used with TEI, then the 10° limit falls within the RCS.

For consistency, any SPS abort maneuver will be made with the identical procedures used during TEI, which is in accordance with the time-critical nature of execution of abort maneuvers. During TLC, an abort that uses up to 7000 fps may be required, while lunar phase aborts generally require approximately 3000 fps. Even though the previously described takeover limits can result in large MCC's, smaller limits will probably still require an SPS MCC. Also, the simplicity of having one monitoring procedure for all SPS burns is an important consideration, especially for the flight crew.

9.2 Aborts During TEI and Transearth Coast

9.2.1 Introduction.- The TEI burn transfers the spacecraft from the 60-n. mi. altitude circular LPO to the TEC. The transfer consists of a single SPS burn of approximately 160 seconds and results in a ΔV of 3251 fps.

To reiterate the philosophy of TEI burn monitoring: completion of the TEI burn is mandatory; that is, a manual shutdown will not be initiated for any CSM systems problem. If an early automatic SPS shutdown occurs, an immediate restart will be attempted. Only if immediate reignition is not possible will an RTCC abort solution be required. Therefore, because abort targeting implies severe SPS problems and because an additional failure of communications would be required before an onboard backup is needed, the extensive preflight effort to generate TEI crew charts is unwarranted.

In the following paragraphs, general parametric data of abort ΔV and total flight times are included to illustrate the possible tradeoffs that can be made in the final selection of the RTCC abort solution.

9.2.2 Characteristics of lunar trajectories that result from premature TEI termination.- The description of the three classes of trajectories made in section 8.2.2 applies here, with the exception of the respective TEI burn times.

1. Class 3 stable ellipses, TEI ignition to 90 seconds

2. Class 2 unstable ellipses, 90 seconds to 120 seconds

3. Class 1 hyperbola, 120 seconds to nominal TEI termination

The conic parameters at TEI termination are shown in figure 9-2 as a function of SPS burn time.

9.2.3 Abort modes.- The description of the lunar phase abort maneuvers in section 8.2.3 applies here.

9.2.4 Abort ground rules.- If an automatic SPS shutdown occurs prematurely and an immediate SPS reignition is not possible, the following abort criteria will be followed. The SPS is assumed to be subsequently available.

1. TEI burn time - 00^m00^s to 1^m30^s. Because a stable lunar ellipse exists, a mode III RTCC abort will be initiated.

2. TEI burn time - 1^m30^s to 2^m00^s. The preabort trajectory is an unstable ellipse and a corrective maneuver is required; therefore, a mode II RTCC abort that requires two SPS burns is initiated.

3. TEI burn time - 2^m00^s to end TEI. The preabort trajectory is a hyperbola and a mode I abort is required.

It was stated in the previous TEI discussion that crew charts are unwarranted because several CSM system failures must occur before they would be needed. However, an important onboard backup is still available. After a TEI burn greater than 138 seconds in duration, the SC will exit the MSI. The onboard return to earth program (P37) is now available to calculate the return to earth maneuver. The high ΔV requirements would be for a termination at 138 seconds because at this time the SC is in the lowest energy ellipse of the region. However, for this case, the $\Delta V = 2400$ fps and the TFT = 100 hours, which is well within the available ΔV of figure 9-3.

9.2.5 Mode I abort requirements.- The discussion in section 8.2.5 of LOI mode I aborts applies here. The FCUA abort requirements are shown in figure 9-4(a) as a function of TEI burn time for various times of abort ignition. The GETL for these aborts is shown in figure 9-4(b). The abort ΔV required to return to the MPL at the nominal time of landing (GETL = 192 hours) is shown in figure 9-5.

9.2.6 Mode II abort requirements.- The mode II abort is required because for this range of TEI termination times the mode I abort is not available ($t_{IG} = TEI_{IG}$ plus 2 hr is assumed). In the TEI abort situation, two SPS burns are required (section 8.2.6). The nominal value of ΔV_1 is shown in figure 9-6(a) as a function of TEI burn time. An additional line on the figure shows the maximum time at which the nominal ΔV_1 could be applied to keep pericynthion above 40 n. mi.

As the first burn is delayed past 2 hours from TEI_{IG}, the intermediate period increases [fig. 9-6(b)]. For the nominal ΔV_1 at TEI_{IG} plus 2 hours, the total abort ΔV ($\Delta V_1 + \Delta V_2$) for FCUA and MPL returns is shown in figure 9-6(c). The GETL for the FCUA returns is shown in figure 9-6(d).

9.2.7 Mode III abort requirements.- The abort ΔV required for FCUA and MPL aborts for terminations in the mode III region is shown in figure 9-7(a). The GETL for the FCUA aborts are presented in figure 9-7(b).

9.2.8 SPS abort capability as a function of TEI burn duration.- In the previous three sections, the abort requirements for the three modes of abort were discussed. The purpose of this section is to summarize the availability of an SPS abort capability for the entire TEI burn. The FCUA abort capability for all three modes of abort is shown in figure 9-8(a). The mode I abort and mode II first burn are initiated at TEI_{IG} plus 2 hours. The FCUA abort capability is seen to exist throughout the burn. The effect on this abort capability when MPL returns are desired is shown in figure 9-8(b). The net effect is an increase in abort requirements and a reduction in the availability of the three abort modes.

An important fact to be considered here is that mode II aborts were definitely required for LOI DPS backup because a minimum time constraint existed (2 hr from LOI_{IG}) for the DPS ignition. In that case, time was required to activate and aline the LM systems in addition to the normal time required to target the RTCC abort maneuver.

Because the mode I abort initiated at LOI_{IG} plus 2 hours did not overlap the mode III region because of high ΔV requirements, the mode II abort was required for LOI backup.

Although the Mode I and mode II aborts are nominally scheduled at TEI_{IG} plus 2 hours, the constraint on the time of ignition is not as restrictive as that for LOI aborts. The maximum allowable delay time from TEI_{IG} to abort is shown in figure 9-9 for premature TEI terminations in the mode I/mode II overlap region (FCUA returns). From the figure, it can be seen that the earliest required time of ignition is 4.6 hours after TEI ignition.

Finally, it should be reiterated that under no circumstances would the TEI burn be terminated unless a total SPS engine or control system failure occurred; that is, the SPS problems would have to be corrected before a subsequent TEI abort could occur. The previous sections have indicated that if a total failure occurred, complete TEI abort capability exists.

9.2.9 Transearth coast aborts.- Other than small MCC's, aborts during the TEC only would be initiated if a faster than nominal earth return is required. The reduction in the available return time is discussed in appendix C.

10.0 CONCLUSIONS

A continuous method has been defined for safe return of the flight crew to earth for the Apollo 10 mission, with or without ground control help. The rationale and supporting data are contained in this operational abort plan. The supporting data consist primarily of (1) maneuver monitoring techniques and limits used for protection against known constraints, and (2) abort trajectory data produced by computer simulations of the recommended abort procedures.

10.1 Launch Phase

Although continuous suborbital abort capability is provided during the launch phase, the primary objective, besides to provide for crew safety, is to continue to orbit. The orbit continuation can be accomplished when early S-IVB staging capability becomes available, when the S-II is burning, and when SPS COI capability becomes available during the first S-IVB burn.

10.2 TLI and Translunar Coast

The postabort trajectories that result from early S-IVB shutdown and the 10-minute abort procedure may result in land landings. Based on the expected inaccuracies in the attitude alinement for the 10-minute abort, an MCC will be required for aborts that occur after approximately 200 seconds into TLI.

All return to earth maneuvers from the translunar coast mission phase are initiated at an attitude which causes the earth to appear in the CDR's window.

The SM RCS provides a backup capability to return the SC to earth after premature S-IVB shutdowns during TLI for most of the TLI burn.

10.3 LOI and Lunar Orbit

A complete return to earth capability exists for premature shutdowns during the LOI burn as well as during the nominal lunar orbit phase. The LM DPS provides a backup abort capability through the entire LOI burn, and the LM communications backup to the CSM insures that necessary ground targeting will be available.

10.4 TEI and Transearth Coast

Shutdowns during the TEI burn can occur only as a result of inadvertent automatic shutdown because manual shutdowns are not required. Immediate SPS restarts will be initiated. The only time an abort is required is when an immediate SPS restart is not possible, which implies serious SPS problems. Because communications failures would also have to occur in addition to very serious SPS problems, backup crew charts are not warranted.

During the TEC, an abort can shorten the return time if CSM system problems occur. The primary constraint is the maximum entry velocity possible.

TABLE 5-I.- LAUNCH SEQUENCE AND EVENT TIMES FOR

APOLLO 10 (MISSION F) LAUNCH[a]

Time, min:sec, g.e.t.	Event description
-20:00	Arm the LEV, begin mode IA
00:00	First motion
00:42	Begin mode IB
01:45	Fixed time abort 1 (mode IB)
01:56	100 000-ft. altitude, begin mode IC
02:15	Cut off S-IC center engine (TB-2)
02:40.4	Cut off S-IC outboard engine (TB-3)
02:41.2	S-IC/S-II physical separation
02:42.8	S-II ignition
03:00	Fixed time abort 2 (mode IC)
03:10.9	Jettison S-II aft interstage
03:16.6	Jettison launch escape tower, begin mode II
04:30	Fixed time abort 3 (mode II)
06:00	Achieve S-IVB early staging orbit capability
08:14.6	LOX low-level sense arm, (TB-3 + 334.2)
08:37	Begin mode IV COI (two-impulse)
08:49.3	S-II engine cutoff (TB-4)
08:50.1	S-II/S-IVB physical separation
08:53.3	S-IVB ignition

[a]Launch sequence for 72° launch azimuth

TABLE 5-I.- LAUNCH SEQUENCE AND EVENT TIMES FOR

APOLLO 10 (MISSION F) LAUNCH[a] - Concluded

Time, min:sec, g.e.t.	Event description
09:10	Fixed time abort 4 (mode II)
09:41	Begin mode III (no burn)
09:50	Begin no voice/no G&N mode III procedure, $t_b = 2(t_a - 590)$
09:56	Begin mode IV COI (one-impulse)
10:04	Begin mode III (with burn)
11:14.0	S-IVB first guidance cutoff signal
11:24.0	Earth orbit insertion

[a]Launch sequence for 72° launch azimuth

TABLE 5-II.- SC CONSTANTS USED FOR APOLLO 10 (MISSION F)

LAUNCH ABORT STUDIES[a]

Launch pad	39A
Launch pad geodetic latitude, deg N	28.608421
Launch pad longitude, deg W	-80.604132
SPS thrust, lb	20 560
SPS I_{sp}, sec	314.8
SPS thrust vector pitch offset (c.g. at launch), deg	3.629
Mode III landing target range ($\Delta R = 0$), n. mi.	3350
Mode III target geodetic latitude, deg N	26.48
Mode III target longitude, deg W	-17.05
Mode IV target apogee altitude for two-impulse, n. mi.	200
Mode IV minimum target perigee altitude, n. mi.	75
Mode IV minimum altitude during burn, n. mi.	75
Average time from drogue to main chute deployment for launch aborts, sec	308
Operational ΔV pad above hp = 75 n. mi. for COI, fps	100

[a]Additional constants which were used for these launch abort studies are presented in appendix A.

TABLE 5-III.- TYPICAL DSKY PARAMETERS DURING LAUNCH FOR THE APOLLO 10 (MISSION F)

Time, min:sec, g.e.t.	Inertial velocity, fps	Altitude, n. mi.	Altitude rate, fps	SPLERROR,[a] n. mi.	Predicted perigee, n. mi.	Predicted apogee, n. mi.	Predicted time of free fall[b] to 300 000 ft, min:sec	Pitch gimbal angle, deg
0:00	1 342	0.0	0	-3 350.0	-3 436.7	0.0	-59.59	90.00
0:10	1 344	0.1	68	-3 350.0	-3 436.7	0.1	-59.59	90.00
0:20	1 355	0.3	169	-3 350.0	-3 436.7	0.4	-59:59	89.00
0:30	1 390	0.7	295	-3 350.0	-3 436.6	0.9	-59:59	86.39
0:40	1 471	1.3	446	-3 349.9	-3 436.3	1.8	-59:59	81.90
0:50	1 618	2.2	624	-3 349.7	-3 435.6	3.1	-59:59	75.75
1:00	1 846	3.4	825	-3 349.4	-3 434.3	5.1	-59:59	68.64
1:10	2 158	4.9	1 038	-3 348.6	-3 432.1	7.6	-59:59	61.20
1:20	2 557	6.8	1 257	-3 347.4	-3 428.6	10.8	-59:59	53.85
1:30	3 050	9.0	1 485	-3 345.7	-3 423.0	14.7	-59:59	47.67
1:40	3 638	11.7	1 723	-3 342.9	-3 414.7	19.4	-59:59	41.96
1:50	4 327	14.7	1 961	-3 339.3	-3 402.5	24.8	-59:59	36.64
2:00	5 125	18.1	2 198	-3 334.5	-3 385.1	31.0	-59:59	32.53
2:10	6 045	22.0	2 446	-3 328.3	-3 360.4	38.2	-59:59	29.04
2:20	6 987	26.2	2 654	-3 321.6	-3 329.4	45.7	-59:59	25.88
2:30	7 915	30.7	2 814	-3 018.0	-3 292.9	53.2	-2:20	23.43

[a]SPLERROR = R_{TO-GO} (distance from current position to target - perigee greater than 300 000 ft).

[b]Time of free fall = POSMAX (-59:59) - perigee greater than 300 000 feet.

TABLE 5-III.- TYPICAL DSKY PARAMETERS DURING LAUNCH FOR THE APOLLO 10 (MISSION F) - Continued

Time, min:sec, g.e.t.	Inertial velocity, fps	Altitude, n. mi.	Altitude rate, fps	SPLERROR, n. mi.	Predicted perigee, n. mi.	Predicted apogee, n. mi.	Predicted time of free fall to 300 000 ft, min:sec	Pitch gimbal angle, deg
2:40	8 981	35.5	3 000	-2 995.3	-3 243.8	61.9	-3:03	21.46
[a]2:41.2	8 996	36.1	2 977	-2 993.0	-3 242.2	62.1	-3:03	21.51
2:50	9 102	40.3	2 807	-2 972.3	-3 233.6	63.6	-3:01	21.86
3:00	9 266	44.8	2 632	-2 947.7	-3 221.3	65.4	-3:01	21.78
3:10	9 444	49.0	2 465	-2 925.2	-3 208.3	67.2	-3:01	21.77
[b]3:16.6	9 571	51.0	2 357	-2 902.5	-3 199.0	68.4	-3:00	21.77
3:20	9 637	52.9	2 304	-2 895.9	-3 194.4	69.0	-3:00	21.77
3:30	9 840	56.6	2 161	-2 874.1	-3 179.9	70.8	-3:00	28.17
3:40	10 047	60.1	2 042	-2 850.7	-3 165.1	72.9	-3:01	27.38
3:50	10 265	63.3	1 924	-2 826.6	-3 149.4	74.8	-3:02	26.34
4:00	10 494	66.4	1 809	-2 801.9	-3 132.7	76.7	-3:03	25.27
4:10	10 734	69.3	1 699	-2 775.1	-3 113.0	78.4	-3:03	24.26
4:20	10 985	72.0	1 588	-2 750.3	-3 095.6	80.0	-3:04	23.25
4:30	11 248	74.6	1 482	-2 723.2	-3 075.2	81.6	-3:04	22.25
4:40	11 522	76.9	1 380	-2 695.2	-3 053.3	83.1	-3:05	21.25

[a]S-IC/S-II separation.

[b]Launch escape tower jettison.

TABLE 5-III.- TYPICAL DSKY PARAMETERS DURING LAUNCH FOR THE APOLLO 10 (MISSION F) - Continued

Time, min:sec, g.e.t.	Inertial velocity, fps	Altitude, n. mi.	Altitude rate, fps	SPLERROR, n. mi.	Predicted perigee, n. mi.	Predicted apogee, n. mi.	Predicted time of free fall to 300 000 ft, min:sec	Pitch gimbal angle, deg
4:50	11 808	79.1	1 280	-2 666.2	-3 029.8	84.5	-3:05	20.24
5:00	12 105	81.2	1 185	-2 636.2	-3 004.6	85.8	-3:05	19.21
5:10	12 415	83.1	1 093	-2 604.9	-2 977.5	87.0	-3:06	18.17
5:20	12 737	84.8	1 005	-2 571.8	-2 948.4	88.1	-3:06	17.12
5:30	13 071	86.4	921	-2 538.4	-2 916.9	89.2	-3:06	16.04
5:40	13 419	87.9	840	-2 502.8	-2 883.0	90.2	-3:07	14.95
5:50	13 781	89.2	764	-2 465.6	-2 846.3	91.1	-3:08	13.84
6:00	14 152	90.4	693	-2 426.3	-2 806.4	92.0	-3:08	12.74
6:10	14 548	91.5	626	-2 384.8	-2 763.1	92.7	-3:10	11.59
6:20	14 956	92.5	561	-2 338.8	-2 714.3	93.4	-3:11	10.39
6:30	15 379	93.4	506	-2 293.9	-2 664.4	94.1	-3:13	9.26
6:40	15 820	94.2	453	-2 243.7	-2 607.8	94.7	-3:15	8.07
6:50	16 280	94.9	406	-2 189.5	-2 545.6	95.3	-3:17	6.89
7:00	16 759	95.6	366	-2 130.7	-2 477.0	95.8	-3:21	5.69
7:10	17 259	96.2	331	-2 066.2	-2 400.9	96.3	-3:25	4.49
7:20	17 781	96.7	304	-1 994.8	-2 316.2	96.7	-3:30	3.28

TABLE 5-III.- TYPICAL DSKY PARAMETERS DURING LAUNCH FOR THE APOLLO 10 (MISSION F) - Continued

Time, min:sec, g.e.t.	Inertial velocity, fps	Altitude, n. mi.	Altitude rate, fps	SPLERROR, n. mi.	Predicted perigee, n. mi.	Predicted apogee, n. mi.	Predicted time of free fall to 300 000 ft, min:sec	Pitch gimbal angle, deg
7:30	18 326	97.2	283	-1 914.7	-2 221.4	97.2	-3:37	2.06
7:40	18 898	97.7	271	-1 821.7	-2 114.7	97.6	-3:45	0.84
7:50	19 496	98.1	267	-1 717.7	-1 994.1	98.1	-3:56	359.64
8:00	20 124	98.6	274	-1 591.9	-1 856.5	98.6	-4:10	358.42
8:10	20 668	99.0	269	-1 469.5	-1 727.1	99.1	-4:24	357.89
8:20	21 177	99.5	270	-1 348.6	-1 596.5	99.6	-4:41	357.05
8:30	21 708	100.0	274	-1 204.0	-1 449.5	100.2	-5:02	355.05
8:40	22 261	100.4	284	-1 022.5	-1 283.3	100.8	-5:30	353.94
8:50	22 806	100.9	300	-795.8	-1 105.4	101.6	-6:08	352.82
[a]8:50.0	22 806	100.9	299	-795.6	-1 105.4	101.6	-6:08	352.80
9:00	22 897	101.4	252	-750.7	-1 072.9	101.7	-6:12	352.34
9:10	23 076	101.8	220	-661.7	-1 010.1	101.9	-6:16	350.40
9:20	23 258	102.2	187	-562.4	-942.9	102.1	-6:26	348.72
9:30	23 443	102.4	155	-451.9	-873.2	102.2	-6:38	347.30
9:40	23 632	102.7	126	-336.1	-800.3	102.3	-6:51	345.99
9:50	23 823	102.9	99	-203.0	-723.9	102.4	-7:08	344.86

[a]S-II/S-IVB separation.

TABLE 5-III.- TYPICAL DSKY PARAMETERS DURING LAUNCH FOR THE APOLLO 10 (MISSION F) - Concluded

Time, min:sec, g.e.t.	Inertial velocity, fps	Altitude, n. mi.	Altitude rate, fps	SPLERROR, n. mi.	Predicted perigee, n. mi.	Predicted apogee, n. mi.	Predicted time of free fall[b] to 300 000 ft, min:sec	Pitch gimbal angle, deg
10:00	24 018	103.0	74	-48.5	-643.7	102.5	-7:29	343.76
10:10	24 216	103.1	54	76.8	-560.9	102.6	-7:56	342.67
10:20	24 417	103.2	35	362.4	-470.7	102.6	-8:31	341.65
10:30	24 522	103.3	21	655.3	-377.5	102.6	-9:49	340.62
10:40	24 830	103.3	10	1 061.6	-279.0	102.6	-10:30	339.58
10:50	25 041	103.3	-2	1 676.0	-175.1	102.6	-12:23	337.19
11:00	25 256	103.3	-8	2 885.1	-65.3	102.6	-16:20	337.71
11:10	25 473	103.3	-4	[a]-2 019.2	50.8	102.6	[b]-59:59	337.99
[c]11:14.039	25 561	103.3	0	[a]-2 003.8	99.1	102.6	[b]-59:59	337.97
11:20	25 567	103.3	0	[a]-1 980.5	102.4	102.6	[b]-59:59	337.95
[d]11:24.039	25 568	103.3	0	[a]-1 965.0	102.5	102.6	[b]-59:59	337.94

[a]SPLERROR = R_{TO-GO} (distance from current position to target - perigee greater than 300 000 ft).

[b]Time of free fall = POSMAX (-59:59) - perigee greater than 300 000 feet.

[c]Guidance cutoff.

[d]Insertion.

TABLE 5-IV.- SUMMARY OF THE NOMINAL MODE I (LEV) ABORT TRAJECTORIES

Abort time, min:sec	Abort altitude, ft	Abort apogee altitude, ft	Landing range, ft	Landing point: North geodetic latitude, deg:min:sec	Landing point: West longitude, deg:min:sec	Time to drogue deploy, min:sec, g.e.t.	Time to main deploy, min:sec, g.e.t.	Time to landing point, min:sec, g.e.t.
(a) Mode IA aborts								
00:00	424	4 738	5021	28:36:30	80:35:18	00:16	00:28	01:34
00:05	511	5 189	4944	28:36:30	80:35:19	00:21	00:33	01:58
00:10	800	5 871	4889	28:36:31	80:35:20	00:26	00:38	02:29
00:15	1314	6 902	4599	28:36:33	80:35:23	00:31	00:43	03:10
00:20	2079	8 186	4411	28:36:37	80:35:26	00:36	00:48	04:00
00:25	3130	9 696	4592	28:36:41	80:35:24	00:41	00:53	04:55
00:30	4493	11 473	5138	28:36:45	80:35:19	00:46	00:58	05:55
00:39	7847	15 655	6808	28:36:50	80:35:01	00:55	01:07	08:10
00:40	8297	16 211	7242	28:36:51	80:34:57	00:56	01:30	06:26
00:42	9250	17 283	7909	28:36:53	80:34:50	00:58	01:37	06:33

TABLE 5-IV.- SUMMARY OF THE NOMINAL MODE I (LEV) ABORT TRAJECTORIES - Concluded

Abort time, min:sec	Abort altitude, ft	Abort apogee altitude, ft	Landing range, n. mi.	Landing point		Time to drogue deploy, min:sec, g.e.t.	Time to main deploy, min:sec, g.e.t.	Time to landing point, min:sec, g.e.t.
				North geodetic latitude, deg:min:sec	West longitude, deg:min:sec			
(b) Mode IB aborts								
00:43	9 749	18 581	1.13	28:36:51	80:35:01	00:59	01:44	06:39
00:45	10 803	19 812	1.28	28:36:54	80:34:52	01:01	01:51	06:46
00:50	13 758	23 139	1.70	28:37:01	80:34:24	01:06	02:09	07:05
00:55	17 192	26 877	2.20	28:37:10	80:33:52	01:11	02:28	07:24
01:00	21 137	31 186	2.79	28:37:21	80:33:14	01:16	02:49	07:44
01:05	25 616	36 270	3.54	28:37:36	80:32:25	01:21	03:11	08:06
01:06	26 578	37 521	3.72	28:37:39	80:32:14	01:22	03:16	08:11
01:07	27 560	39 043	4.80	28:38:03	80:31:05	02:12	02:56	07:51
01:10	30 642	43 803	5.77	28:38:13	80:29:59	02:24	03:07	08:02
01:20	42 414	63 633	11.04	28:39:57	80:24:20	03:05	03:48	08:44
01:30	56 628	92 680	22.41	28:43:38	80:12:07	03:53	04:36	09:32
01:40	73 338	134 903	49.85	28:51:56	79:42:21	04:45	05:29	10:24
01:45	82 601	162 743	71.25	28:58:09	79:19:02	05:13	05:56	10:52
01:50	92 460	185 866	92.91	29:04:36	78:55:27	05:33	06:17	11:12
01:53	98 659	202 518	105.17	29:08:24	78:42:10	05:47	06:30	11:26
(c) Mode IC aborts								
01:54	100 773	207 976	109.95	29:10:13	78:37:06	05:52	06:35	11:30
02:00	113 955	240 130	143.27	29:19:23	78:00:29	06:17	07:00	11:56
02:10	137 901	296 129	207.18	29:36:57	76:50:08	06:57	07:40	12:36
02:20	164 254	350 884	278.53	29:56:11	75:31:14	07:35	08:18	13:14
02:30	192 258	402 445	356.21	30:16:33	74:04:53	08:12	08:55	13:51
02:40	221 994	462 499	451.77	30:40:01	72:17:46	08:47	09:31	14:26
02:50	251 517	467 106	465.74	30:43:37	72:02:07	08:59	09:43	14:38
03:00	279 153	475 362	486.02	30:49:05	71:39:27	09:09	09:52	14:48
03:10	305 072	483 431	506.38	30:55:04	71:16:48	09:16	09:59	14:55
03:15.9	319 596	488 151	519.0	30:58:18	71:02:47	09:21	10:04	15:04

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR APOLLO 10 (MISSION F)

(a) Entry parameters

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Maximum entry load factor, g	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Range at landing, n. mi.
[a] 3:16.6	9 570.6	8.95	9 295.24	-5.88	30.51	72.63	431.72
3:20	9 637.1	9.09	9 389.88	-6.40	30.54	72.50	438.79
3:30	9 839.9	9.40	9 670.10	-7.73	30.62	72.09	460.81
3:40	10 047.2	9.75	9 948.51	-8.90	30.71	71.64	484.31
3:50	10 265.2	10.08	10 229.69	-9.69	30.79	71.19	508.45
4:00	10 494.3	10.39	10 515.33	-10.28	30.88	70.72	533.26
4:10	10 739.8	10.66	10 808.83	-10.65	30.96	70.23	559.24
4:20	10 985.8	10.93	11 102.17	-11.01	31.06	69.73	585.22
4:30	11 248.5	11.24	11 404.38	-11.22	31.14	69.21	612.51
4:40	11 522.5	11.53	11 713.08	-11.34	31.23	68.67	640.75
4:50	11 808.1	11.78	12 028.83	-11.39	31.32	68.11	670.04
5:00	12 105.5	12.02	12 352.25	-11.39	31.41	67.53	700.45
5:10	12 414.9	12.24	12 683.86	-11.33	31.50	66.92	732.09
5:20	12 736.6	12.44	13 024.38	-11.23	31.59	66.29	765.07
5:30	13 071.3	12.61	13 374.58	-11.08	31.68	65.62	799.55
5:40	13 419.3	12.69	13 735.26	-10.90	31.77	64.93	835.68
5:50	13 781.2	12.88	14 107.13	-10.69	31.87	64.19	873.64

[a] Launch escape tower jettison.

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR APOLLO 10 (MISSION F) - Continued

(a) Entry parameters - Continued

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Maximum entry load factor, g	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Range at landing, n. mi.
6:00	14 157.4	12.94	14 490.96	-10.46	31.96	63.42	913.68
6:10	14 548.6	13.02	14 887.68	-10.19	32.05	62.59	956.03
6:20	14 963.4	13.03	15 306.71	-9.89	32.13	61.67	1 002.46
6:30	15 379.2	13.02	15 723.89	-9.60	32.22	60.78	1 049.11
6:40	15 820.3	12.98	16 165.46	-9.28	32.31	59.77	1 100.70
6:50	16 279.8	12.86	16 624.18	-8.94	32.39	58.68	1 156.47
7:00	16 758.9	12.73	17 101.36	-8.58	32.47	57.48	1 217.23
7:10	17 258.7	12.55	17 598.44	-8.20	32.54	56.17	1 283.95
7:20	17 780.7	12.27	18 116.99	-7.82	32.60	54.71	1 357.94
7:30	18 326.5	11.96	18 658.77	-7.41	32.65	53.07	1 441.02
7:40	18 897.9	11.56	19 225.68	-6.99	32.67	51.21	1 535.67
7:50	19 496.1	11.10	19 819.25	-6.56	32.66	49.04	1 645.36
8:00	20 123.9	10.51	20 442.44	-6.11	32.59	46.47	1 775.58
8:10	20 667.7	9.92	20 979.82	-5.71	32.47	43.94	1 904.17
8:20	21 176.7	9.30	21 486.00	-5.34	32.27	41.18	2 044.85
8:30	21 707.7	8.58	22 014.09	-4.95	31.94	37.91	2 212.78
8:40	22 261.3	7.75	22 564.79	-4.53	31.41	33.86	2 422.38

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR APOLLO 10 (MISSION F) - Continued

(a) Entry parameters - Concluded

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Maximum entry load factor, g	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Range at landing, n. mi.
8:50	22 805.6	6.85	23 095.87	-4.10	30.58	29.05	2 675.19
[a] 8:50.1	22 805.6	6.84	23 095.98	-4.10	30.58	29.05	2 675.27
9:00	22 897.2	6.66	23 193.12	-4.02	30.38	28.06	2 727.84
9:10	23 077.4	6.32	23 374.62	-3.86	29.94	26.09	2 834.47
9:20	23 257.9	5.97	23 555.15	-3.70	29.50	24.12	2 940.02
9:30	23 444.6	5.61	23 741.16	-3.53	28.89	21.80	3 067.17
9:40	23 631.5	5.25	23 928.18	-3.35	28.28	19.49	3 194.28
[b] 9:41	23 650.5	5.20	23 947.11	-3.33	28.21	19.24	3 208.52
9:50	23 823.1	4.85	24 118.75	-3.15	27.47	16.80	3 345.39
10:00	24 017.7	4.43	24 312.07	-2.95	26.47	13.74	3 519.66
10:10	24 218.1	3.96	24 509.36	-2.71	24.98	9.86	3 748.81
10:20	24 417.2	3.50	24 707.70	-2.48	23.50	5.99	3 977.57
10:30	24 621.5	3.00	24 909.59	-2.22	21.17	0.78	4 298.57
10:40	24 829.6	2.66	25 114.82	-1.92	17.69	-6.07	4 738.30
10:50	25 040.9	2.55	25 322.55	-1.57	12.04	-15.78	5 394.18
11:00	25 255.6	2.33	25 531.97	-1.13	0.17	-33.58	6 667.67

[a] SII/S-IVB separation.

[b] End mode II.

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Event times

Time of abort, min:sec, g.e.t.	Predicted time of free fall from abort to 300 000 ft min:sec	Time at 400 000 ft min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Time at landing, min:sec, g.e.t.
[a] 3:16.6	2:59.72	4:08.64	--	--	9:36.27	14:44.27
3:20	2:59.60	5:25.01	--	--	9:39.59	14:47.59
3:30	2:59.87	5:40.39	--	--	9:50.14	14:58.14
3:40	3:01.21	5:55.66	--	--	10:01.60	15:09.60
3:50	3:02.26	6:09.56	--	--	10:13.13	15:21.13
4:00	3:03.05	6:22.53	--	--	10:24.75	15:32.75
4:10	3:03.41	6:34.63	--	--	10:36.56	15:44.56
4:20	3:04.15	6:46.73	--	--	10:48.42	15:56.42
4:30	3:04.54	6:58.25	--	--	11:00.53	16:08.53
4:40	3:04.86	7:09.50	--	--	11:12.84	16:20.84
4:50	3:05.15	7:20.55	--	--	11:25.38	16:33.38
5:00	3:05.42	7:31.46	--	--	11:38.19	16:46.19
5:10	3:05.71	7:42.26	--	--	11:51.29	16:59.29
5:20	3:06.05	7:53.00	--	--	12:04.71	17:12.71
5:30	3:06.46	8:03.72	--	--	12:18.50	17:26.50

[a] Launch escape tower jettison.

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Event times - Continued

Time of abort, min:sec, g.e.t.	Predicted time of free fall from abort to 300 000 ft min:sec	Time at 400 000 ft min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Time at landing, min:sec, g.e.t.
5:40	3:06.97	8:14.46	--	--	12:32.70	17:40.70
5:50	3:07.62	8:25.26	--	--	12:47.37	17:55.37
6:00	3:08.46	8:36.17	--	--	13:02.57	18:10.57
6:10	3:09.53	8:47.24	--	--	13:18.37	18:26.37
6:20	3:11.06	8:58.89	--	--	13:34.41	18:42.41
6:30	3:12.58	9:10.05	--	--	13:52.13	19:00.13
6:40	3:14.72	9:21.95	--	--	14:10.32	19:18.32
6:50	3:17.39	9:34.29	--	--	14:29.61	19:37.61
7:00	3:20.72	9:47.21	11:04	11:09	14:50.21	19:58.21
7:10	3:24.90	10:00.85	11:15	11:27	15:12.38	20:20.38
7:20	3:30.15	10:15.45	11:29	11:45	15:36.44	20:44.44
7:30	3:36.78	10:31.28	11:44	12:05	16:02.88	21:10.88
7:40	3:45.23	10:48.75	12:01	12:26	16:32.34	21:40.34
7:50	3:56.15	11:08.45	12:21	12:50	17:05.70	22:13.70
8:00	4:10.53	11:31.31	12:44	13:18	17:44.40	22:52.40

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Event times - Continued

Time of abort, min:sec, g.e.t.	Predicted time of free fall from abort to 300 000 ft min:sec	Time at 400 000 ft min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t	Time at drogue chute deployment, min:sec, g.e.t.	Time at landing, min:sec, g.e.t.
8:10	4:24.51	11:53.60	13:07	13:46	18:21.82	23:29.82
8:20	4:40.74	12:18.08	13:33	14:16	19:02.11	24:10.11
8:30	5:01.56	12:46.62	14:03	14:52	19:49.26	24:57.26
8:40	5:29.93	13:22.04	14:42	15:37	20:47.07	25:55.07
8:50	6:07.79	14:05.17	15:29	16:32	21:55.64	27:03.64
[a]8:50.1	6:07.68	14:05.18	15:29	16:32	21:55.66	27:03.66
9:00	6:07.25	14:14.09	15:39	16:43	22:09.80	27:17.80
9:10	6:16.86	14:31.54	15:59	17:07	22:38.61	27:46.61
9:20	6:26.02	14:48.99	16:18	17:30	23:06.49	28:14.49
9:30	6:38.70	15:09.01	16:41	17:57	23:39.51	28:47.51
9:40	6:51.39	15:29.04	17:04	18:24	24:13.68	29:21.68
[b]9:41	6:52.92	15:31.25	17:06	18:28	24:17.43	29:25.43
9:50	7:08.16	15:52.34	17:31	18:57	24:53.34	30:01.34
10:00	7:29.02	16:18.95	18:03	19:35	25:38.87	30:46.87
10:10	7:59.86	16:53.68	18:45	20:27	26:37.96	31:45.96

[a]SII/S-IVB separation.

[b]End mode II.

TABLE 5-V.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE II ABORTS FOR THE APOLLO 10 (MISSION F) - Concluded

(b) Event times - Concluded

Time of abort, min:sec, g.e.t.	Predicted time of free fall from abort to 300 000 ft min:sec	Time at 400 000 ft min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t	Time at drogue chute deployment, min:sec, g.e.t.	Time at landing, min:sec, g.e.t.
10:20	8:30.70	17:28.23	19:27	21:18	27:37.74	32:45.74
10:30	9:18.89	18:16.71	20:27	22:29	29:00.62	34:08.62
10:40	10:29.96	19:23.24	21:50	24:09	30:53.80	36:01.80
10:50	12:22.77	21:00.96	23:58	26:42	33:42.02	38:50.02
11:00	16:19.83	24:10.71	28:17	31:52	39:08.04	44:16.04

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F)

(a) Burn parameters

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Time at SPS ignition, min:sec, g.e.t.	SPS burn time, min:sec	SPS ΔV, fps	Predicted time of free fall from SPS cutoff to 300 000 ft, fps	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg
9:41	23 650.5	--	0	0	[a]6:52	23 947.1	-3.32
9:42	23 669.2	--	0	0	[a]6:54	23 966.2	-3.30
9:44	23 707.7	--	0	0	[a]6:57	24 004.1	-3.27
9:46	23 746.0	--	0	0	[a]7:01	24 042.2	-3.23
9:48	23 784.5	--	0	0	[a]7:04	24 080.4	-3.19
9:50	23 823.1	--	0	0	[a]7:08	24 118.8	-3.15
9:52	23 861.7	--	0	0	[a]7:11	24 157.0	-3.11
9:54	23 900.0	--	0	0	[a]7:15	24 195.7	-3.07
9:56	23 939.4	--	0	0	[a]7:20	24 234.1	-3.03
9:58	23 978.4	--	0	0	[a]7:24	24 273.0	-2.98
10:00	24 017.7	--	0	0	[a]7:29	24 312.1	-2.94
10:02	24 057.1	--	0	0	[a]7:33	24 351.2	-2.89
10:04	24 096.7	--	0	0	[a]7:38	24 390.4	-2.85
10:04	24 096.7	12:09	00:00.4	4.69	5:33	24 387.5	-2.86
10:06	24 136.4	12:11	00:03.0	40.46	5:27	24 404.1	-2.86

[a]Time of free fall at abort.

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(a) Burn parameters - Continued

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Time at SPS ignition, min:sec, g.e.t.	SPS burn time, min:sec	SPS ΔV, fps	Predicted time of free fall from SPS cutoff to 300 000 ft, fps	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg
10:08	24 176.1	12:13	00:07	77.0	5:20	24 420.4	-2.86
10:10	24 215.9	12:15	00:11	114.0	5:14	24 436.7	-2.86
10:12	24 255.9	12:17	00:14	151.6	5:08	24 452.8	-2.87
10:14	24 296.0	12:19	00:18	189.9	5:01	24 468.8	-2.88
10:16	24 336.3	12:21	00:21	228.6	4:55	24 484.6	-2.88
10:18	24 376.6	12:23	00:26	268.9	4:49	24 500.2	-2.90
10:20	24 417.1	12:25	00:29	308.3	4:43	24 516.3	-2.91
10:22	24 452.2	12:27	00:33	349.6	4:36	24 536.5	-2.92
10:24	24 498.5	12:29	00:37	390.2	4:30	24 547.5	-2.93
10:26	24 539.3	12:31	00:41	432.5	4:24	24 562.8	-2.95
10:28	24 580.3	12:33	00:45	475.5	4:17	24 577.9	-2.97
10:30	24 621.5	12:35	00:49	519.5	4:11	24 592.8	-2.99
10:32	24 662.8	12:37	00:53	564.3	4:05	24 607.5	-3.02
10:34	24 704.2	12:39	00:57	610.0	3:58	24 622.2	-3.04
10:36	24 745.9	12:41	01:01	656.5	3:52	24 636.7	-3.07
10:38	24 787.7	12:43	01:05	704.9	3:46	24 650.5	-3.10

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(a) Burn parameters - Continued

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Time at SPS ignition, min:sec, g.e.t.	SPS burn time, min:sec	SPS ΔV, fps	Predicted time of free fall from SPS cutoff to 300 000 ft, fps	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg
10:40	24 829.6	12:45	1:12	753.7	3:39	24 664.4	-3.13
10:42	24 871.5	12:47	1:13	803.8	3:33	24 678.0	-3.17
10:44	24 913.7	12:49	1:19	853.8	3:27	24 692.0	-3.20
10:46	24 955.9	12:51	1:23	902.8	3:21	24 706.9	-3.24
10:48	24 998.3	12:53	1:28	955.5	3:14	24 720.2	-3.28
10:50	25 040.9	12:55	1:32	1 008.8	3:08	24 733.6	-3.33
10:52	25 083.6	12:57	1:37	1 064.0	3:02	24 746.4	-3.37
10:54	25 126.4	12:59	1:42	1 122.0	2:56	24 758.2	-3.42
10:56	25 169.3	13:01	1:47	1 181.3	2:49	24 769.7	-3.48
10:58	25 212.4	13:03	1:52	1 243.7	2:43	24 780.2	-3.54
11:00	25 255.5	13:05	1:57	1 306.7	2:36	24 790.9	-3.60
11:02	25 298.9	13:07	2:03	1 373.0	2:30	24 800.6	-3.67
11:04	25 342.3	13:09	2:09	1 441.8	2:23	24 809.5	-3.74
11:06	25 385.9	13:11	2:15	1 514.0	2:16	24 917.5	-3.82
11:08	25 429.6	13:13	2:21	1 589.0	2:10	24 824.8	-3.91

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(a) Burn parameters - Concluded

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Time at SPS ignition, min:sec, g.e.t.	SPS burn time, min:sec	SPS ΔV, fps	Predicted time of free fall from SPS cutoff to 300 000 ft, fps	Inertial velocity at 400 000 ft fps	Inertial flight-path angle 400 000 ft, deg
11:10	25 473.4	13:15	2:28	1 667.2	2:03	24 831.4	-4.00
11:12	25 517.3	13:17	2:34	1 748.4	1:56	24 837.3	-4.09
11:14	25 561.4	13:19	2:41	1 835.9	1:49	24 841.3	-4.20
[a]11:14.039	25 562.3	13:19.039	2:42	1 836.0	1:48	24 839.7	-4.21
11:16	25 567.4	13:21	2:43	1 857.7	1:46	24 833.5	-4.24
11:18	25 567.4	13:23	2:45	1 877.1	1:44	24 825.1	-4.28
11:20	25 567.4	13:25	2:46	1 897.0	1:41	24 816.1	-4.33
11:22	25 567.4	13:27	2:48	1 917.3	1:40	24 807.8	-4.37
11:24	25 567.5	13:29	2:50	1 938.1	1:36	24 799.0	-4.41
[b]11:24.039	25 567.5	13:29.039	2:50	1 938.9	1:35	24 798.2	-4.42

[a]S-IVB cutoff.

[b]Insertion.

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Entry parameters

Time of abort, min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Maximum load factor
9:41	17:06	18:22	22:13	28.36	-23.35	7.32
9:42	17:18	18:24	22:16	28.29	-23.11	7.26
9:44	17:23	18:30	22:21	28.16	-22.62	7.17
9:46	17:28	18:37	22:30	28.02	-22.12	7.06
9:48	17:34	18:43	22:37	27.88	-21.61	6.96
9:50	17:40	18:50	22:45	27.73	-21.09	6.85
9:52	17:37	18:58	22:54	27.57	-20.55	6.75
9:54	17:43	19:05	23:02	27.41	-20.00	6.64
9:56	17:49	19:13	23:10	27.23	-19.43	6.53
9:58	17:56	19:20	23:19	27.05	-18.84	6.42
10:00	18:03	19:28	23:28	26.86	-18.24	6.31
10:02	18:10	19:37	23:37	26.65	-17.61	6.19
10:04	18:17	19:45	23:46	26.45	-16.98	6.07
10:04	18:23	19:43	23:44	26.48	-17.06	6.10
10:06	18:22	19:43	23:45	26.48	-17.06	6.06
10:08	18:21	19:42	23:44	26.48	-17.06	6.04

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Entry parameters - Continued

Time of abort, min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Maximum load factor
10:10	18:20	19:41	23:44	26.48	-17.06	6.03
10:12	18:12	19:42	23.44	26.48	-17.06	6.01
10:14	18:11	19:41	23:43	26.48	-17.07	6.00
10:16	18:11	19:39	24:43	26.48	-17.07	5.99
10:18	18:10	19:40	23:42	26.48	-17.08	5.99
10:20	18:09	19:39	23:42	26.48	-17.08	5.98
10:22	18:08	19:39	23:42	26.48	-17.08	5.99
10:24	18:08	19:38	23:41	26.47	-17.06	5.99
10:26	18:07	19:38	23:41	26.47	-17.06	6.00
10:28	18:07	19:37	23:40	26.47	-17.06	6.01
10:30	18:06	19:37	23:40	26.48	-17.06	6.03
10:32	18:06	19:36	23:39	26.48	-17.06	6.04
10:34	18:06	19:36	23:39	26.48	-17.06	6.06
10:36	18:06	19:35	23:38	26.48	-17.05	6.09
10:38	18:05	19:35	23:38	26.48	-17.06	6.13
10:40	18:05	19.34	23:37	26.48	-17.06	6.16

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Entry parameters - Continued

Time of abort, min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Maximum load factor
10:42	18:05	19:34	23:37	26.48	-17.06	6.20
10:44	18:05	19:34	23:36	26.48	-17.05	6.25
10:46	18:05	19:33	23:36	26.48	-17.04	6.29
10:48	18:05	19:33	23:35	26.49	-17.06	6.35
10:50	18:05	19:32	23:34	26.49	-17.05	6.41
10:52	18:05	19:32	23:34	26.49	-17.05	6.48
10:54	18:05	19:32	23:33	26.50	-17.05	6.56
10:56	18:05	19:31	23:32	26.50	-17.05	6.65
10:58	18:05	19.31	23:31	26.51	-17.06	6.75
11:00	18:06	19:31	23.30	26.51	-17.05	6.85
11:02	18:07	19:31	23:30	26.52	-17.05	6.97
11:04	18:07	19:30	23.29	26:52	-17.04	7.10
11:06	18:08	19:30	23:28	26.53	-17.04	7.24
11:08	18:09	19:30	23:27	26.54	-17.04	7.40
11:10	18:10	19:30	23:26	26.54	-17.04	7.58
11:12	18:11	19:30	23:25	26.55	-17.02	7.76

TABLE 5-VI.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE III ABORTS FOR THE APOLLO 10 (MISSION F) - Concluded

(b) Entry parameters - Concluded

Time of abort, min:sec, g.e.t.	Time at S-band blackout entry, min:sec, g.e.t.	Time at S-band blackout exit, min:sec, g.e.t.	Time at drogue chute deployment, min:sec, g.e.t.	Geodetic latitude at landing, deg N	Longitude at landing, deg W	Maximum load factor
11:14	18:13	19:30	23:24	26.56	-17.03	7.98
[a]11:14.039	18:13	19:30	23:24	25:56	-17.03	8.02
11:16	18:13	19:30	23:23	26.57	-17.03	8.07
11:18	18:15	19:30	23:23	26.57	-17.01	8.16
11:20	18:15	19:31	23.23	26.57	-17.02	8.26
11:22	18:16	19:31	23:23	26.58	-17.02	8.36
11:24	18:17	19:31	23:22	26.58	-17.02	8.46
[b]11:24.039	15:18	19:32	23:22	26.58	-17.02	8.50

[a]S-IVB guidance cutoff.

[b]Insertion.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F)

(a) Two-impulse parameters

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Predicted perigee altitude after first impulse, n. mi.	Time of free fall to 300 000 ft at SPS cutoff after the first impulse, min:sec	SPS ΔV-[a] first impulse, fps	Coast time to apogee (time from abort to ignition of second burn), min:sec	SPS ΔV second impulse, fps
9:34	23 517.9	64.0	--	2 457.9	34:31	8.5
9:36	23 556.1	65.1	--	2 413.9	34:38	6.3
9:38	23 593.6	66.3	--	2 369.8	34:45	4.0
9:40	23 632.0	67.3	--	2 325.8	34:52	1.4
9:42	23 669.5	68.2	--	2 281.7	34:59	0.0
9:44	23 707.7	69.5	--	2 237.6	35:07	0.0
9:46	23 746.1	70.6	--	2 193.4	35:15	0.0
9:48	23 784.5	71.1	--	2 149.1	35:23	0.0
9:50	23 823.0	72.2	--	2 103.2	35:31	0.0
9:52	23 851.1	73.6	--	2 060.3	35:38	0.0
9:54	23 900.5	74.6	--	2 016.0	35:47	0.0
9:56	23 939.4	75.1	--	1 971.5	35:55	0.0

[a]First SPS burn ignition begins 90 seconds after abort.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(a) Two-impulse parameters - Continued

Time of abort, min:sec g.e.t.	Inertial velocity at abort, fps	Predicted perigee altitude after first impulse, n. mi.	Time of free fall to 300 000 ft at SPS cutoff after the first impulse, min:sec	SPS ΔV-[b] first impulse, fps	Coast time to apogee (time from abort to ignition of second burn), min:sec	SPS ΔV second impulse, fps
8:37	22 092.8	29.5	62:57	4 237.4	34:56	75.8
8:38	22 149.7	30.1	62:58	4 155.1	34:42	73.7
8:40	22 261.3	31.6	63:00	4 004.1	34:14	70.7
8:42	22 375.1	32.8	63:01	3 852.9	33:46	68.0
8:44	22 489.4	34.0	63:01	3 701.6	33:17	65.6
8:46	22 605.0	35.0	62:58	3 550.1	32:48	63.5
8:48	22 721.3	35.8	62:54	3 398.5	32:20	61.7
8:50	22 806.1	36.9	63:08	3 304.9	32:08	59.6
[a] 8:50.1	22 805.6	36.9	63:11	3 305.0	32:10	59.4
8:52	22 806.2	38.4	63:46	3 307.8	32:25	56.8
8:54	22 806.3	39.9	64:41	3 310.4	32:43	54.1
8:56	22 827.1	41.3	65.23	3 282.9	32:52	51.5
8:58	22 862.4	42.5	66:00	3 239.9	32:56	49.1
9:00	22 897.5	43.7	66:40	3 196.9	32:59	46.9
9:02	22 933.1	45.0	67:22	3 153.8	33:03	44.6

[a]S-II/S-IVB staging.

[b]First SPS burn ignition begins 90 seconds after abort.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(a) Two-impulse parameters - Concluded

Time of abort, min:sec, g.e.t.	Inertial velocity at abort, fps	Predicted perigee altitude after first impulse n. mi.	Time of free fall to 300 000 ft at SPS cutoff after the first impulse, min:sec	SPS ΔV_-[b] first impulse, fps	Coast time to apogee (time from abort to ignition of second burn), min:sec	SPS ΔV second impulse, fps
9:04	22 968.2	46.3	68:12	3 110.7	33:07	42.2
9:06	23 004.0	47.4	69:13	3 067.6	33:12	39.9
9:08	23 039.8	50.2	a	3 024.2	33:16	37.7
9:10	23 075.8	50.8	a	2 980.9	33:21	35.5
9:12	23 112.0	51.1	a	2 937.6	33:26	33.1
9:14	23 148.3	53.0	a	2 894.1	33:31	31.0
9:16	23 185.0	53.5	a	2 850.7	33:36	28.6
9:18	23 221.2	55.2	a	2 807.2	33:41	26.4
9:20	23 258.0	55.9	a	2 763.8	33:47	24.2
9:22	23 294.6	57.4	a	2 720.2	33:53	21.5
9:24	23 332.0	58.2	a	2 676.7	33:59	19.8
9:26	23 368.5	59.6	a	2 633.1	34:05	17.5
9:28	23 406.0	60.6	a	2 589.4	34:11	15.5
9:30	23 442.9	61.3	a	2 545.7	34:17	12.5
9:32	23 480.3	62.9	a	2 501.9	34:24	10.1

[a]Perigee altitude is greater than 300 000 feet.

[b]First SPS burn ignition begins 90 seconds after abort.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Single impulse parameters without ΔV pad

At abort initiation		At SPS ignition	SPS burn duration, min:sec	After nominal SPS burn[a]		Predicted apogee altitude, n. mi.
Time, min:sec, g.e.t.	Inertial velocity, fps	Time, min:sec, g.e.t.		SPS sensed velocity change, fps	True anomaly, deg	
9:56	23 939.5	11:26	2:52.18	1 971.5	54.82	198.62
9:58	23 978.5	11:28	2:45.29	1 883.8	58.21	181.14
10:00	24 017.7	11:30	2:40.31	1 818.7	61.10	172.23
10:02	24 057.2	11:32	2:35.43	1 757.2	63.89	164.59
10:04	24 096.7	11:34	2:30.54	1 697.4	66.67	158.01
10:06	2 4136.4	11:36	2:26.62	1 638.8	69.50	152.62
10:08	24 176.1	11:38	2:21.71	1 580.6	72.41	147.11
10:10	24 216.0	11:40	2:17.32	1 525.5	75.32	142.01
10:12	24 256.0	11:42	2:12.91	1 470.3	78.23	138.38
10:14	24 296.1	11:44	2:07.09	1 416.5	81.21	134.71
10:16	24 336.3	11:46	2:02.51	1 363.7	84.31	131.52
10:18	24 376.7	11:48	1:58.78	1 309.8	87.29	128.32
10:20	24 417.4	11:50	1:54.64	1 256.9	90.36	125.61
10:22	24 457.8	11:52	1:49.76	1 206.0	93.64	123.14
10:24	24 498.5	11:54	1:45.21	1 154.1	96.89	120.76

[a] Resulting predicted perigee = 75 nautical miles.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Single impulse parameters without ΔV pad - Continued

At abort initiation		At SPS ignition	SPS burn duration, min:sec	After nominal SPS burn[a]		Predicted apogee altitude, n. mi.
Time, min:sec, g.e.t.	Inertial velocity, fps	Time, min:sec, g.e.t.		SPS sensed velocity change, fps	True anomaly, deg	
10:26	24 539.4	11:56	1:41.01	1 104.4	100.13	118.71
10:28	24 580.4	11:58	1:36.33	1 052.7	103.54	116.83
10:30	24 621.5	12:00	1:32.00	1 002.9	106.87	115.07
10:32	24 662.8	12:02	1:27.74	953.1	110.39	113.50
10:34	24 704.3	12:04	1:23.47	903.5	113.87	112.01
10:36	24 745.9	12:06	1:19.16	854.5	117.48	110.73
10:38	24 787.7	12:08	1:14.75	804.9	121.12	109.47
10:40	24 829.6	12:10	1:10.40	756.6	124.87	108.42
10:42	24 871.6	12:12	1:06.11	706.9	128.48	107.38
10:44	24 913.7	12:14	1:01.62	658.2	132.72	106.50
10:46	24 956.0	12:16	0:57.17	609.1	137.49	105.38
10:48	24 998.4	12:18	0:52.63	561.3	142.31	104.62
10:50	25 040.9	12:20	0:48.34	511.7	146.92	103.88
10:52	25 083.6	12:22	0:44.07	464.5	151.23	103.31
10:54	25 126.4	12:24	0:39.76	415.5	155.57	102.90

[a]Resulting predicted perigee = 75 nautical miles.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(b) Single impulse parameters without ΔV pad - Concluded

At abort initiation		At SPS ignition Time, min:sec, g.e.t.	After nominal SPS burn[a]			
Time, min:sec, g.e.t.	Inertial velocity fps		SPS burn duration, min:sec	SPS sensed velocity change, fps	True anomaly, deg	Predicted apogee altitude, n. mi.
10:56	25,169.4	12:26	0:35.95	369.3	159.72	102.64
10:58	25,212.4	12:28	0:30.42	320.7	163.81	102.41
11:00	25,255.6	12:30	0:26.06	274.0	167.85	102.23
11:02	25,298.9	12:32	0:22.04	227.4	171.62	102.20
11:04	25,342.4	12:34	0:17.72	178.9	175.24	102.14
11:06	25,385.9	12:36	0:13.25	131.7	178.73	102.20
11:08	25,429.6	12:38	0:08.73	85.3	181.92	102.31
11:10	25,473.5	12:40	0:04.22	38.1	185.33	102.50
11:12	25,517.4	12:42	0:00.00	0.0	187.99	102.62

[a] Resulting predicted perigee = 75 nautical miles.

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(c) Single impulse parameters with a 100 fps pad

At abort initiation		After over burn of 100 fps				
Time, min:sec, g.e.t.	Inertial velocity, fps	Total SPS burn duration, min:sec	Total SPS sensed velocity change, fps	True anomaly, deg	Predicted apogee altitude, n. mi.	Predicted perigee altitude, n. mi.
9:56	23 939.5	3:01.03	2 071.5	44.51	246.66	77.38
9:58	23 978.5	2:53.31	1 983.8	46.22	226.52	77.90
10:00	24 017.7	2:48.47	1 918.7	47.29	217.13	78.51
10:02	24 057.2	2:43.39	1 857.2	48.17	208.91	79.02
10:04	24 096.7	2:38.41	1 797.4	49.06	201.82	79.61
10:06	24 136.4	2:33.78	1 738.8	49.90	195.87	80.15
10:08	24 176.1	2:29.37	1 680.6	50.73	189.89	80.68
10:10	24 216.0	2:25.03	1 625.5	51.35	184.95	81.28
10:12	24 256.0	2:19.78	1 570.3	52.03	180.09	81.92
10:14	24 296.1	2:14.97	1 516.5	52.51	175.92	82.47
10:16	24 336.3	2:10.85	1 463.7	52.96	172.12	83.09
10:18	24 376.7	2:07.06	1 409.8	53.40	168.38	83.67
10:20	24 417.4	2:02.33	1 356.9	53.82	165.14	84.36
10:22	24 457.8	1:58.21	1 306.0	54.09	161.93	85.01
10:24	24 498.5	1:53.48	1 254.1	54.26	159.07	85.72

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Continued

(c) Single impulse parameters with a 100 fps pad - Continued

At abort initiation		After over burn of 100 fps				
Time, min:sec, g.e.t.	Inertial velocity, fps	Total SPS burn duration, min:sec	Total SPS sensed velocity change, fps	True anomaly, deg	Predicted apogee altitude, n. mi.	Predicted perigee altitude, n. mi.
10:26	24 539.4	1:48.76	1 204.4	54.41	156.37	86.44
10:28	24 580.4	1:44:94	1 152.7	54.52	153.91	87.09
10:30	24 621.5	1:41.03	1 102.9	54.50	151.60	87.92
10:32	24 662.8	1:35.90	1 053.1	54.44	149.36	88.47
10:34	24 704.3	1:31.76	1 003.5	54.34	147.30	89.20
10:36	24 745.9	1:27.65	954.5	53.91	145.29	90.02
10:38	24 787.7	1:21.59	904.9	53.62	143.48	90.71
10:40	24 829.6	1:19.12	856.6	53.14	141.66	91.39
10:42	24 871.6	1:15.34	806.9	52.49	140.14	92.21
10:44	24 913.7	1:10.44	758.2	51:58	138.47	93.00
10:46	24 956.0	1:05.65	709.1	50.32	136.55	93.88
10:48	24 998.4	1:01.93	661.3	48.79	135.00	94.84
10:50	25 040.9	0:57.45	611.7	47.06	133.52	95.62
10:52	25 083.6	0:53.32	564.5	45.18	132.27	96.42
10:54	25 126.4	0:48.50	515.5	43.14	131.18	97.21
10:56	25 169.4	0:43.90	469.3	40.70	130.20	97.91

TABLE 5-VII.- TRAJECTORY CHARACTERISTICS FOLLOWING NOMINAL MODE IV ABORTS FOR THE APOLLO 10 (MISSION F) - Concluded

(c) Single impulse parameters with a 100 fps pad - Concluded

At abort initiation		After over burn of 100 fps				
Time, min:sec, g.e.t.	Inertial velocity, fps	Total SPS burn duration, min:sec	Total SPS sensed velocity change, fps	True anomaly, deg	Predicted apogee altitude, n. mi.	Predicted perigee altitude, n. mi.
10:58	25 212.4	0:40.11	420.7	38.14	129.38	98.57
11:00	25 255.6	0:34.87	374.0	35.19	128.72	99.31
11:02	25 298.9	0:31.41	327.4	32.06	128.19	99.89
11:04	25 342.4	0:27.00	278.9	28.83	127.70	100.54
11:06	25 385.9	0:22.09	231.7	25.38	127.40	101.02
11:08	25 429.6	0:17.66	185.3	21.79	127.11	101.39
11:10	25 473.5	0:13.22	138.1	18.20	127.02	101.83
11:12	25 517.4	0:08.80	100.0	13.51	142.4	102.04

TABLE 7-1. - TYPICAL TRANSLUNAR COAST ABORT SOLUTIONS.

Abort, g.e.t., hr:min:sec	Approximate time from TLI cutoff, hr	IMU gimbal angles referenced to PTC REFSMMAT			ΔV_t, fps	Δt_B, min:sec	TAR, hr:min:sec	V_{EI}, fps	γ_{EI}, deg	ϕ_L, deg	λ_L, deg	External ΔV targets		
		OGA, deg	IGA, deg	MGA, deg								ΔV_X, fps	ΔV_Y, fps	ΔV_Z, fps
04:02:04.3	1.5*	180.57*	244.89*	0.68*	7415.3	08:13.6	08:08:33	33844.1	-6.18	-28.70	335.00	-508.21	0.0	7397.9
06:30:00.0	4.C*	180.17*	230.55*	0.63*	6668.3	07:39.0	15:21:50	34826.7	-6.33	-27.80	195.00	-170.21	0.0	6660.8
13:30:00.0	11.0*	271.93*	349.29*	-37.00*	5309.5	06:29.2	32:40:27	35501.3	-6.42	-26.21	195.00	-46.74	0.0	5309.3
27:30:00.0	25.0	271.91	349.57	-43.76	5897.9	07:00.6	42:30:01	35831.2	-6.46	-26.68	195.00	-1.38	0.0	5897.9
37:30:00.0	35.0	95.66	355.07	-38.35	5120.0	06:18.7	56:37:53	35925.4	-6.48	-25.74	195.00	27.80	0.0	5119.9
46:30:00.0	44.0	96.05	355.24	-39.86	6779.5	07:04.3	47:15:54	36101.3	-6.50	-27.41	195.00	47.21	0.0	6779.4
55:30:00.0	53.0	96.57	355.27	-40.42	5552.5	06:42.3	62:29:51	36069.0	-6.49	-25.98	195.00	100.64	0.0	5551.6

*Gimbal angles referenced to launch pad REFSMMAT.

TABLE 8-1.- RECOMMENDED ACTION FOR PROBLEMS DURING LOI

TYPE		
1	GUIDANCE AND CONTROL (IMU DRIFTS, ETC)	MANUAL TAKEOVER AT 10°/SEC OR 10° ATTITUDE DEVIATION AND COMPLETE LOI AT IGNITION ATTITUDE
2	NON-SPS (ECS ETC)	COMPLETE LOI
3	SPS (PRESS ETC)	CREW CHART ABORT SPS AT 15 MINUTES OR DPS AT PERILUNE (RTCC)
4	INADVERTANT SHUTDOWNS (CMC ETC)	DPS ABORT AFTER 2 HOURS (RTCC)

TABLE 8-II.- FIFTEEN-MINUTE CREW CHART RESULTS FOR LOI

(a) Data for burn and for abort

LOI burn time min:sec	LOI ΔV, DVM, fps	Abort ΔV fps	Landing conditions	
			Latitude, deg:min S	Longitude, deg:min E
00:00	0	0	Free-return point	Free-return point
00:40	287	390	25:14	60:39
01:20	583	801	25:09	60:56
02:00	888	1233	24:59	61:18
02:40	1202	1688	24:51	61:51
03:20	1526	2164	24:43	62:32
04:00	1861	2665	24:34	63:25

(b) REFSMMAT

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \begin{bmatrix} 0.93365763 & -.34652012 & -.090594014 \\ -.070754927 & -.42639754 & 0.90176433 \\ -.35110853 & -.83552916 & -.42262731 \end{bmatrix}$$

(c) IMU gimbal angles

Roll, deg	180.21
Pitch, deg	57.09
Yaw, deg	3.56

TABLE 8-III.- EFFECTS OF PITCH DRIFTS ON AN LOI MODE III ABORT MANEUVER[a]

Pitch drift,[b] deg	ΔV for midcourse corrections, fps
15	9.4
10	11.1
5	14.0
0	0.0
-5	30.2
-10	78.7
-15	145.5

[a]Data for SPS shutdown at 320 sec.

[b]Value at DPS cutoff.

TABLE 8-IV.- EFFECTS OF PITCH DRIFTS ON AN LOI

MODE II ABORT MANEUVER[a]

Pitch drift,[b] deg	Intermediate ellipse orientation to EML, deg	Time to pericynthion, hr	Altitude of pericynthion, n. mi.	ΔV_{TOTAL},[c] fps
10.0	0.3	15.17	50.1	1847.0
5.0	-1.1	15.50	87.1	1872.3
0.0	-2.5	15.88	124.6	1893.0
-5.0	-4.1	16.31	162.5	1926.6
-10.0	-5.6	16.81	200.4	1947.3

[a]Data for an SPS shutdown at 140 sec.

[b]Value at DPS cutoff.

[c]ΔV_{TOTAL} includes 600 fps for first maneuver, ΔV available for maneuvers = 2300 fps.

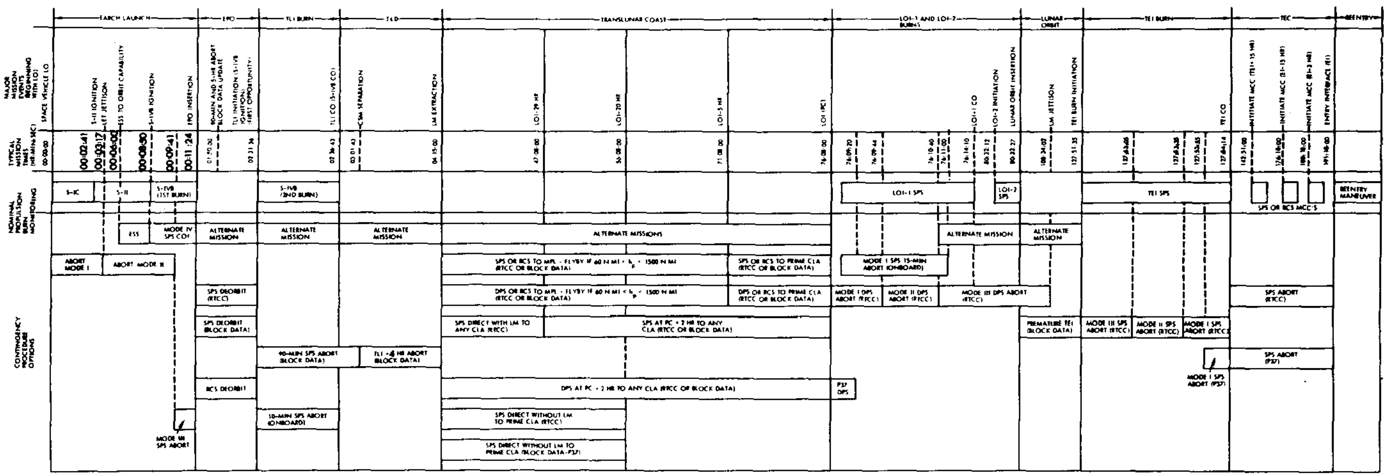
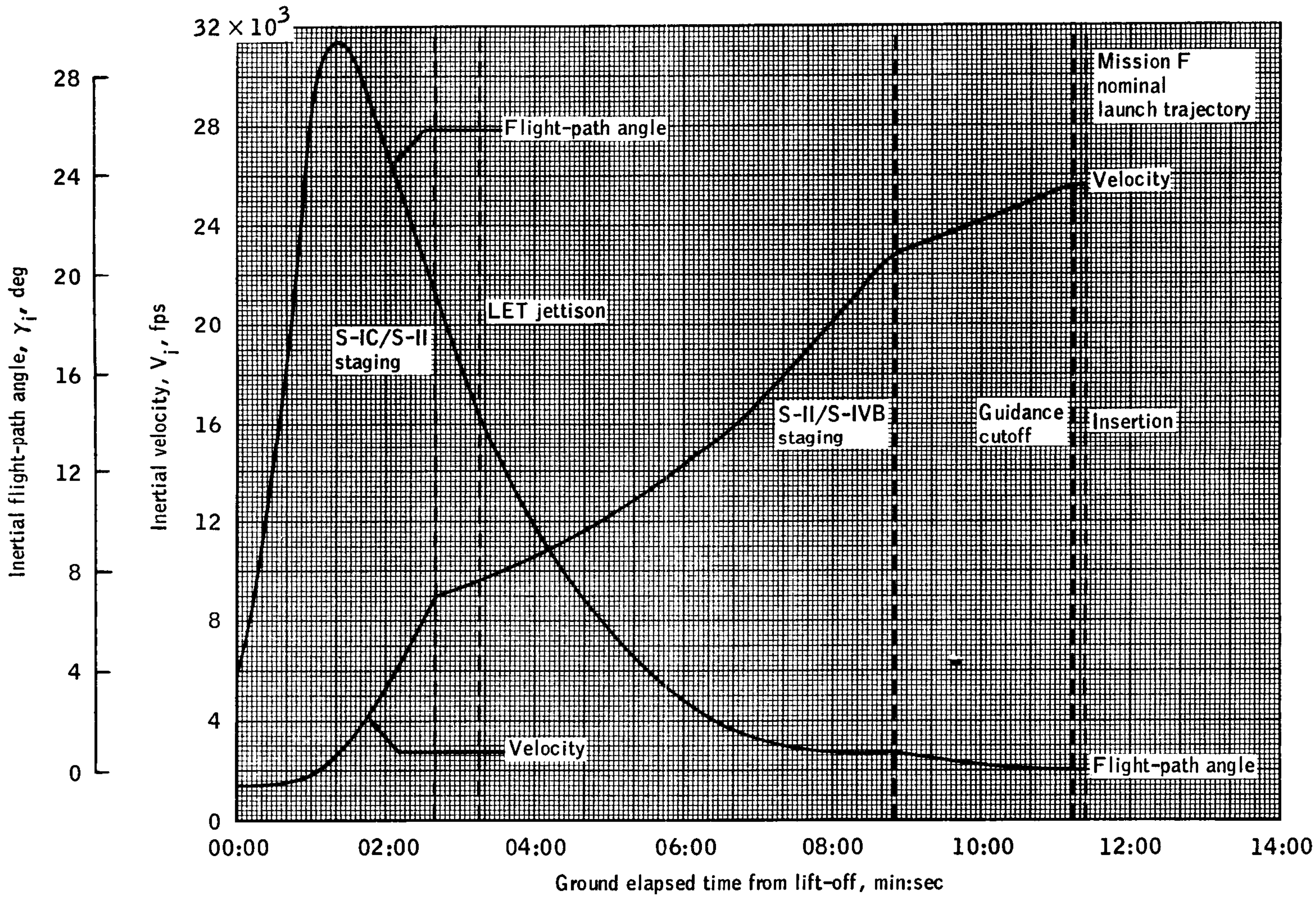
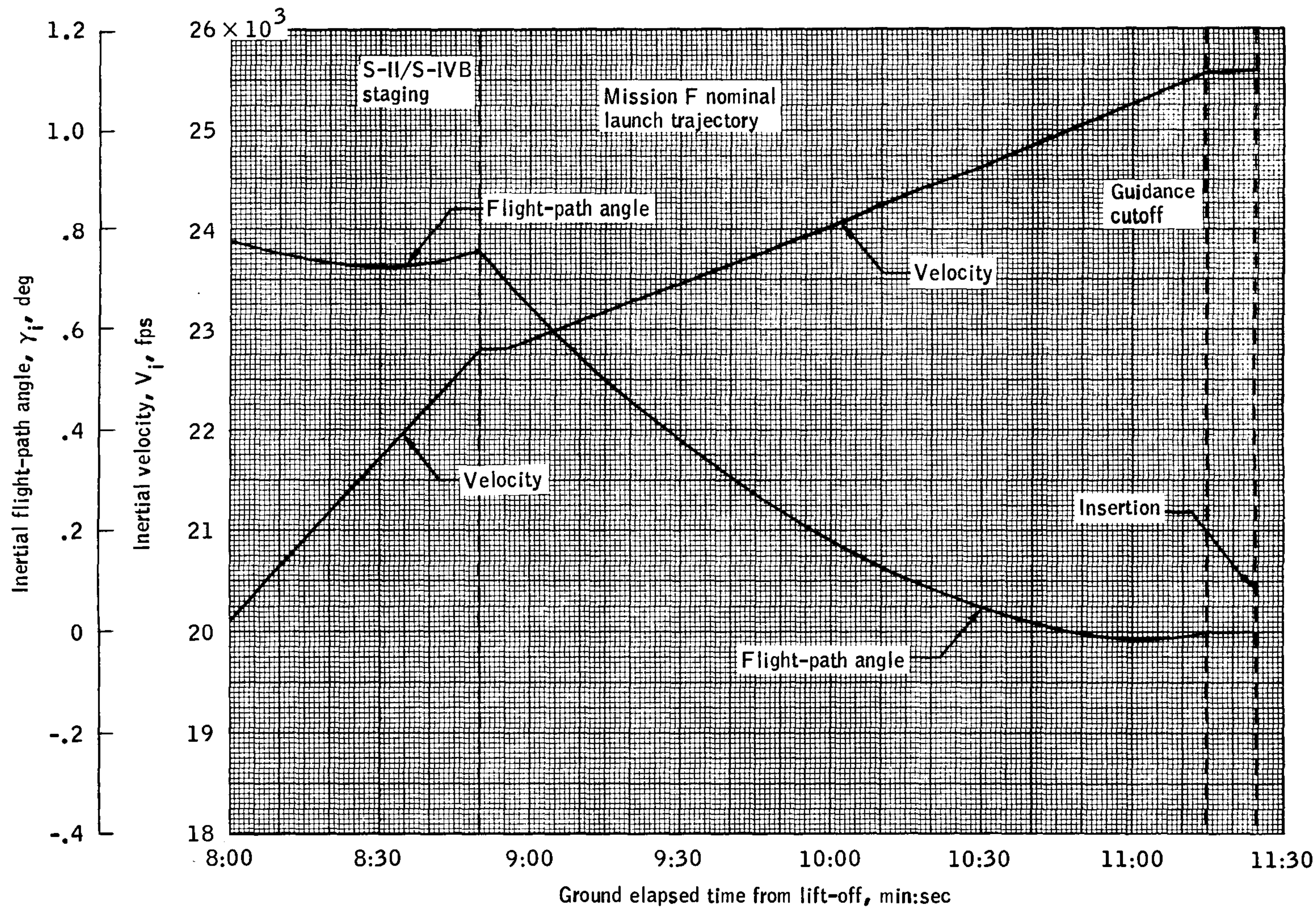


Figure 2-1.- The relationship of the nominal Apollo 10 mission events and the operational abort modes.



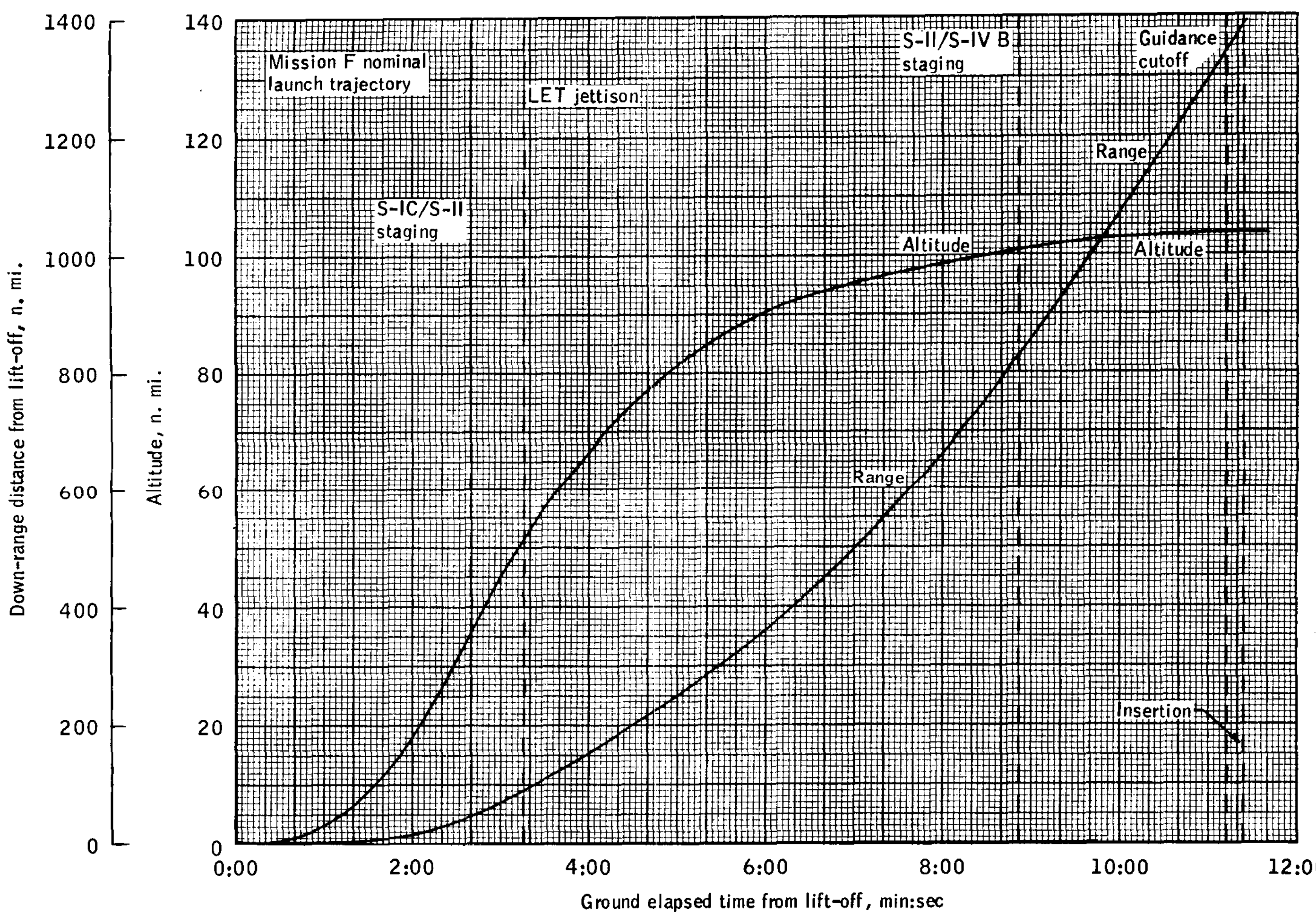
(a) Complete launch.

Figure 5-1.- Inertial velocity and inertial flight-path angle along the nominal launch trajectory.



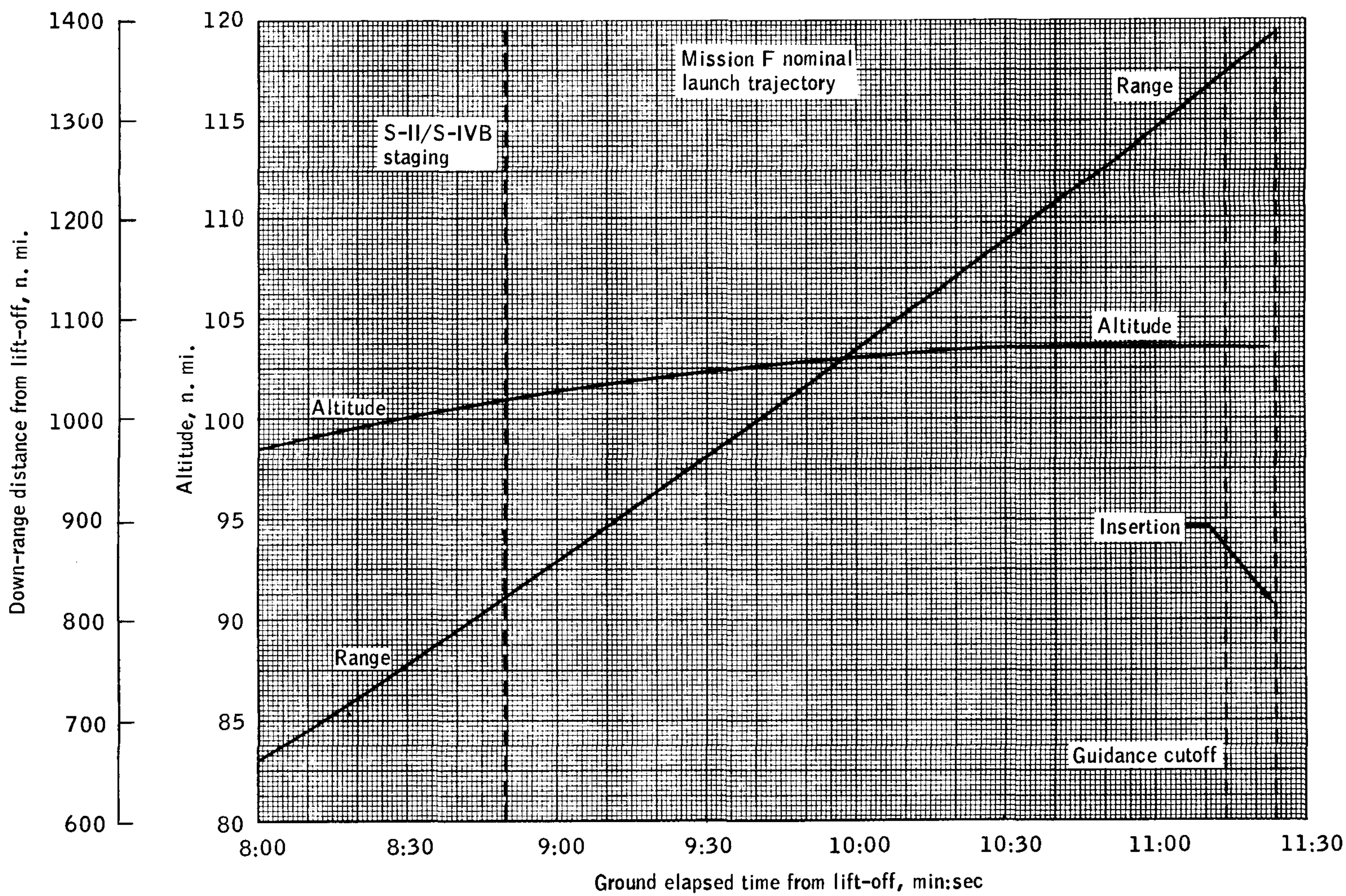
(b) Near insertion.

Figure 5-1.- Concluded.



(a) Complete launch.

Figure 5-2.- Down-range distance and altitude along the nominal launch trajectory.



(b) Near insertion.

Figure 5-2.- Concluded.

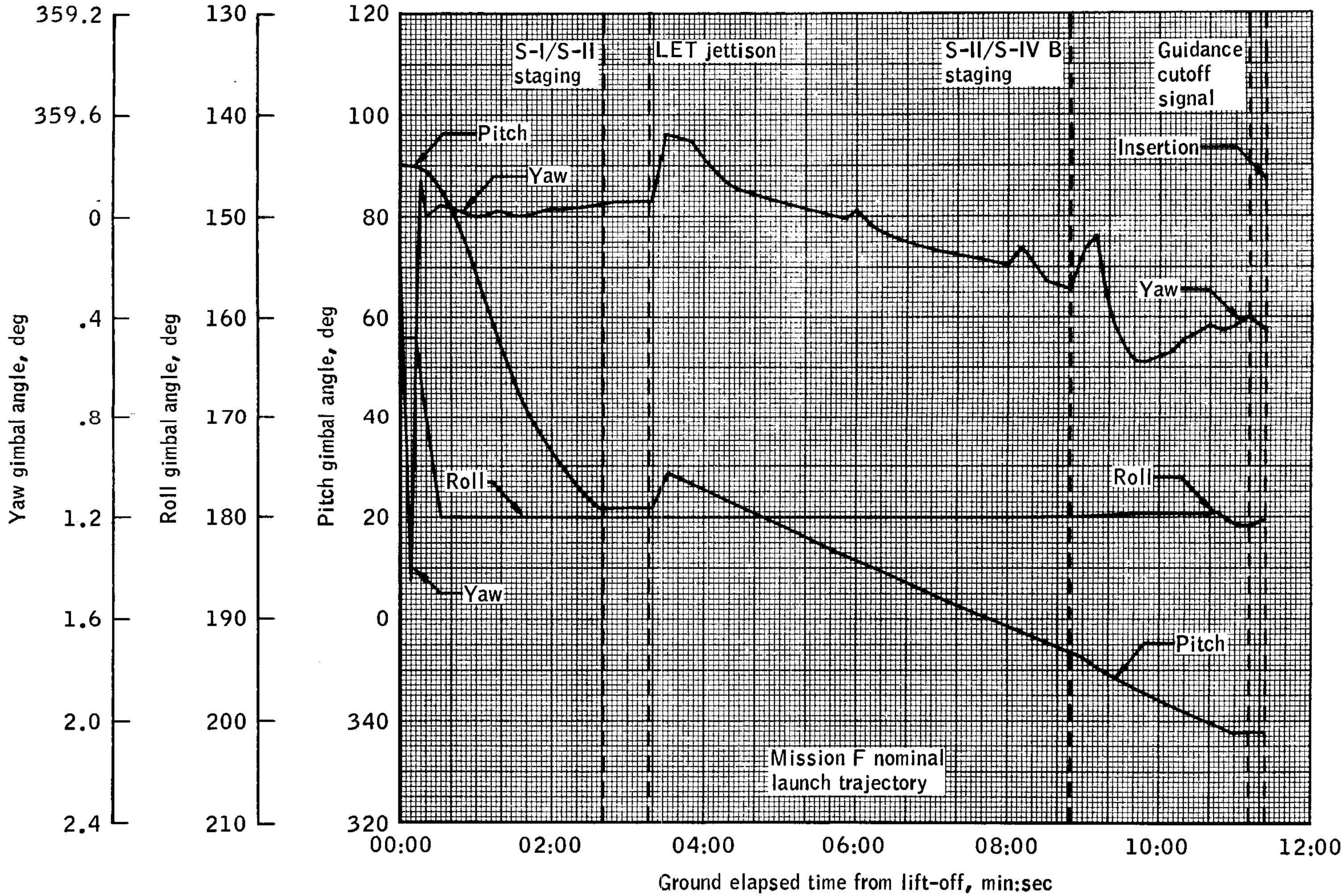


Figure 5-3.- Spacecraft IMU gimbal angle readouts along the nominal launch trajectory.

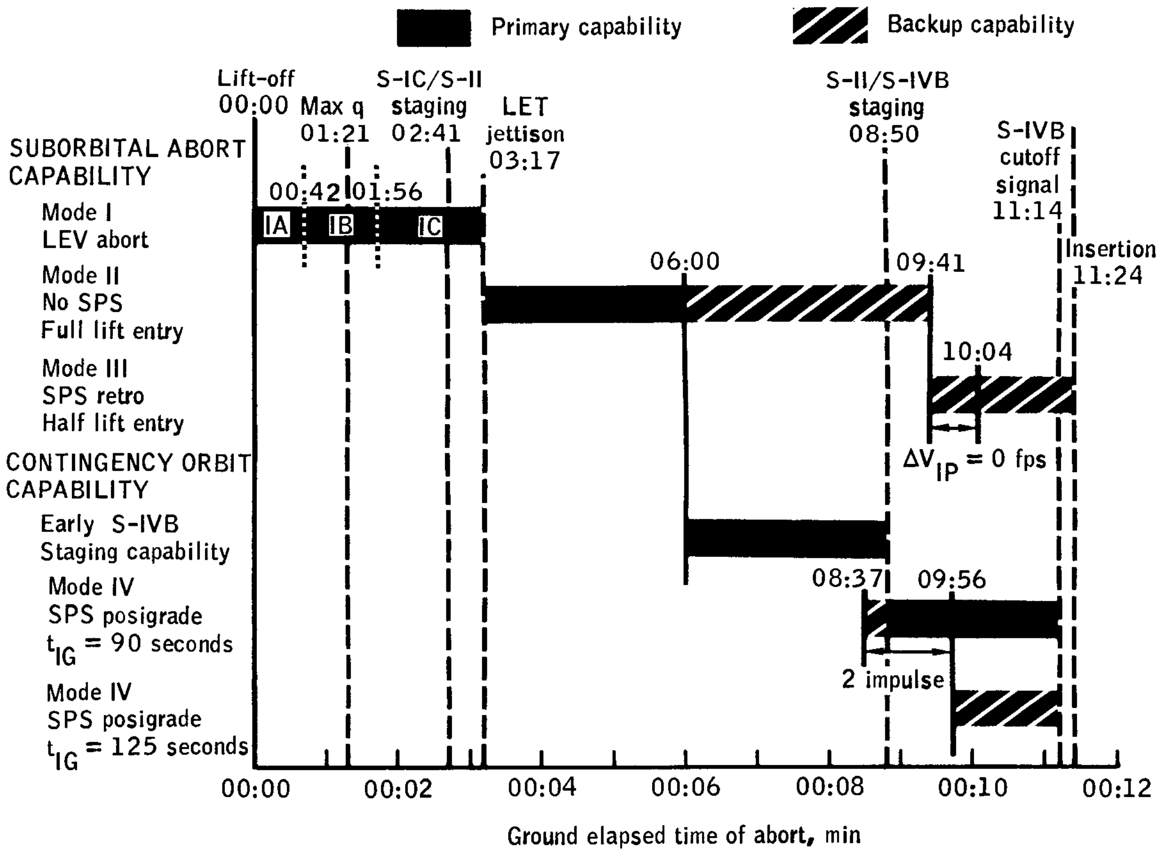


Figure 5-4.- Mission F launch timeline and contingency capability.

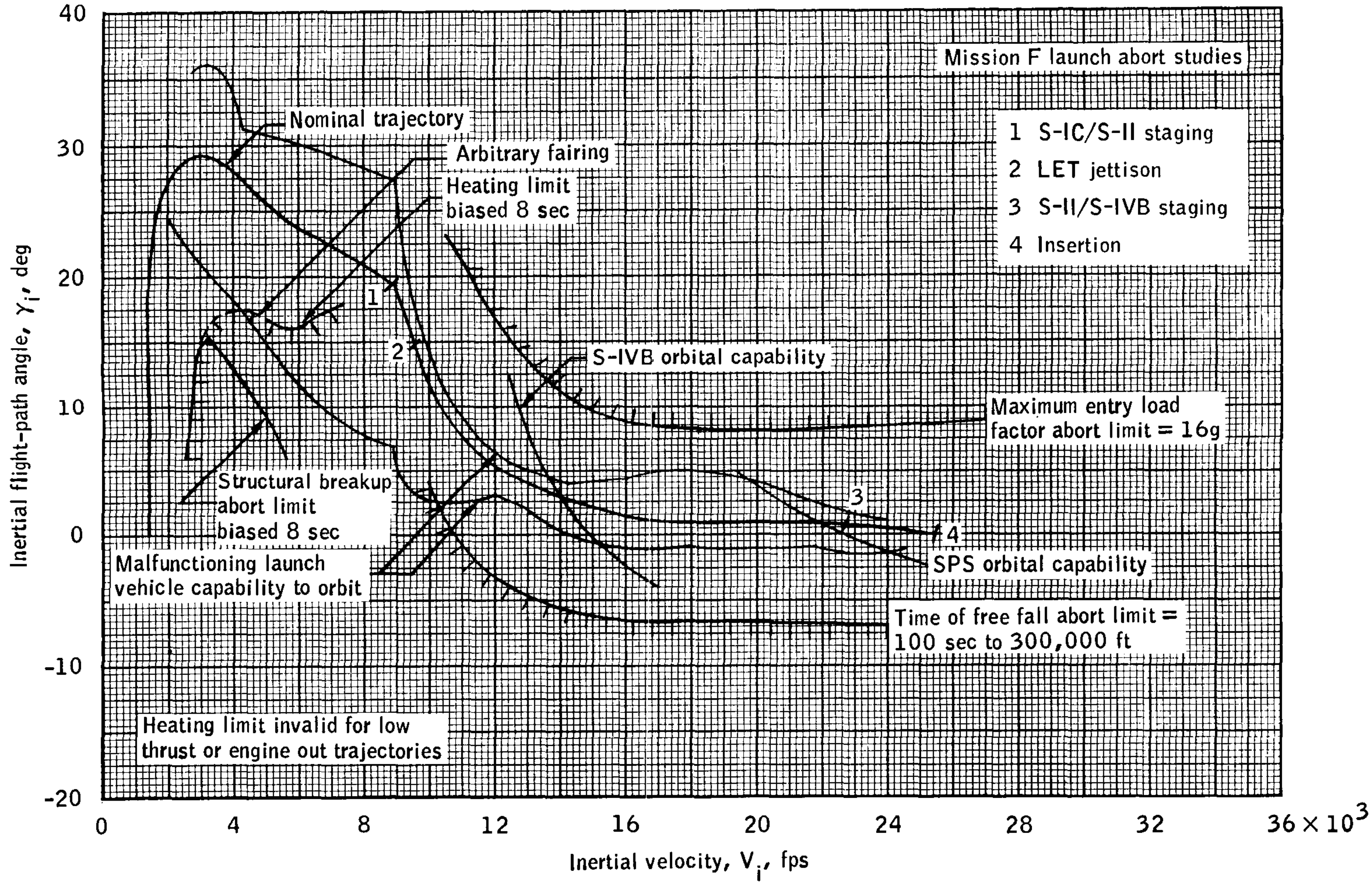


Figure 5-5.- Launch abort and capability limits.

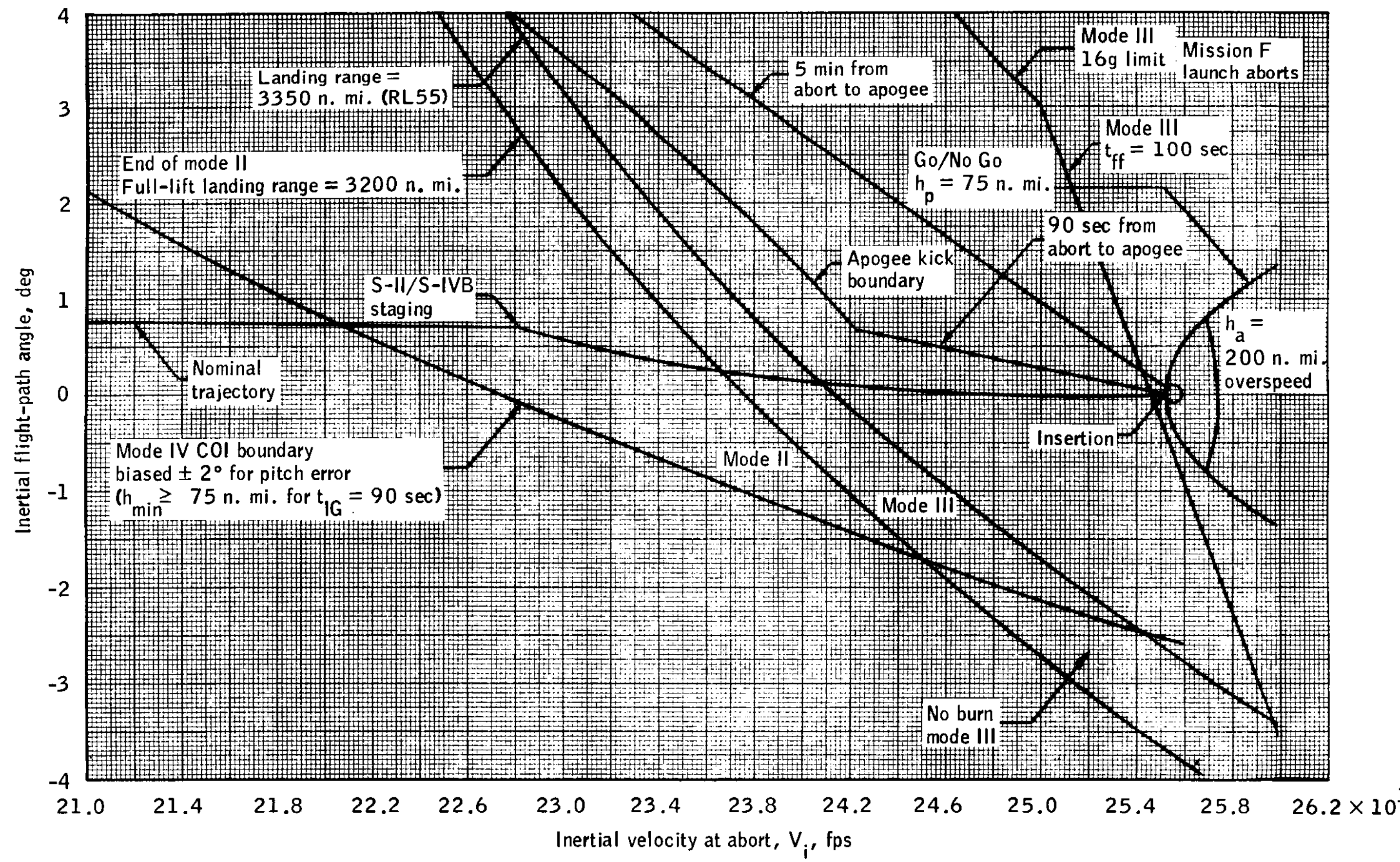


Figure 5-6.- Near-insertion abort mode overlap.

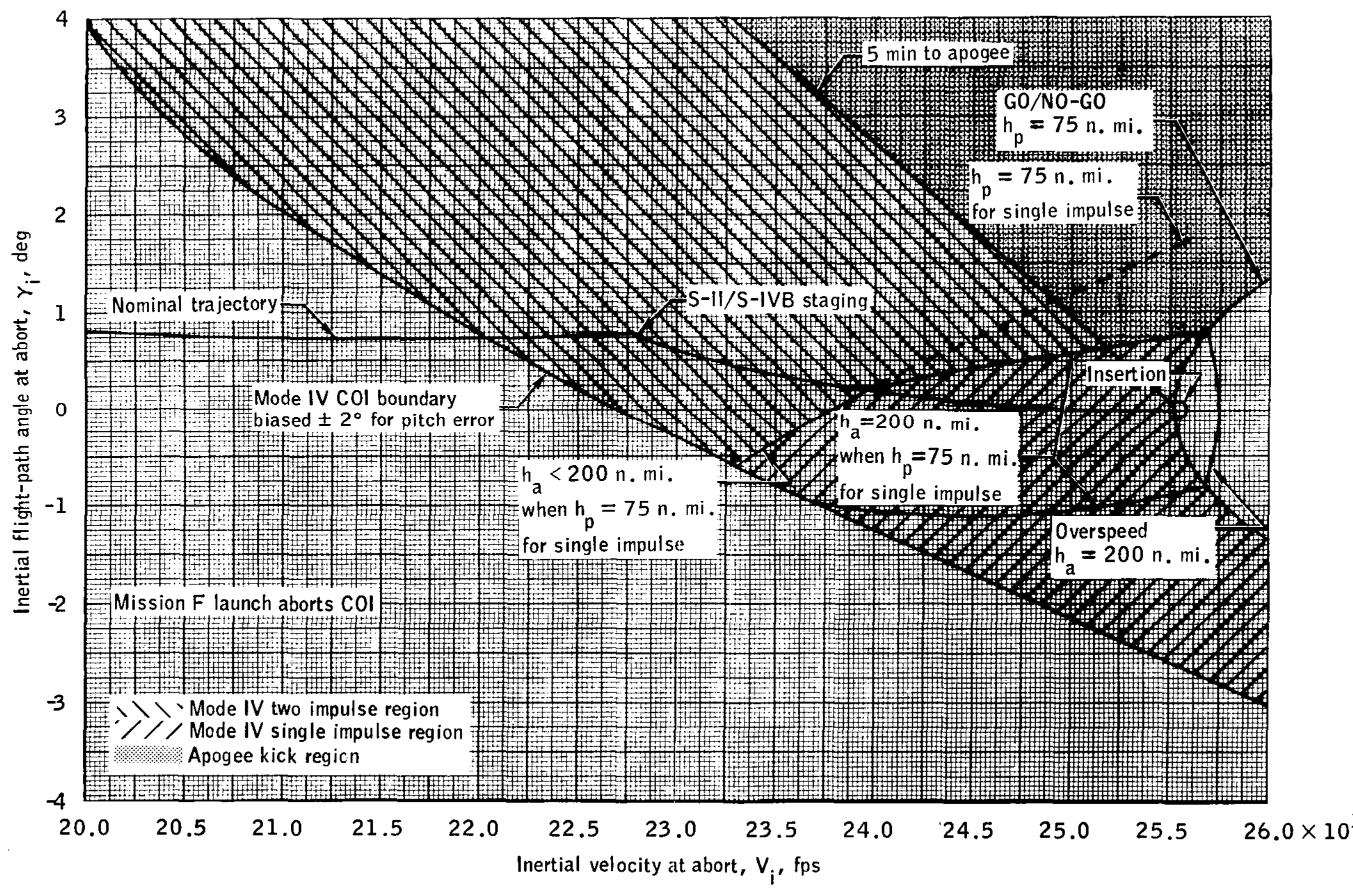


Figure 5-7.- SPS contingency orbital insertion capability.

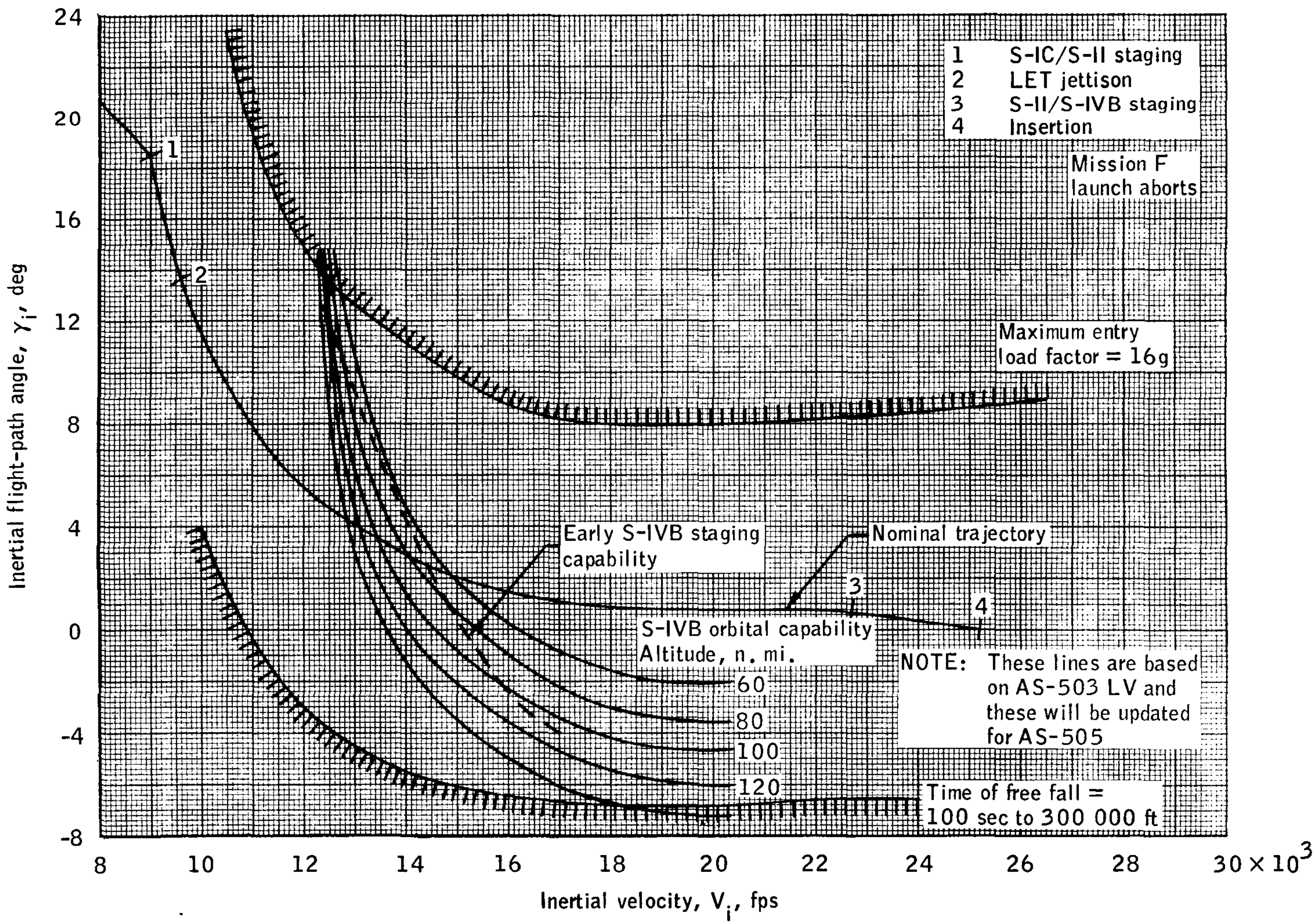


Figure 5-8.- S-IVB early staging to orbit capability variation with altitude.

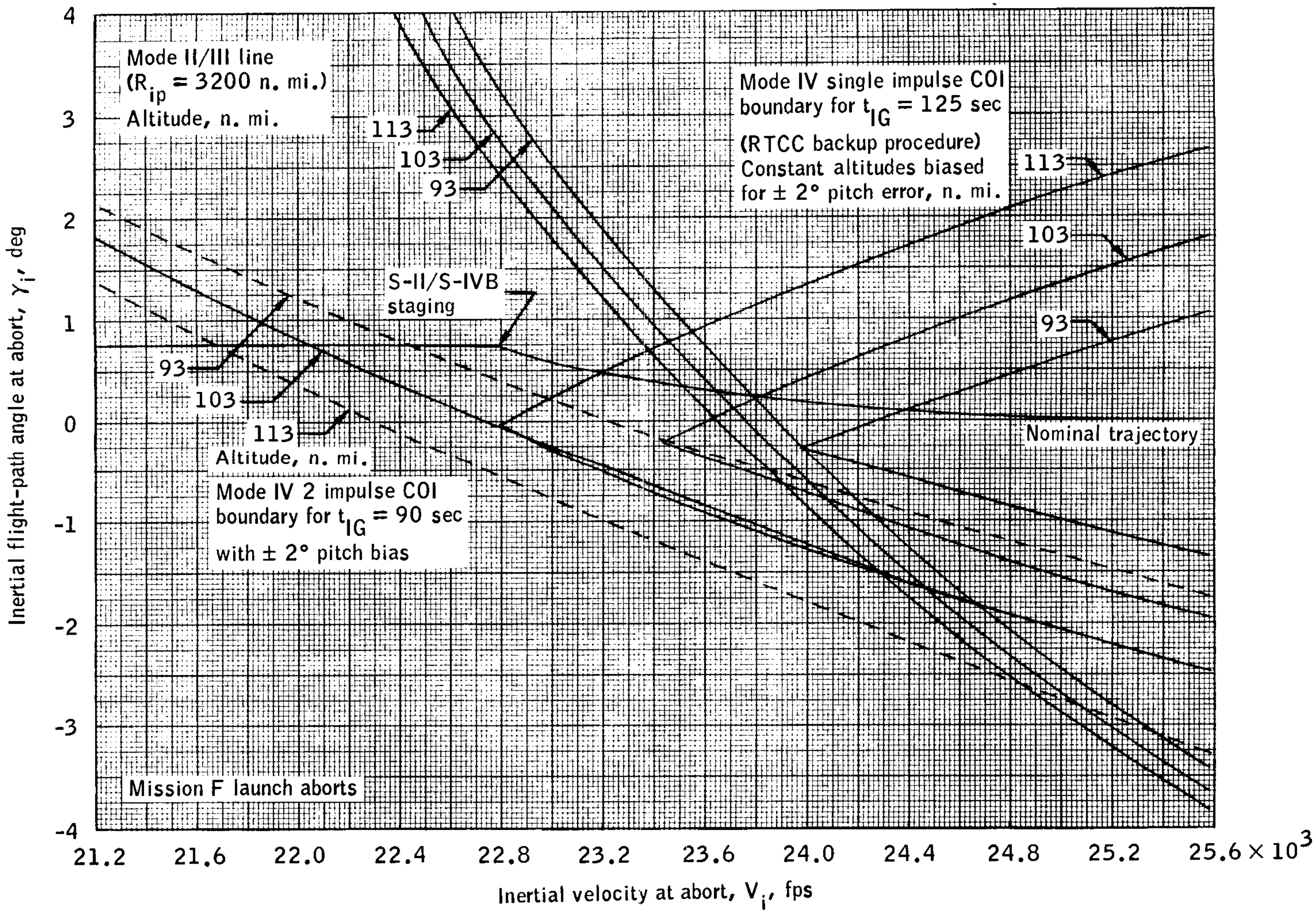


Figure 5-9.- SPS contingency orbital insertion capability variation with altitude.

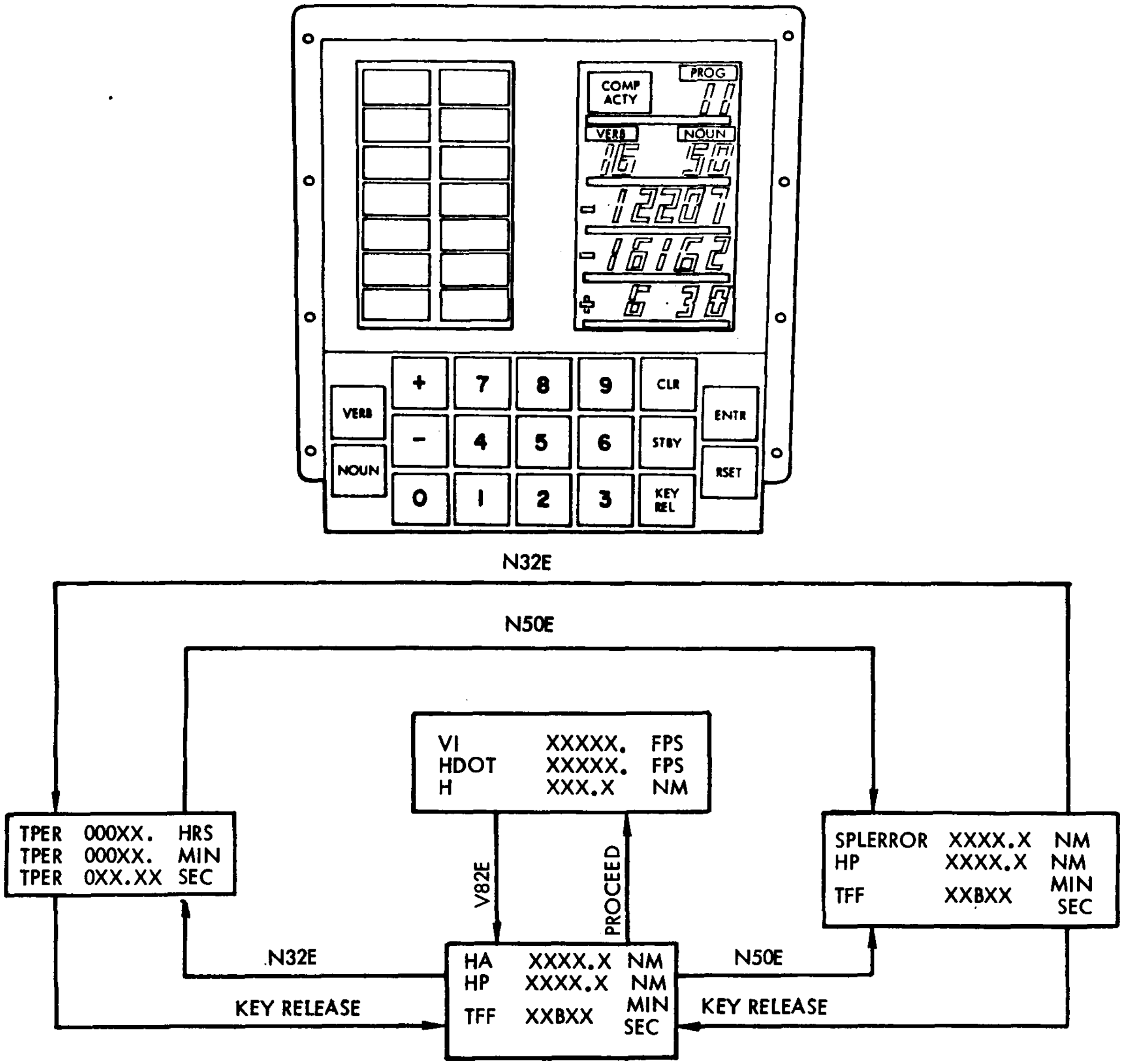


Figure 5-10.- AGC display keyboard and display parameters.

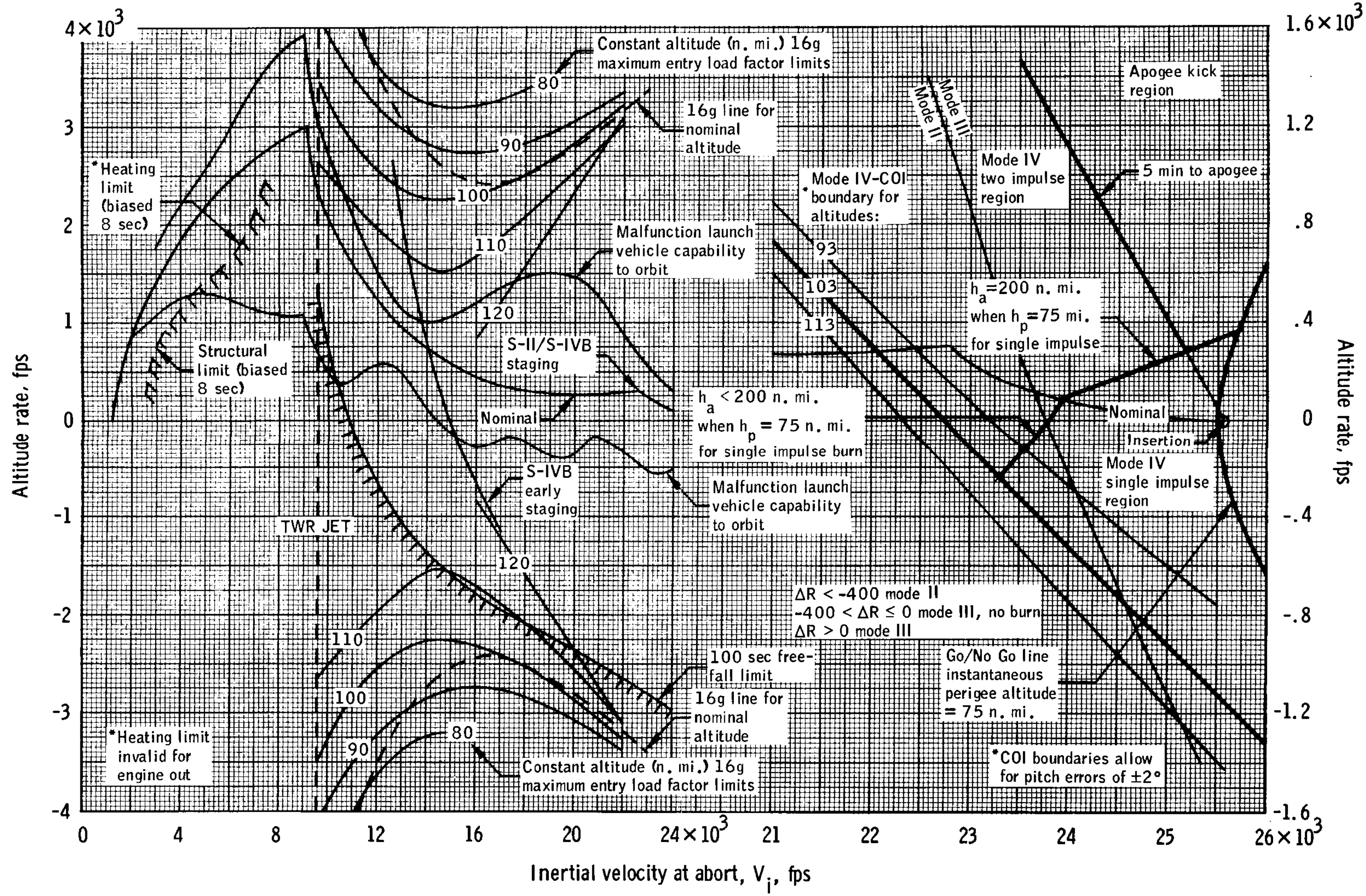


Figure 5-11.- Sample no-voice crew chart for the launch phase.

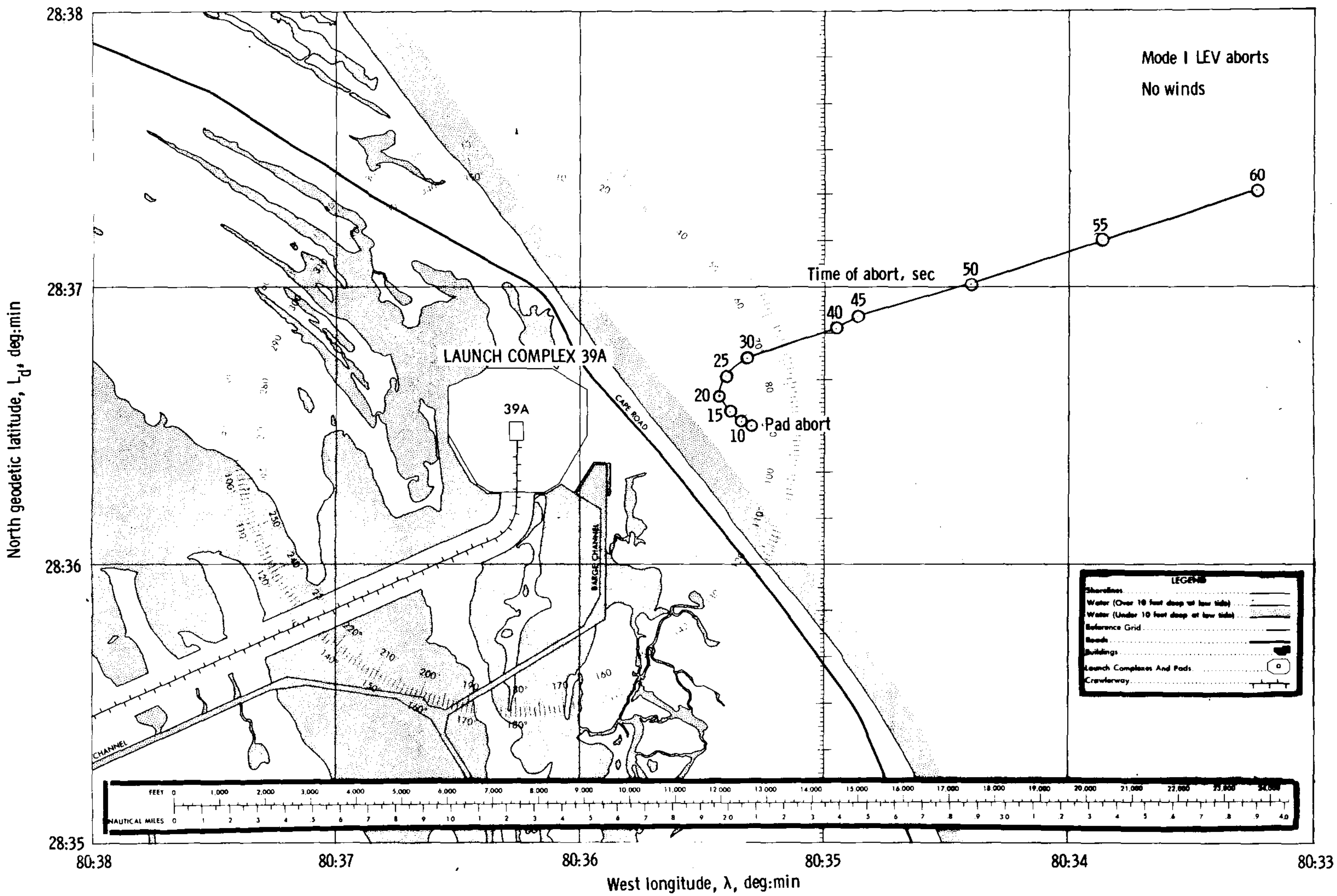


Figure 5-12. - Mode I LEV abort landing points for pad abort through 60 seconds ground elapsed time.

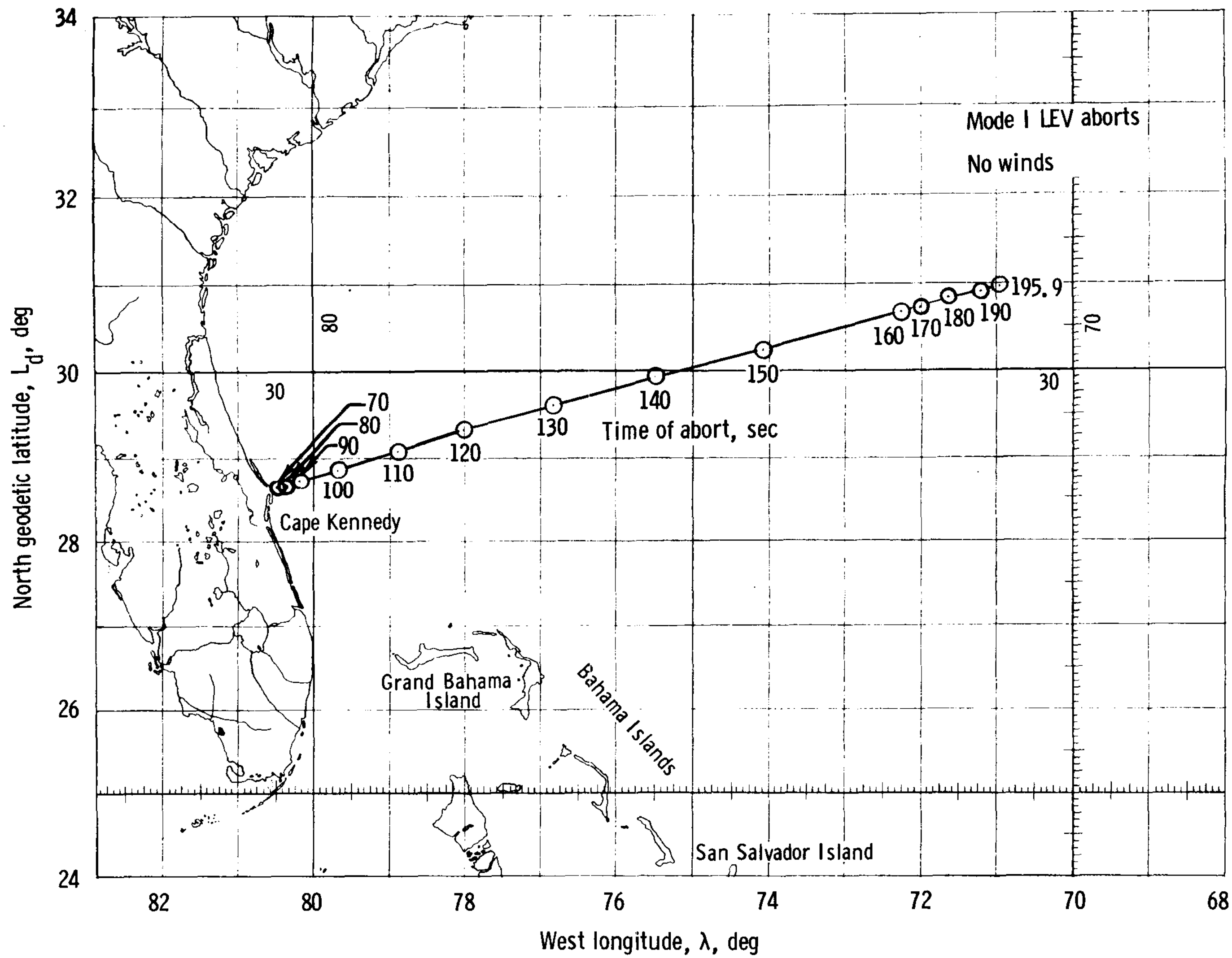


Figure 5-13. - Mode I LEV abort landing points for 70 seconds through 195.9 seconds ground elapsed time.

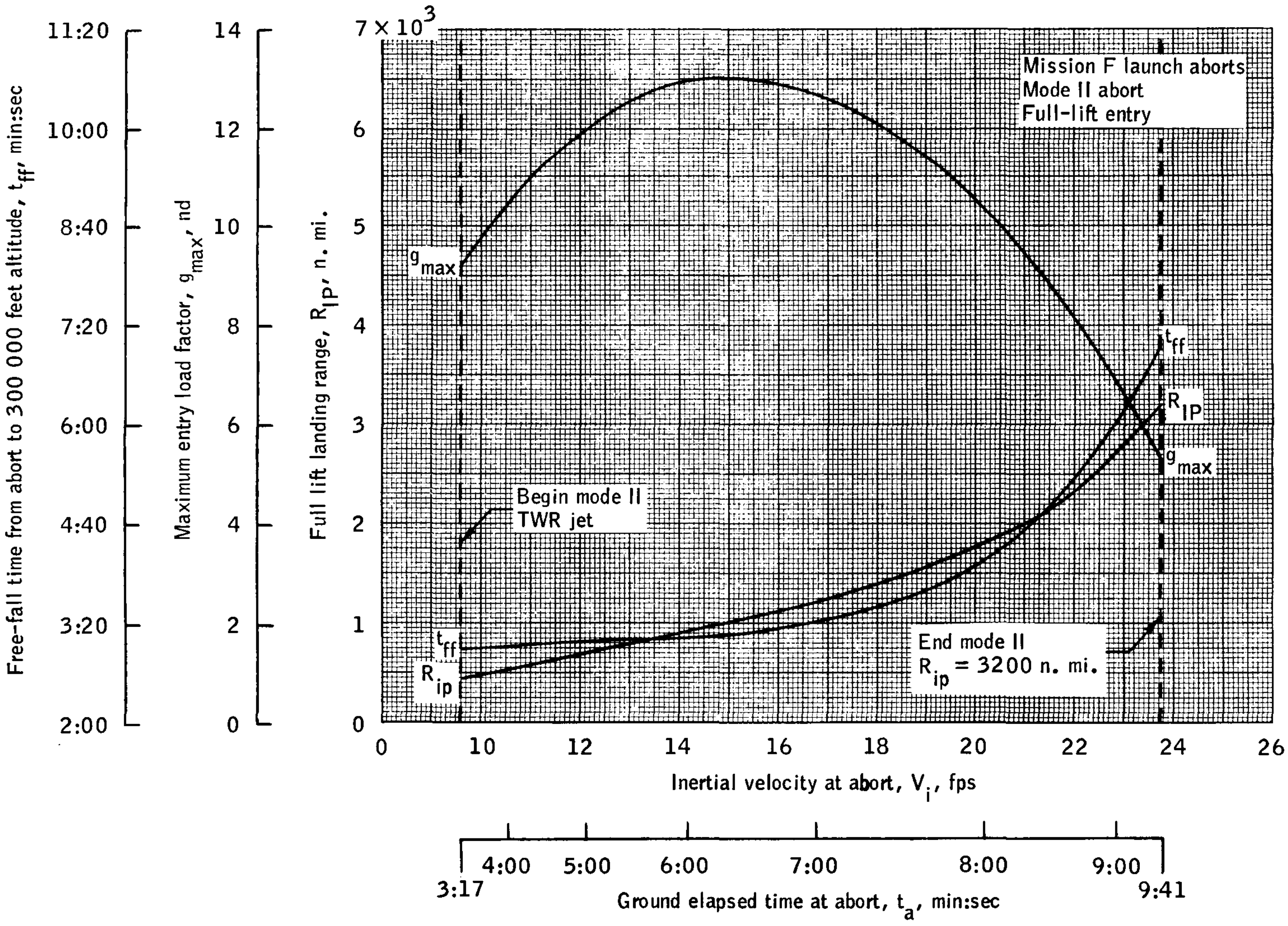


Figure 5-14.- Mode II abort parameters for aborts from the nominal launch trajectory.

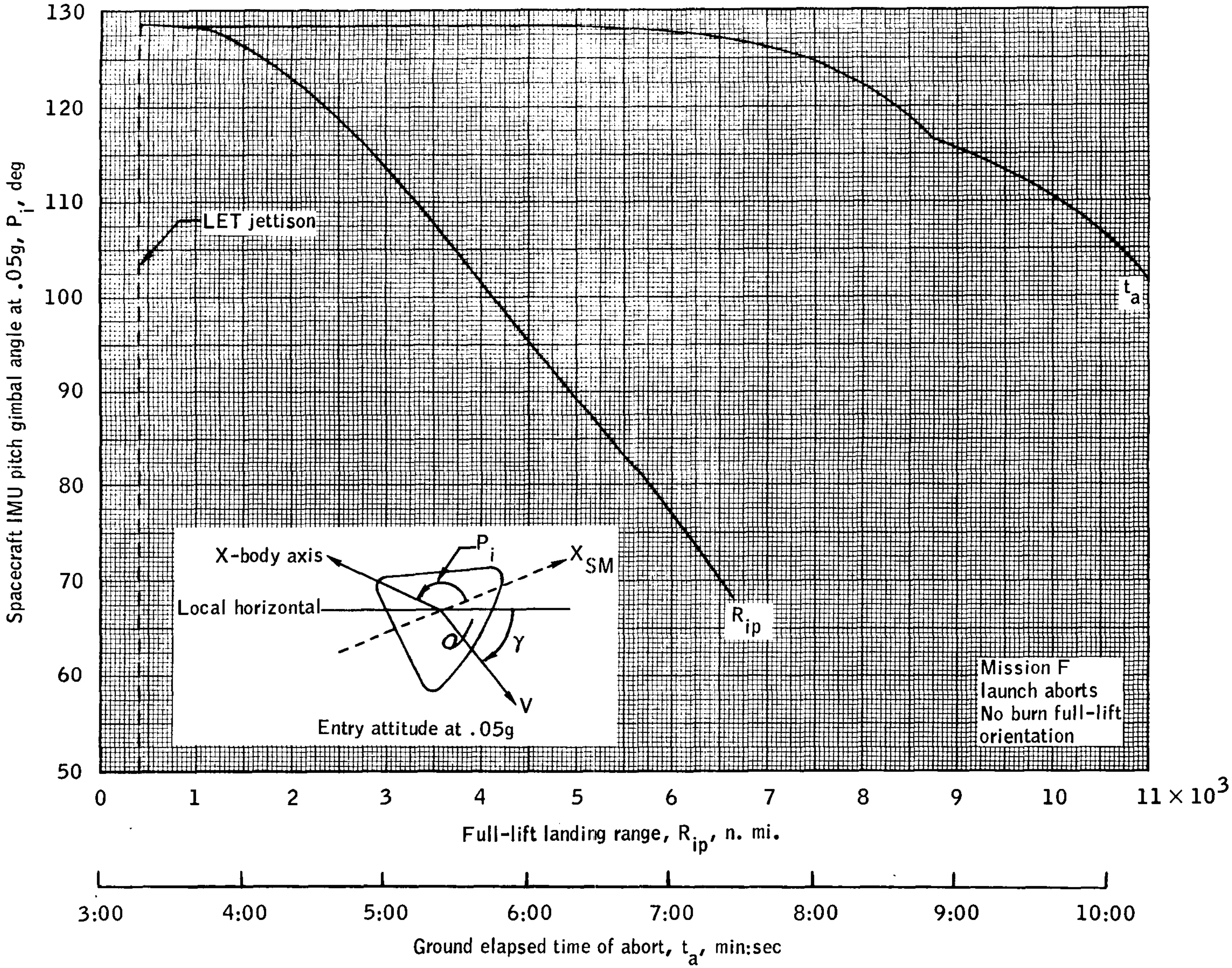


Figure 5-15.- Entry orientation following no burn aborts from the nominal launch trajectory.

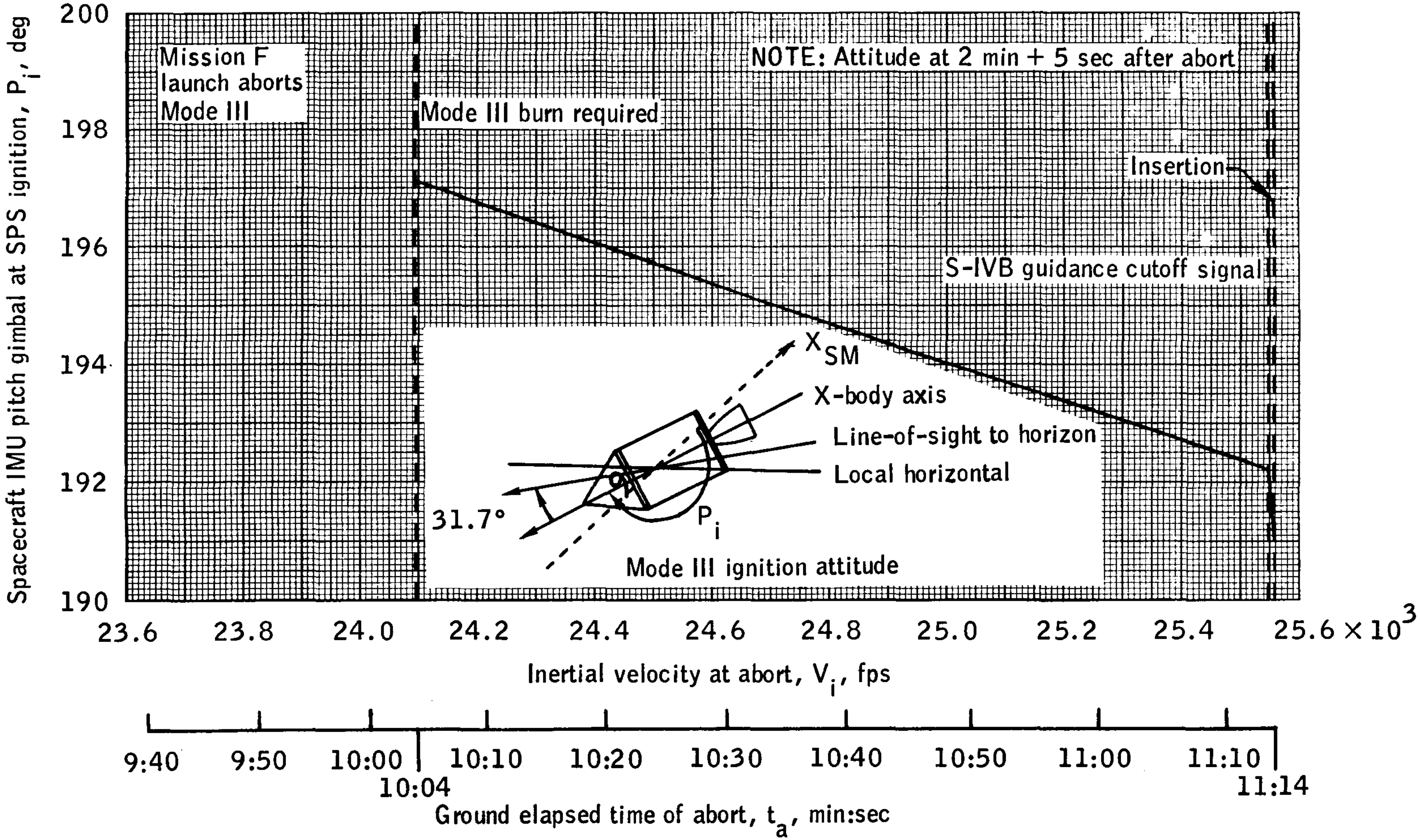


Figure 5-16.- Mode III burn orientation following aborts from the nominal launch trajectory.

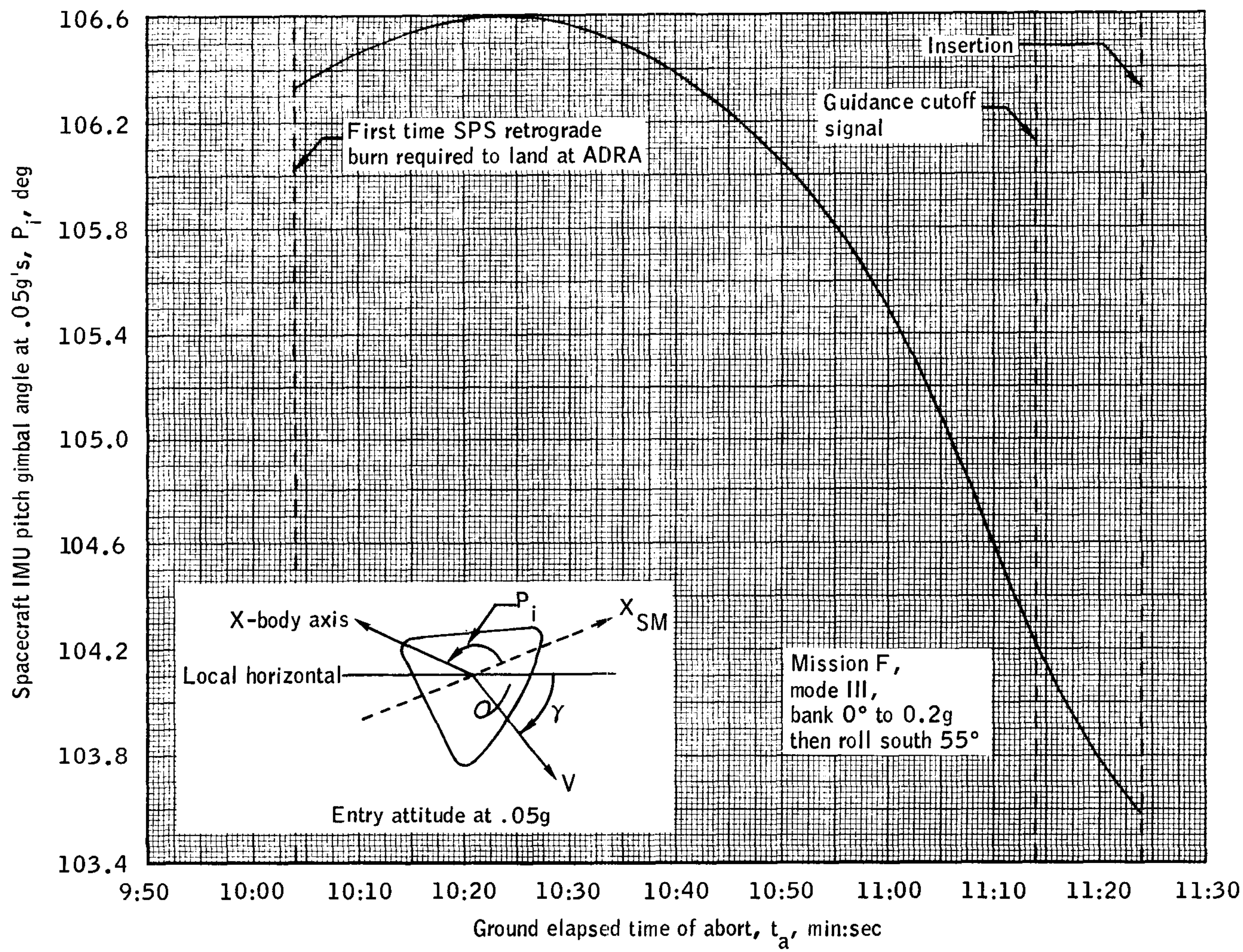


Figure 5-17.- Entry orientation following mode III aborts from the nominal launch trajectory.

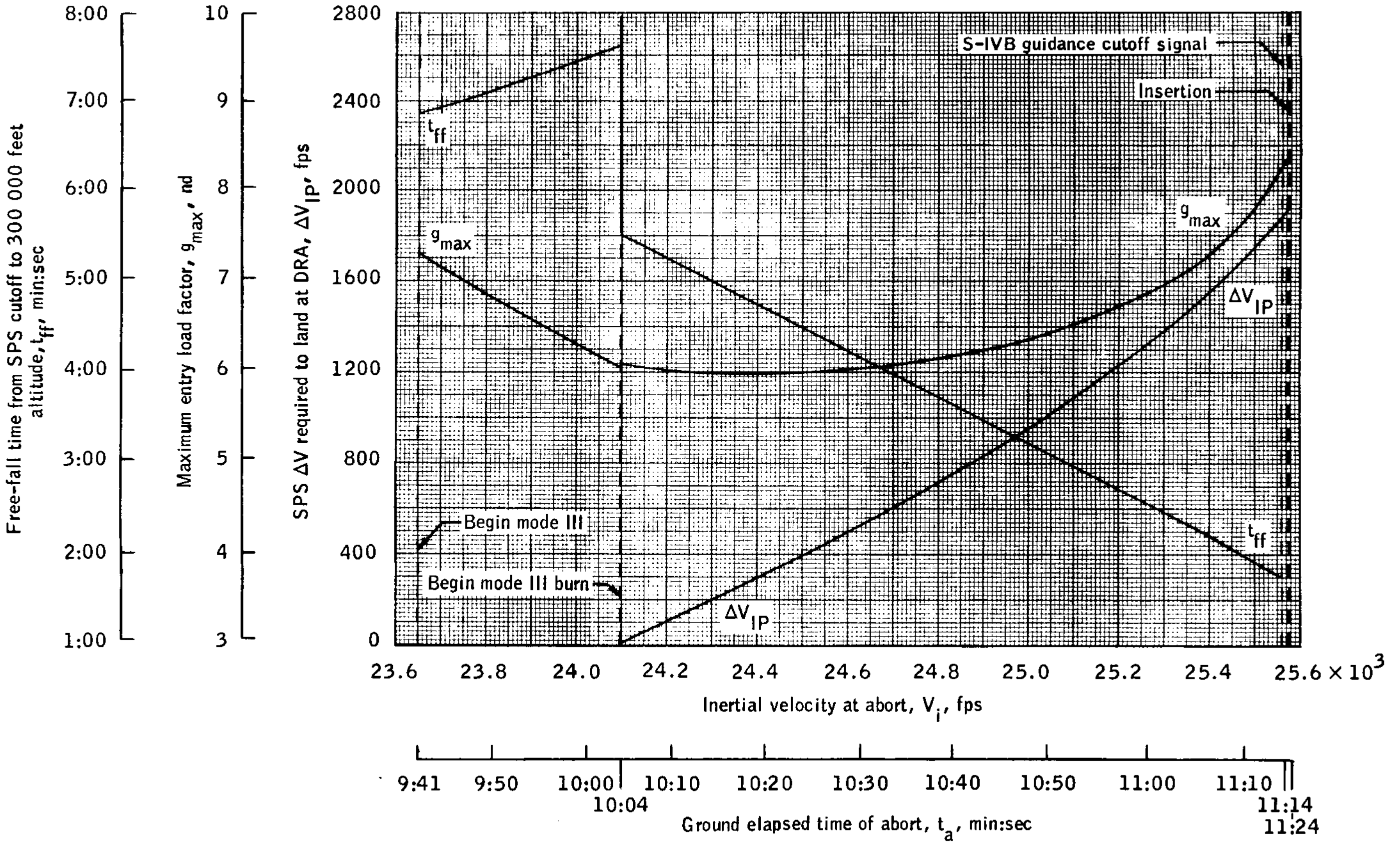


Figure 5-18.- Mode III abort parameters for aborts from the nominal launch trajectory.

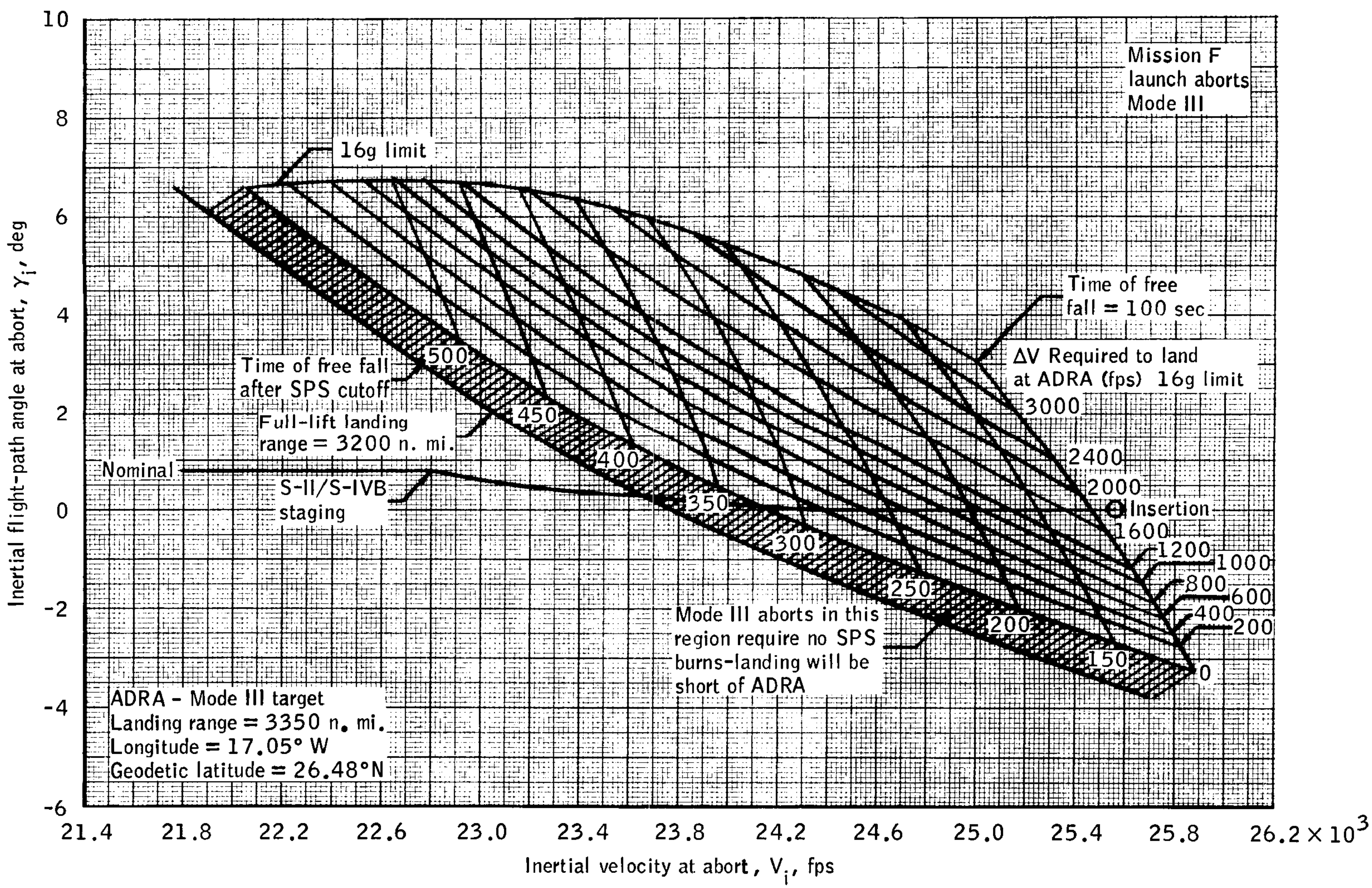


Figure 5-19.- Constant mode III ΔV contours required to land at the Atlantic discrete recovery area.

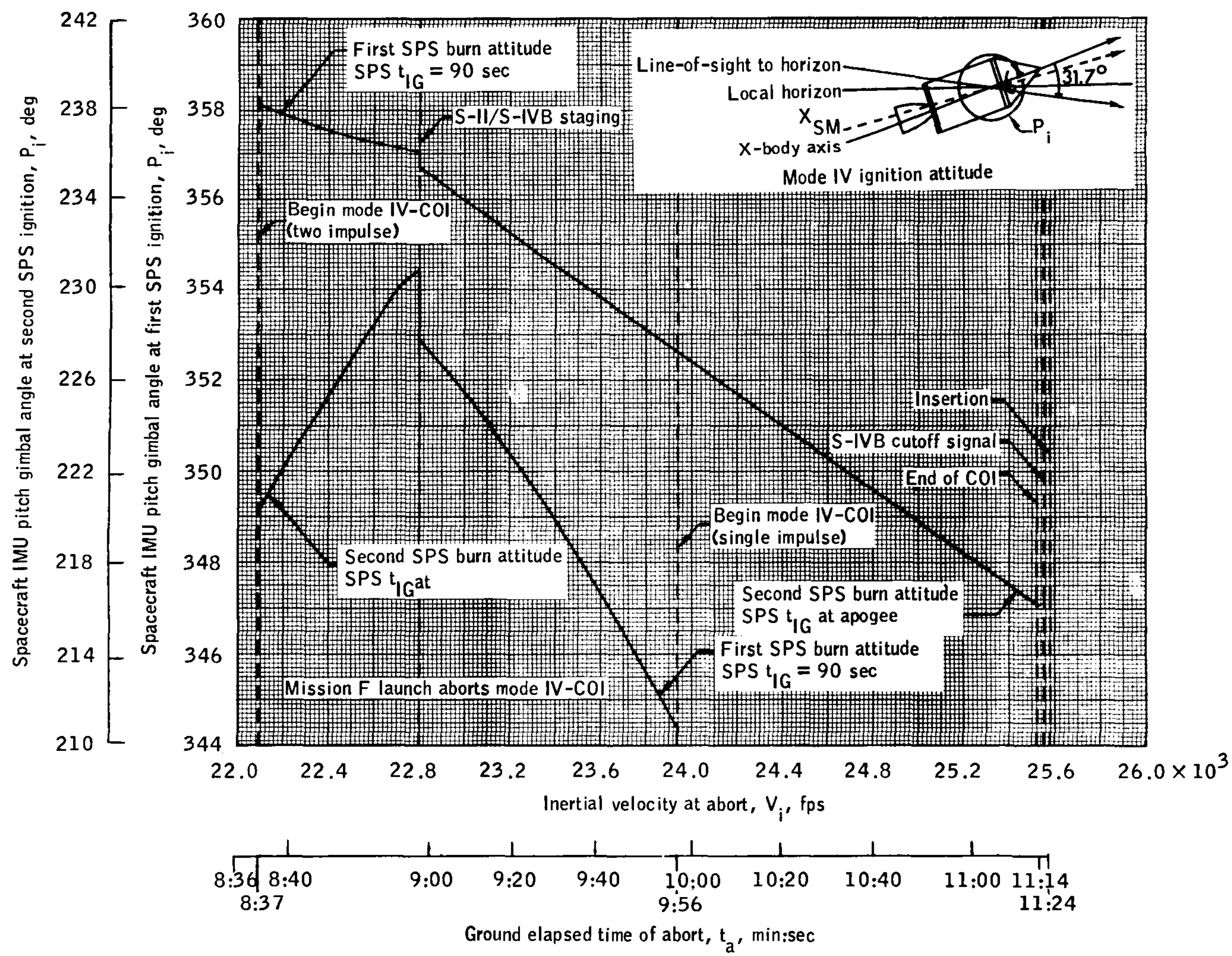


Figure 5-20.- Mode IV burn orientation following aborts from the nominal launch trajectory.

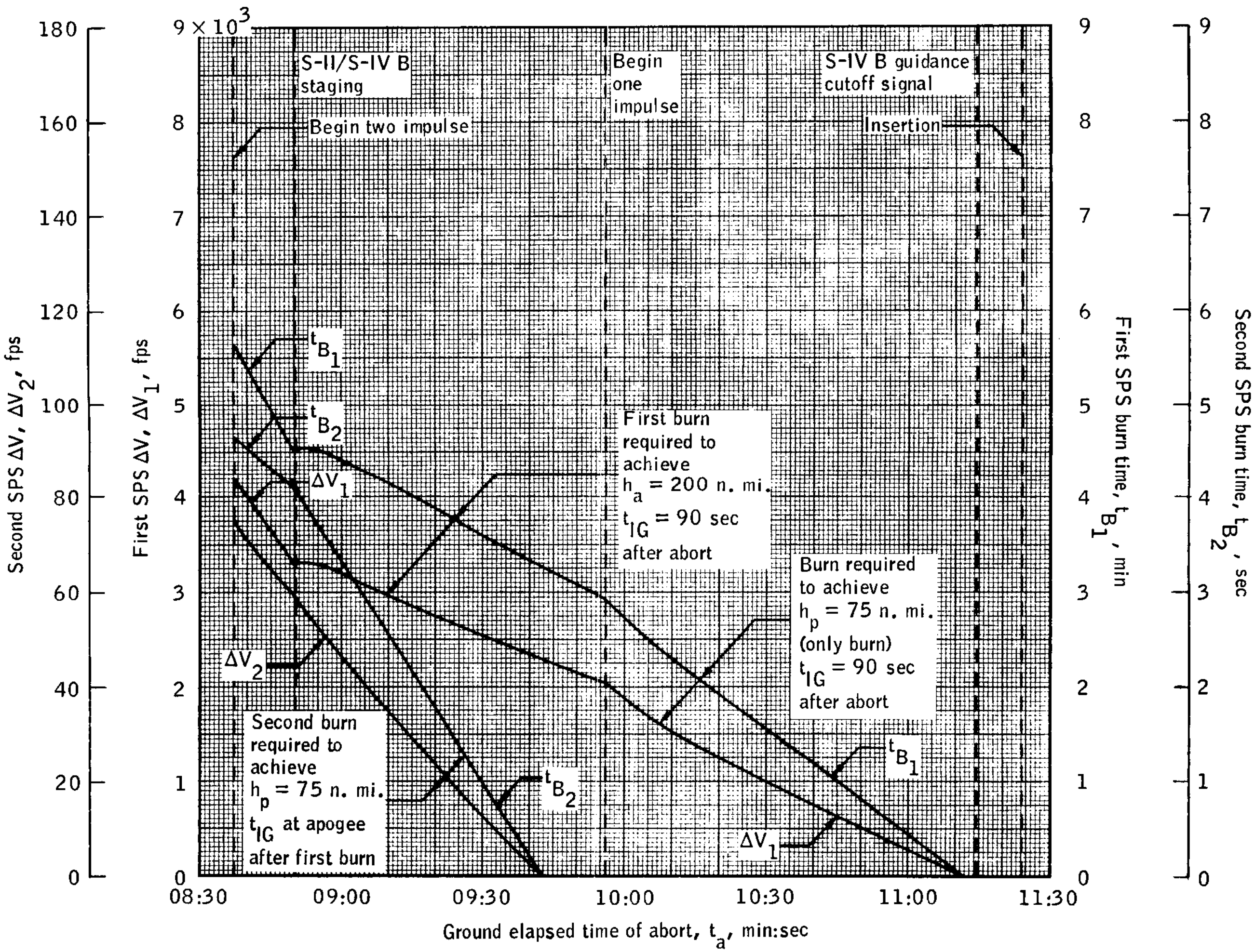


Figure 5-21.- Mode IV abort burn parameters for aborts from the nominal launch trajectory.

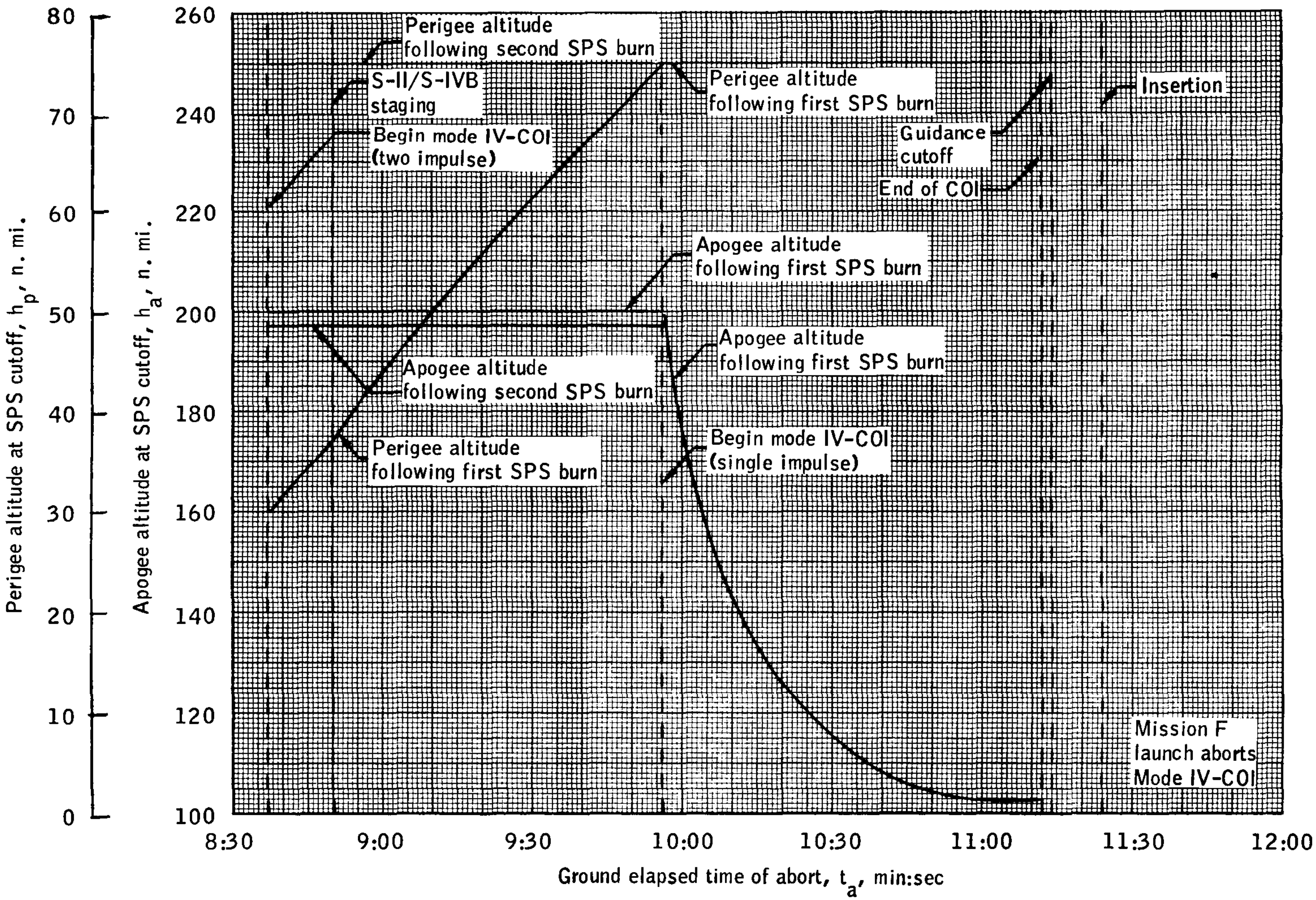


Figure 5-22.- Mode IV abort results for aborts from the nominal launch trajectory.

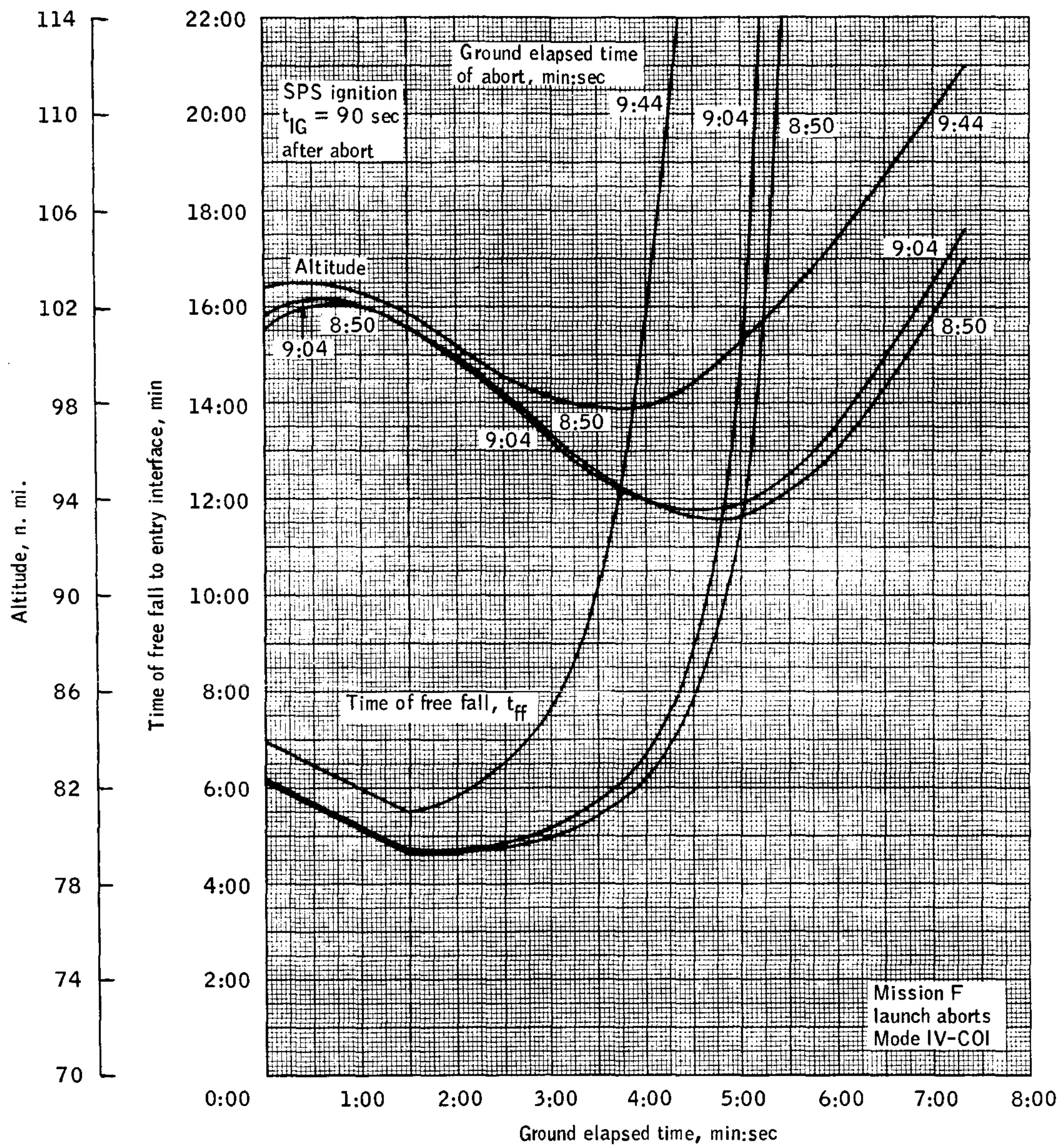


Figure 5-23.- Minimum conditions during typical mode IV abort burns from the nominal trajectory.

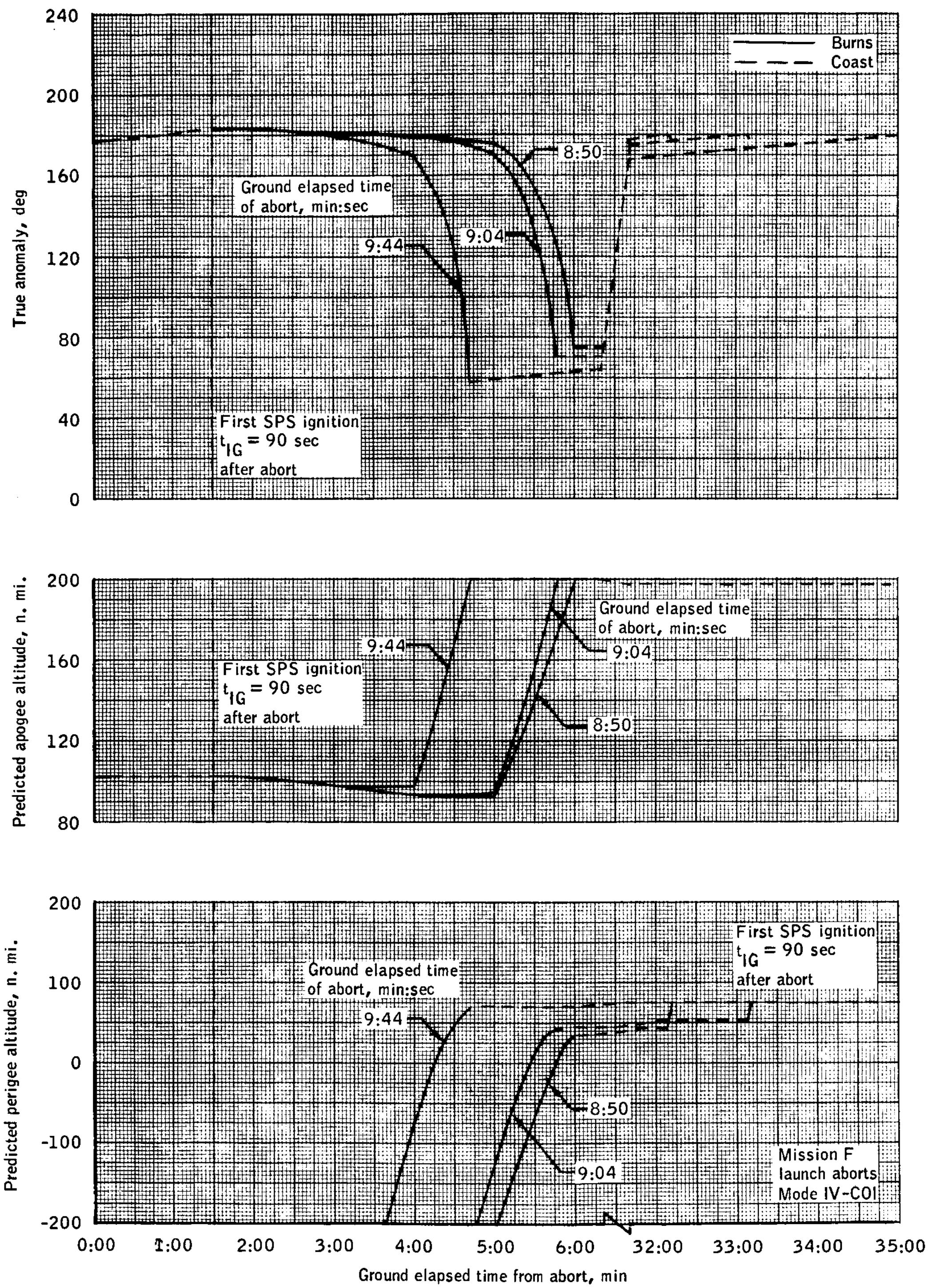


Figure 5-24.- Orbital conditions during typical mode IV abort burns from the nominal trajectory.

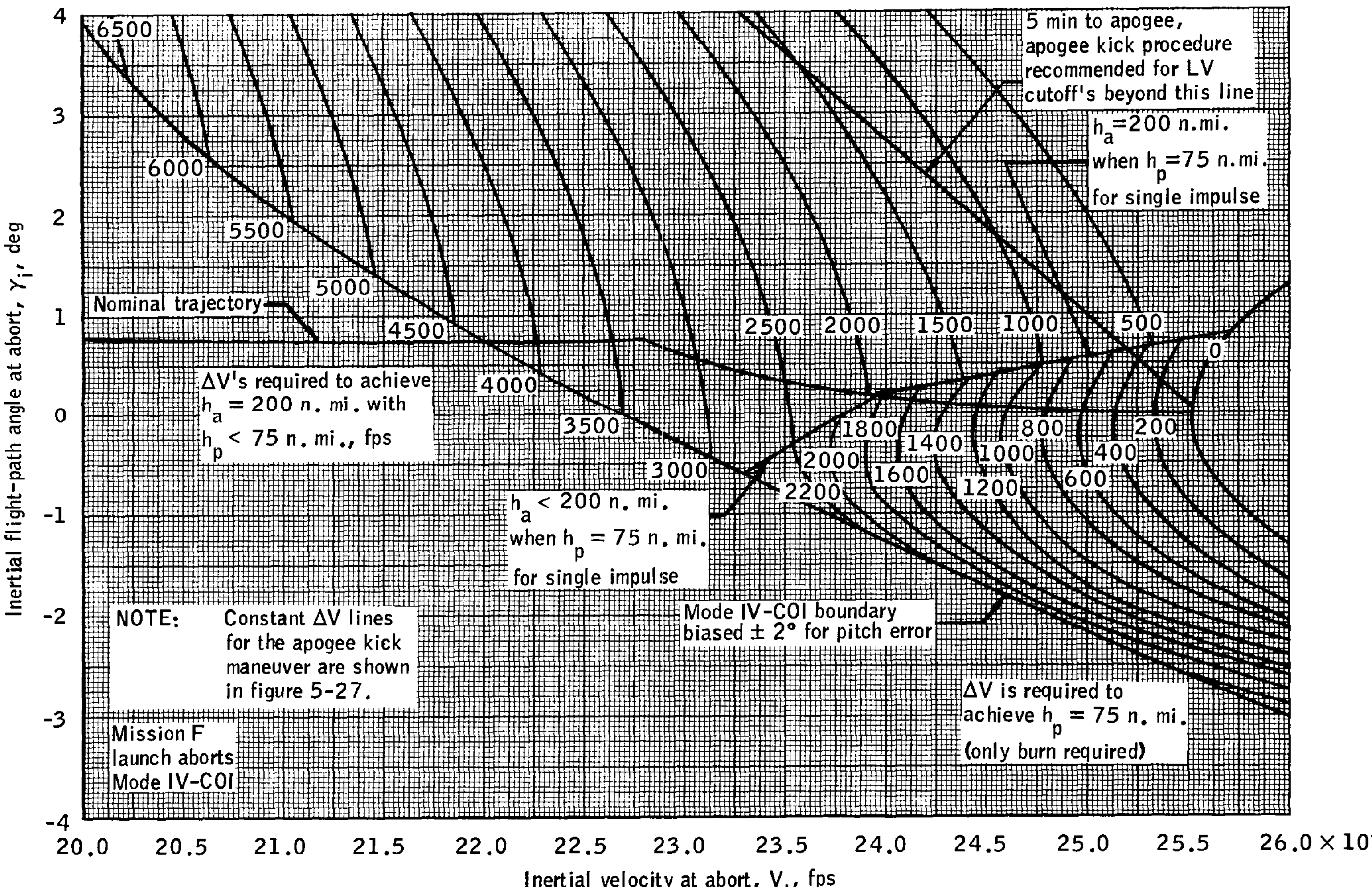


Figure 5-25.- Constant mode IV ΔV contours for the 90-second ignition times, two impulse.

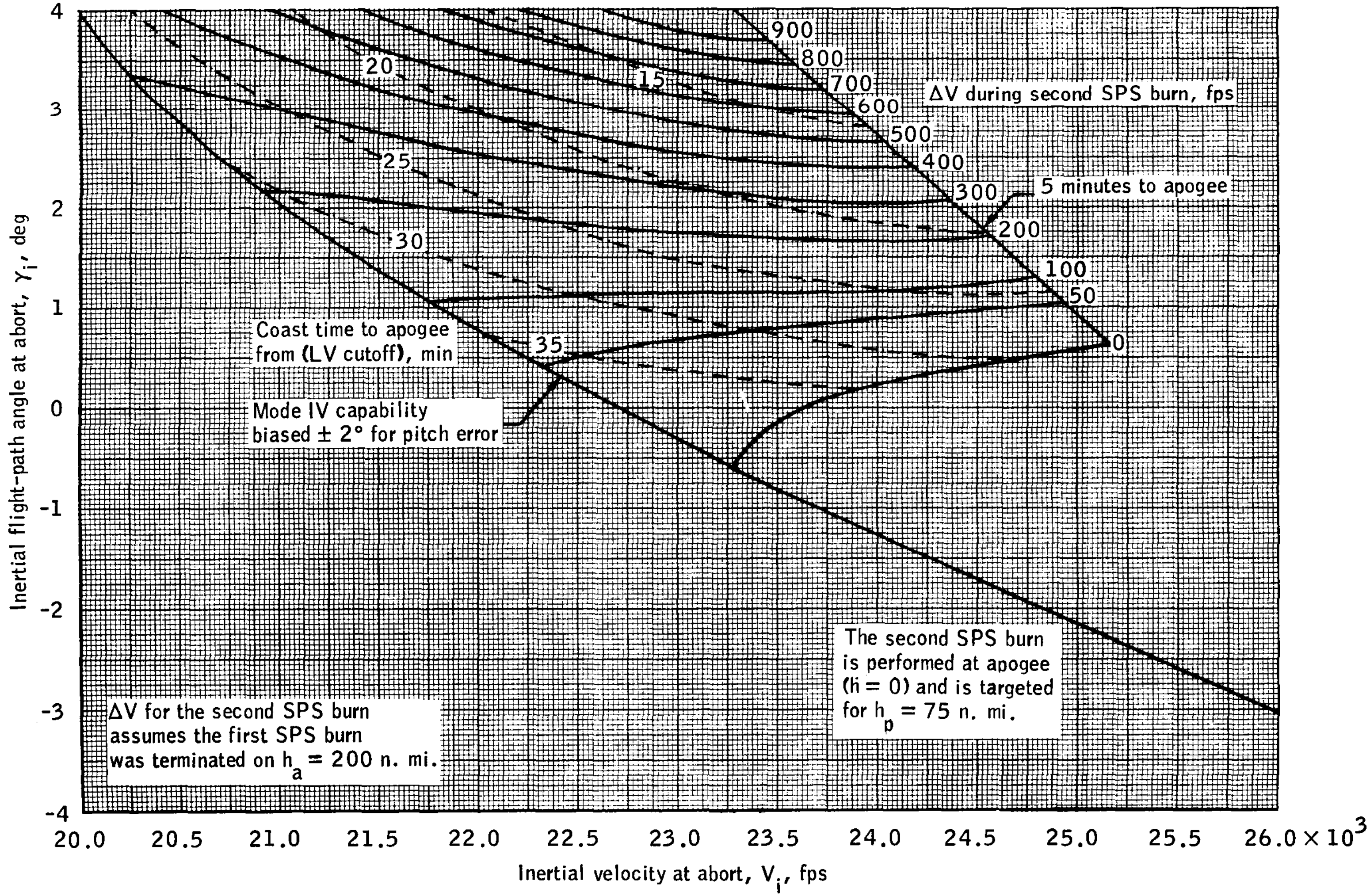


Figure 5-26. - Constant mode IV ΔV contours and ignition times for second impulse.

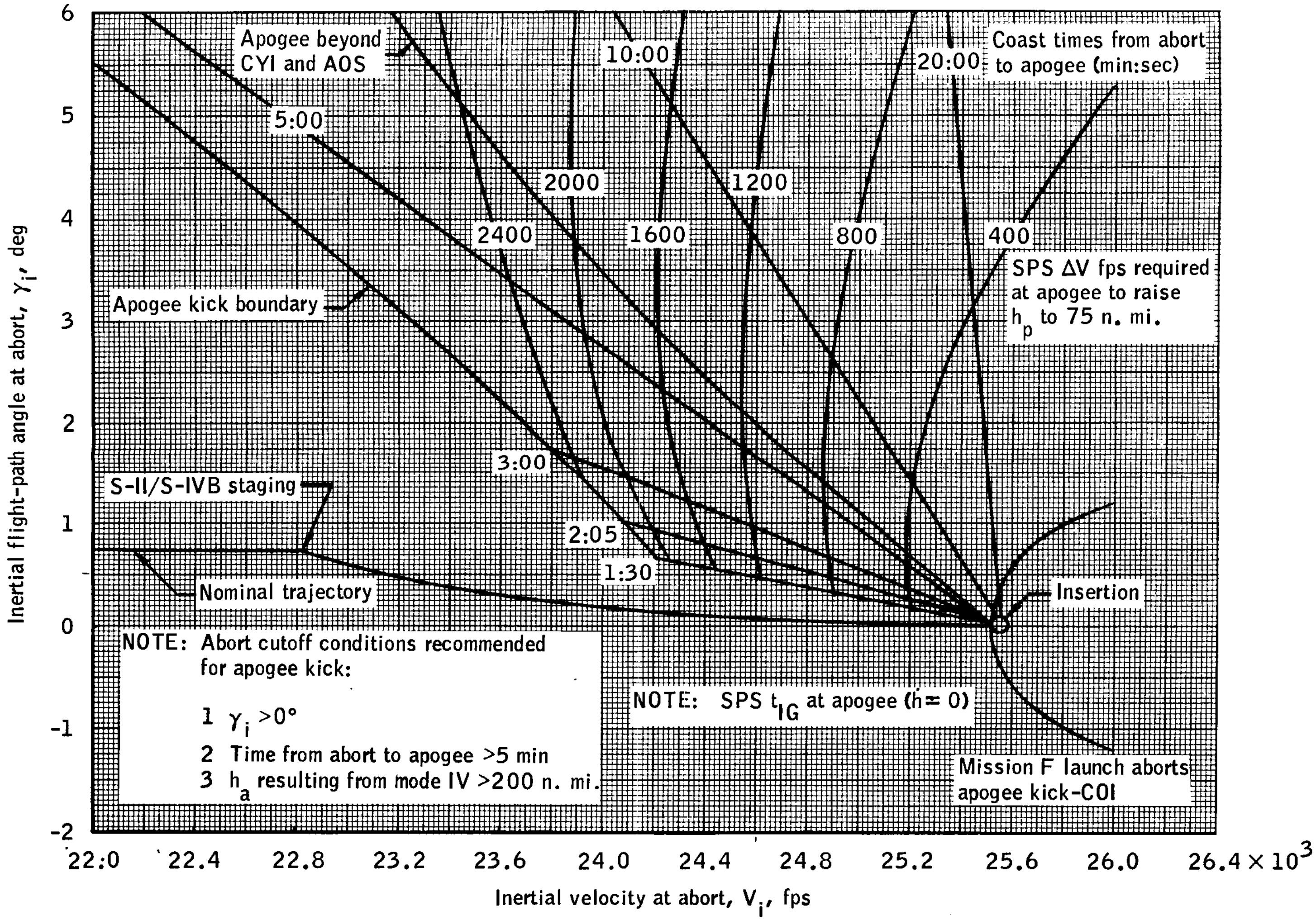


Figure 5-27.- Constant apogee kick ΔV and time to apogee contours for the apogee kick region.

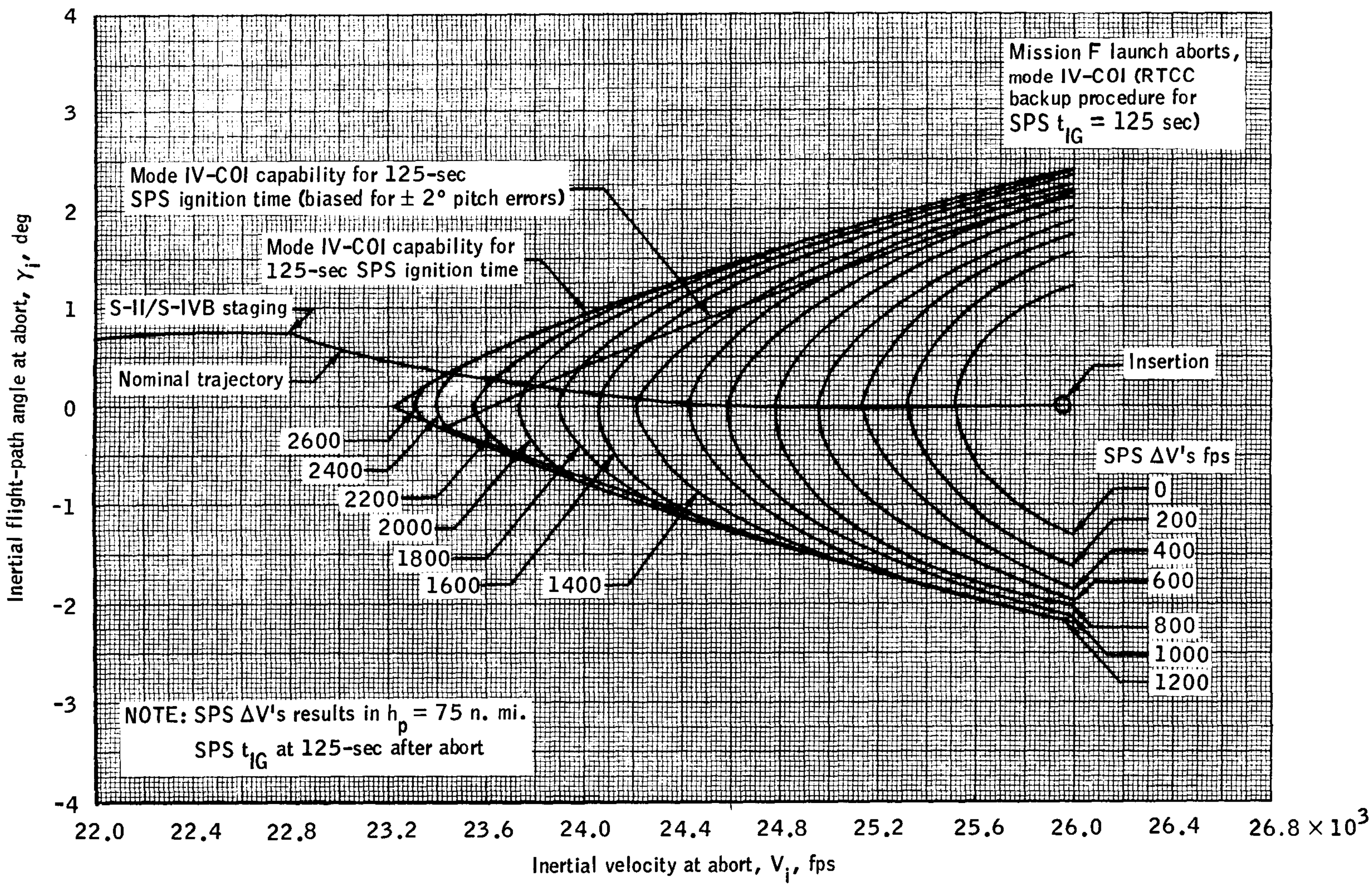


Figure 5-28.- Constant mode IV ΔV contours for the 125 second ignition time, backup.

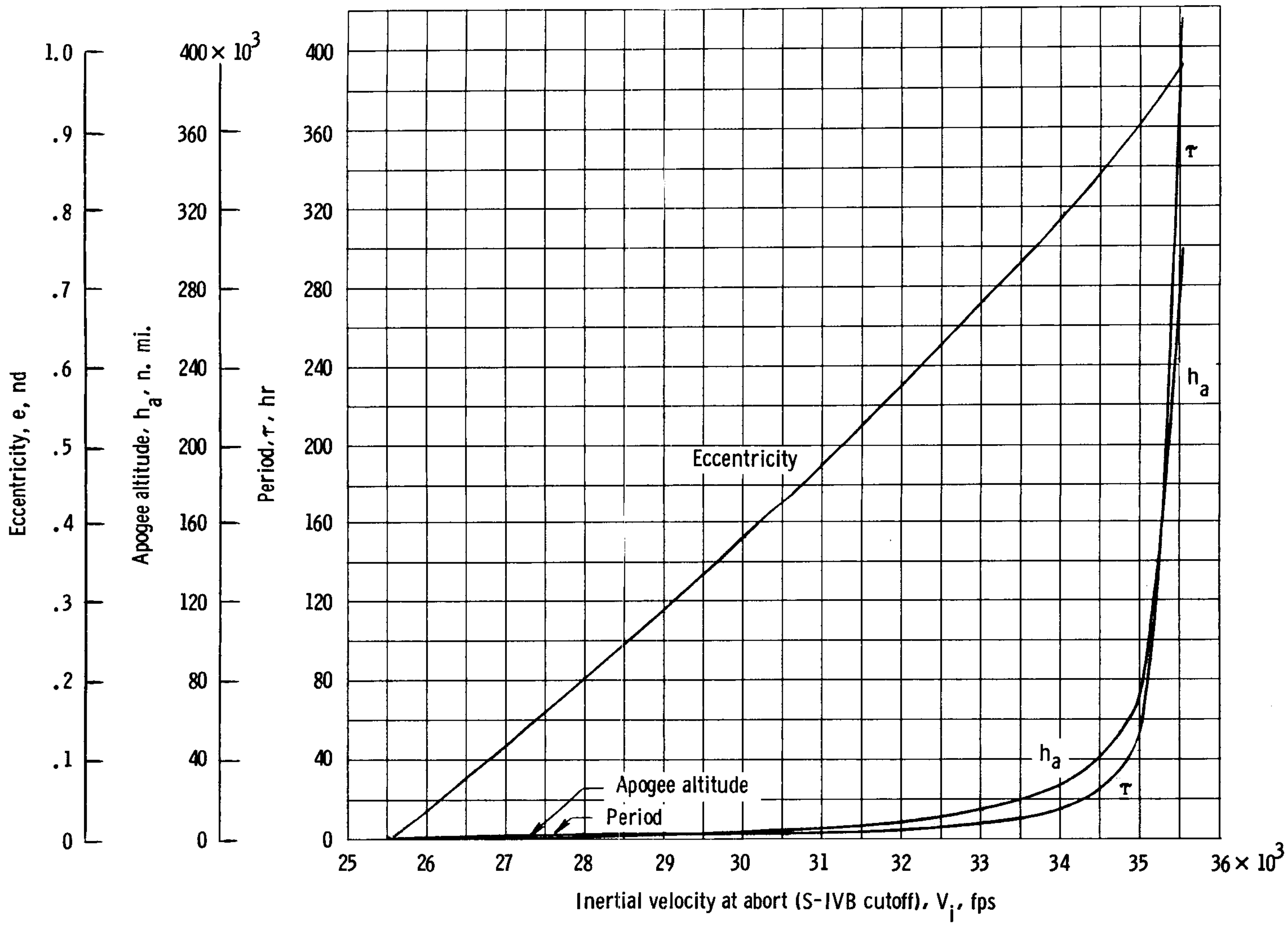


Figure 7-1. - Orbital parameters as functions of inertial velocity during the translunar injection burn.

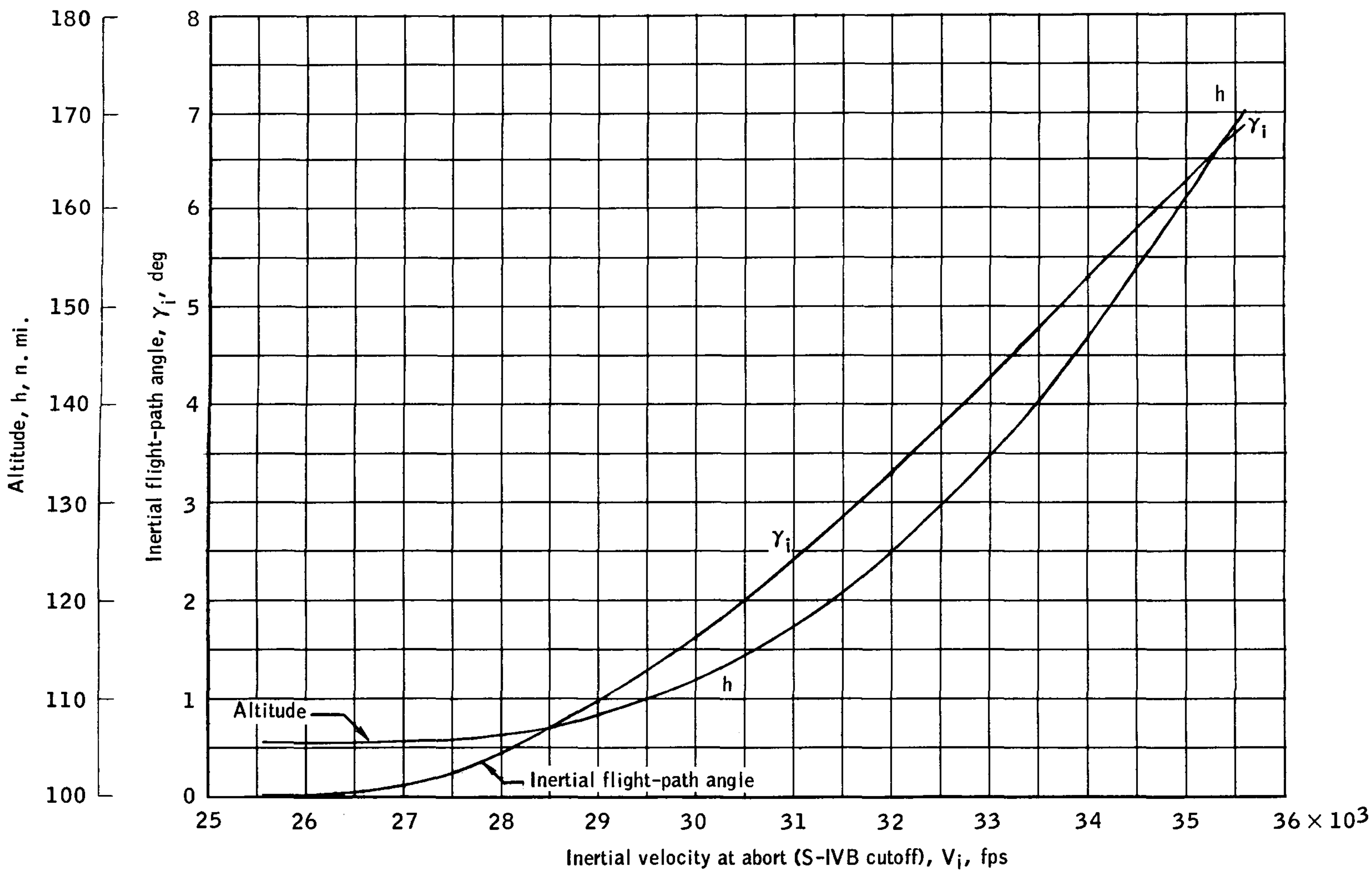


Figure 7-2.- Altitude and inertial flight-path angle as functions of inertial velocity during the translunar injection burn.

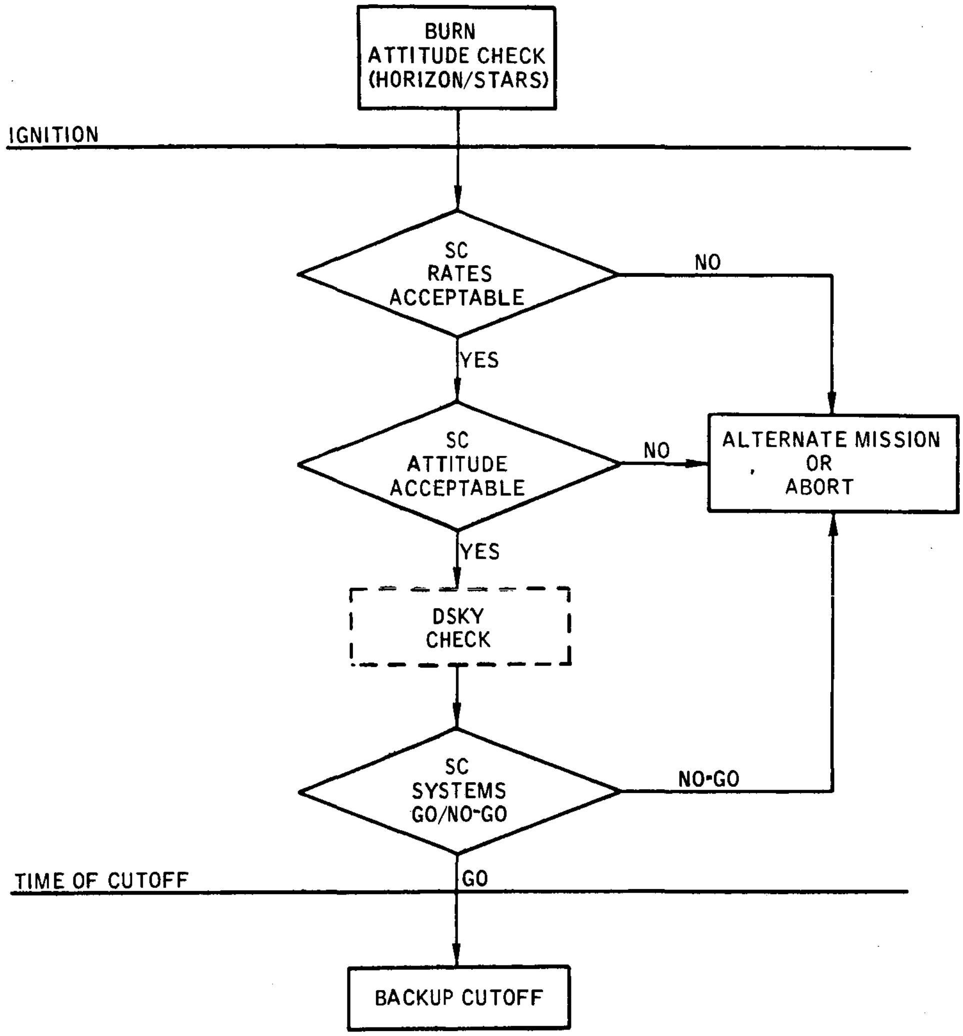


Figure 7-3.- Basic crew maneuver monitoring technique.

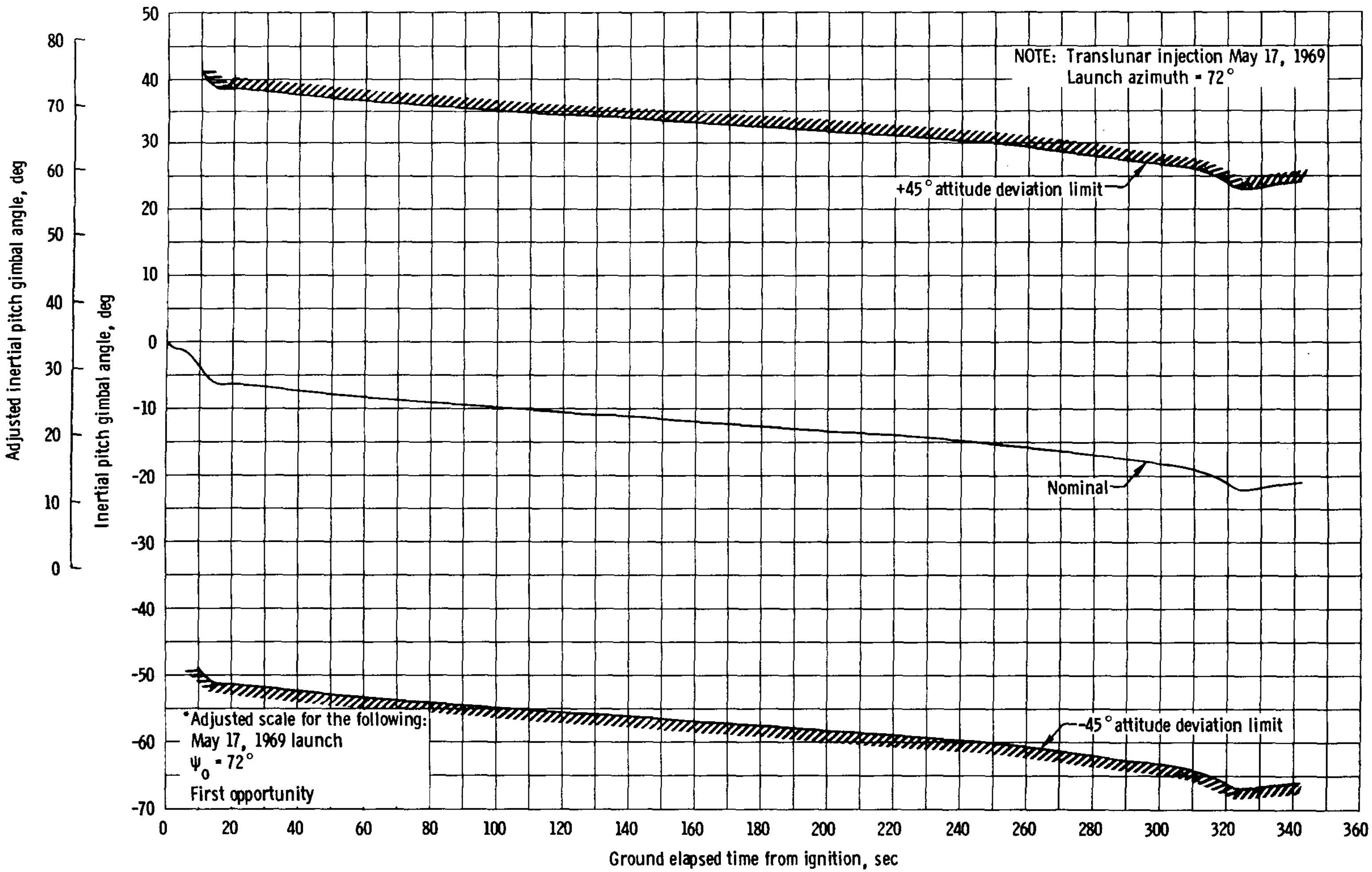


Figure 7-4. - TLI pitch gimbal angle history and attitude deviation limits for first opportunity.

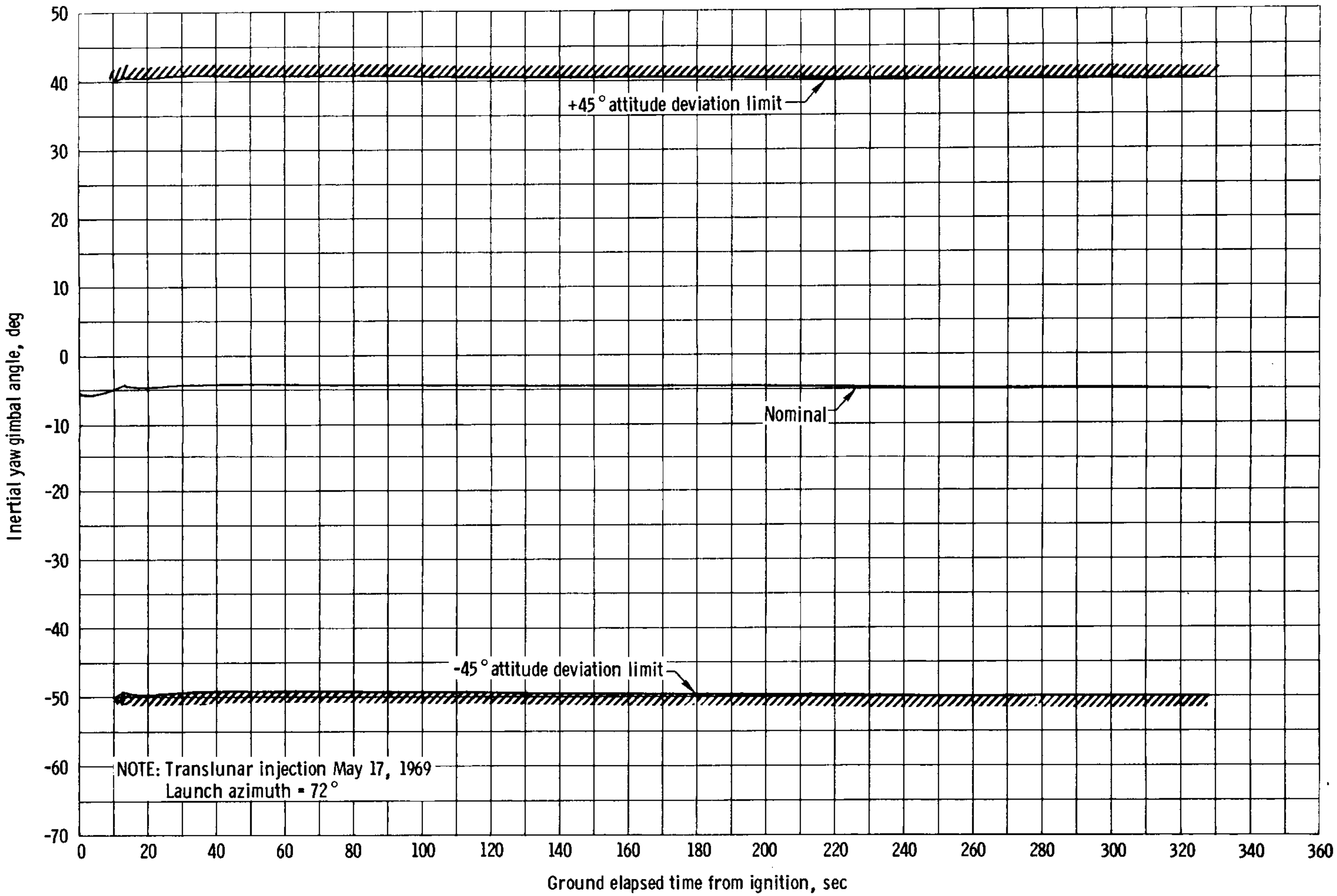


Figure 7-5. - TLI yaw gimbal angle history and attitude deviation limits for first opportunity.

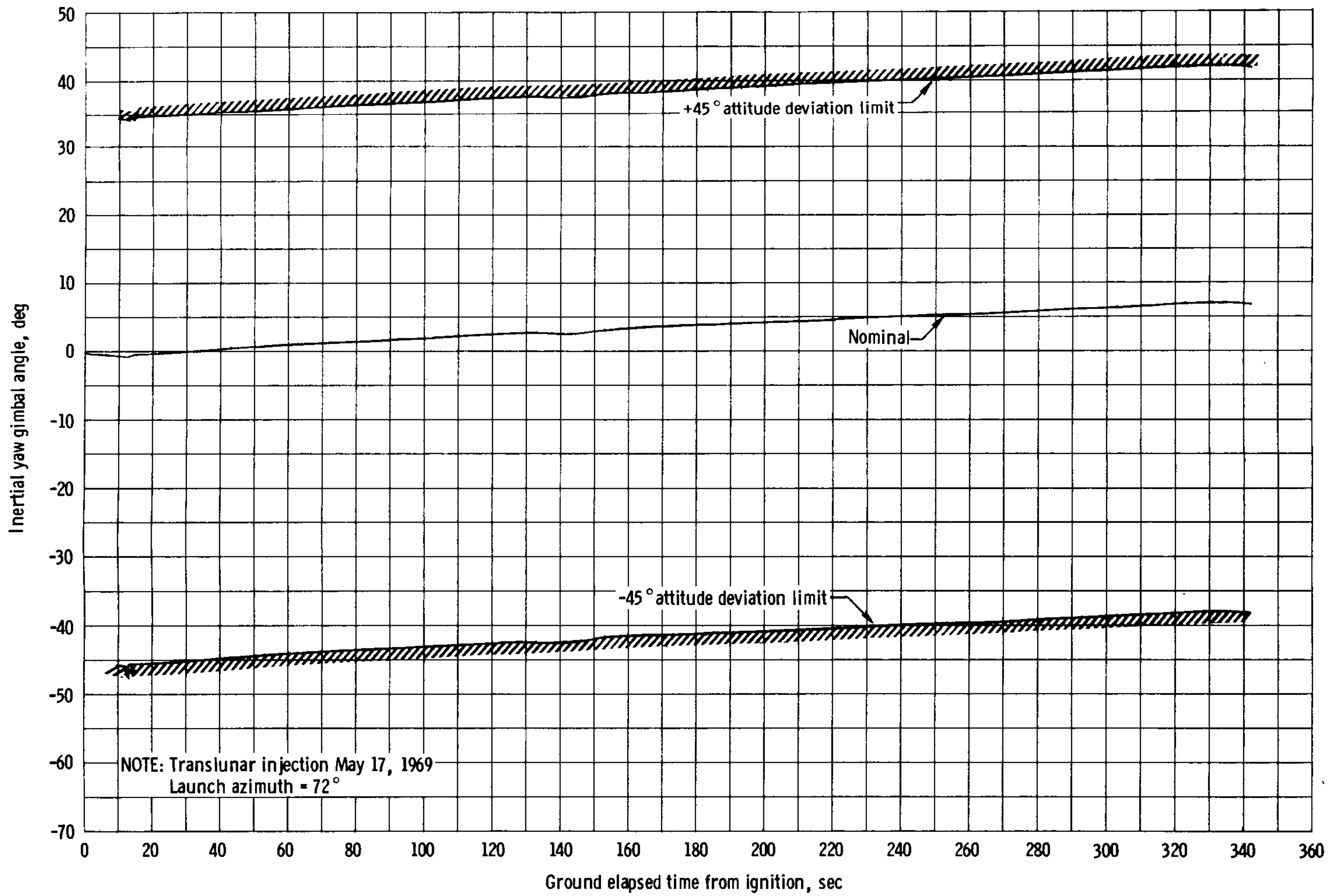


Figure 7-6. - TLI yaw gimbal angle history and attitude deviation limits for second opportunity.

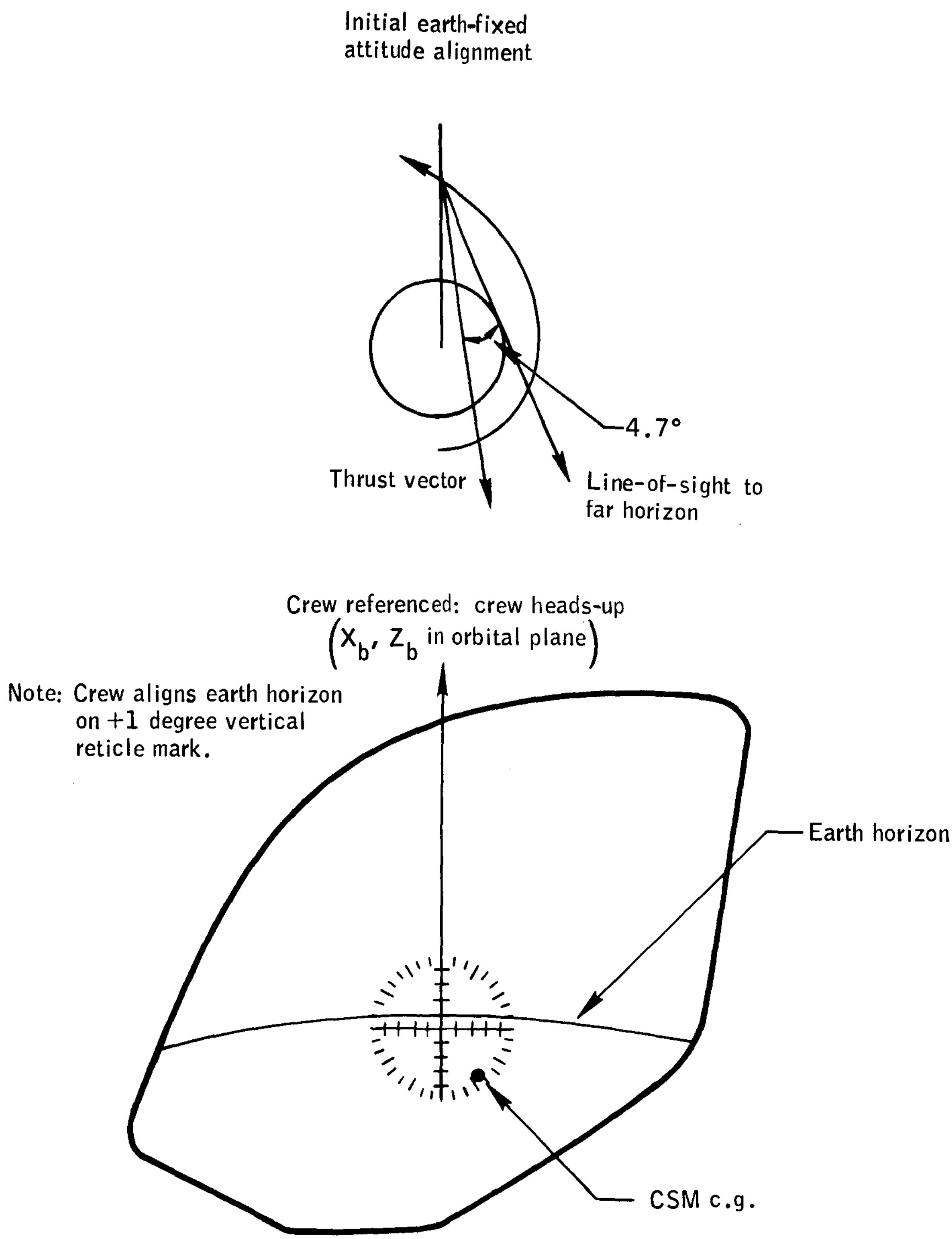


Figure 7-7.- Definition of attitude for fixed-attitude aborts from TLI.

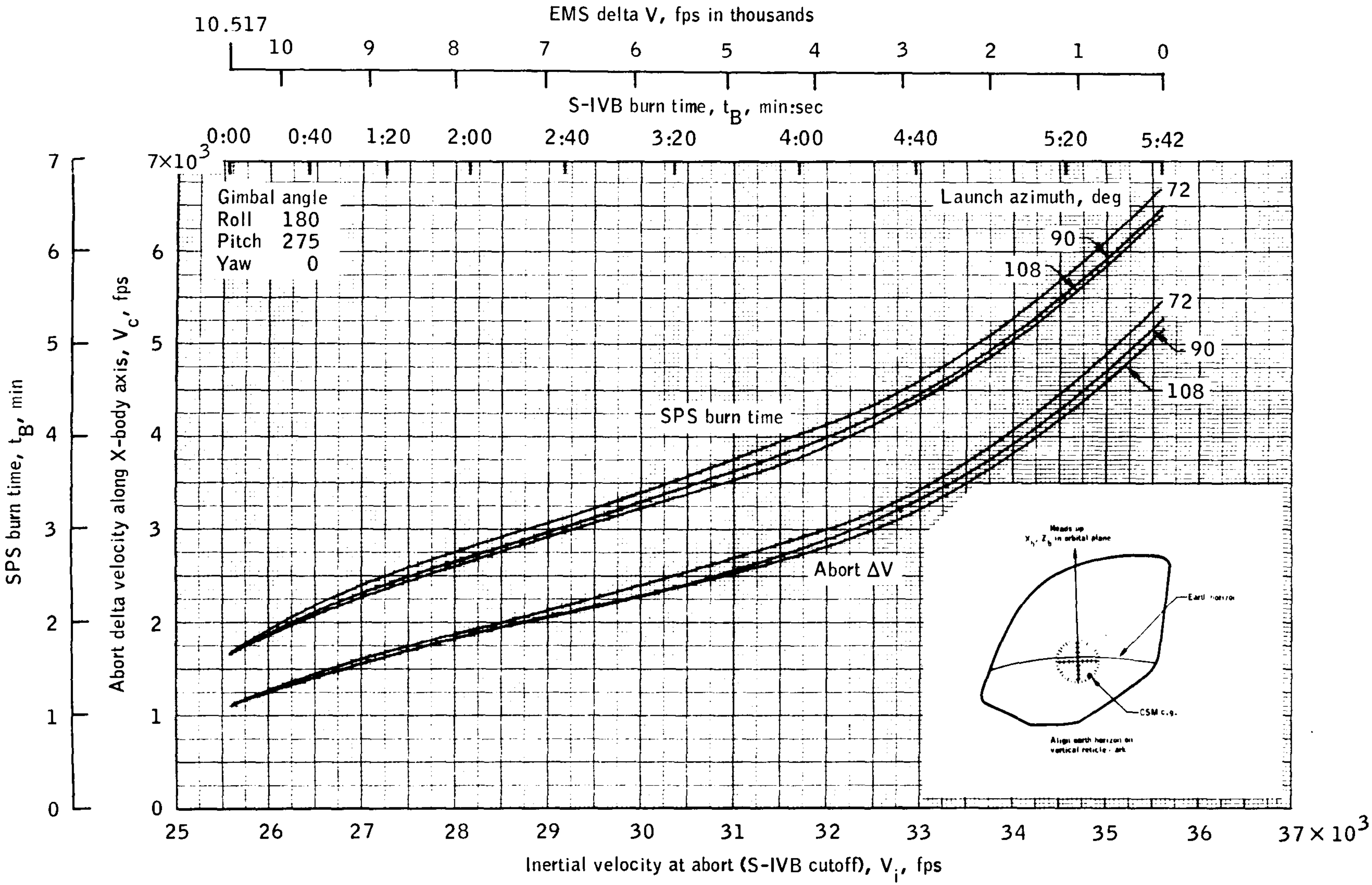


Figure 7-8. - Abort delta velocity and SPS burn time as functions of inertial velocity at abort for fixed-attitude aborts from TLI.

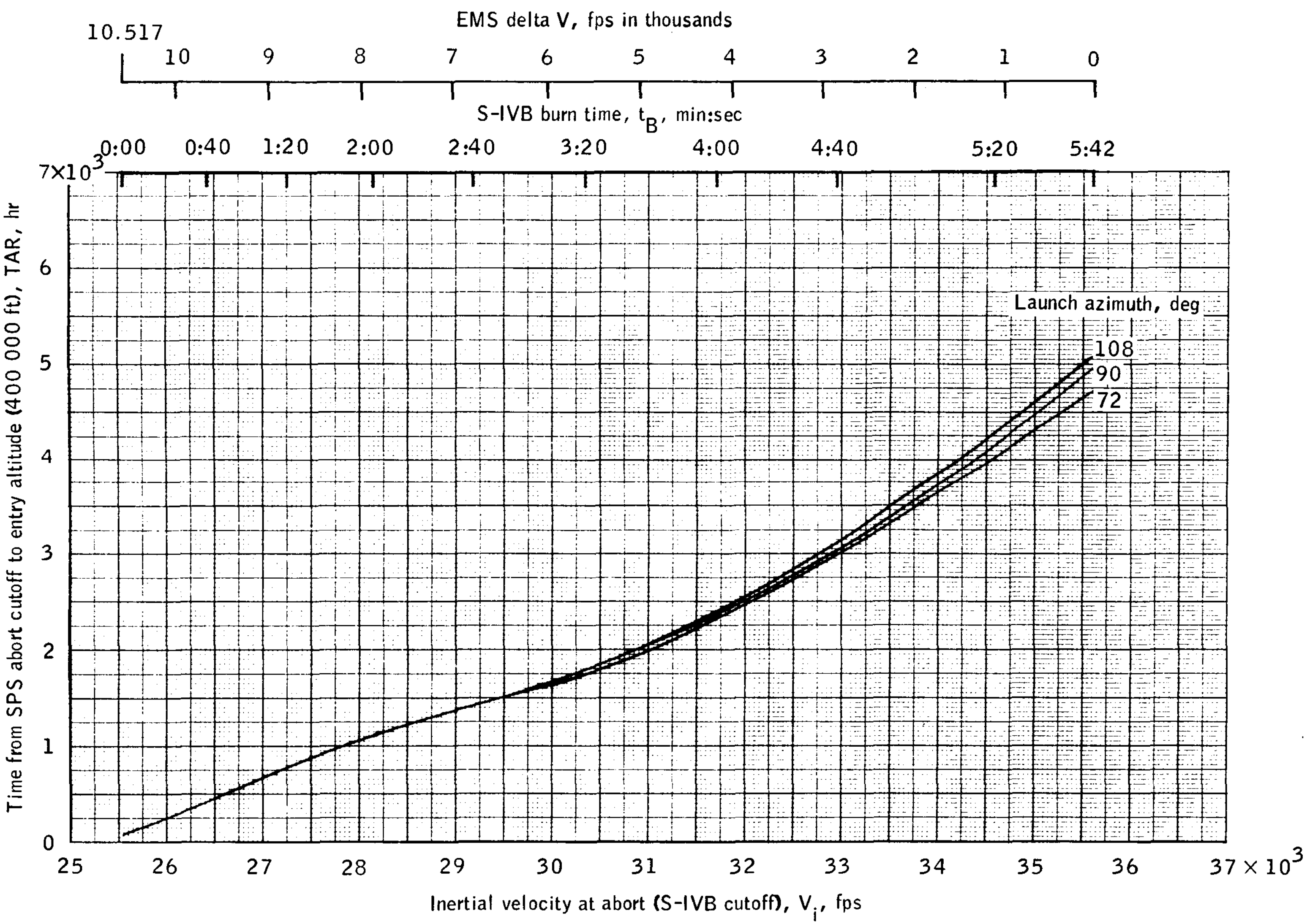


Figure 7-9.- Time from SPS cutoff to 400 000 feet as a function of inertial velocity at abort for fixed-attitude aborts from TLI.

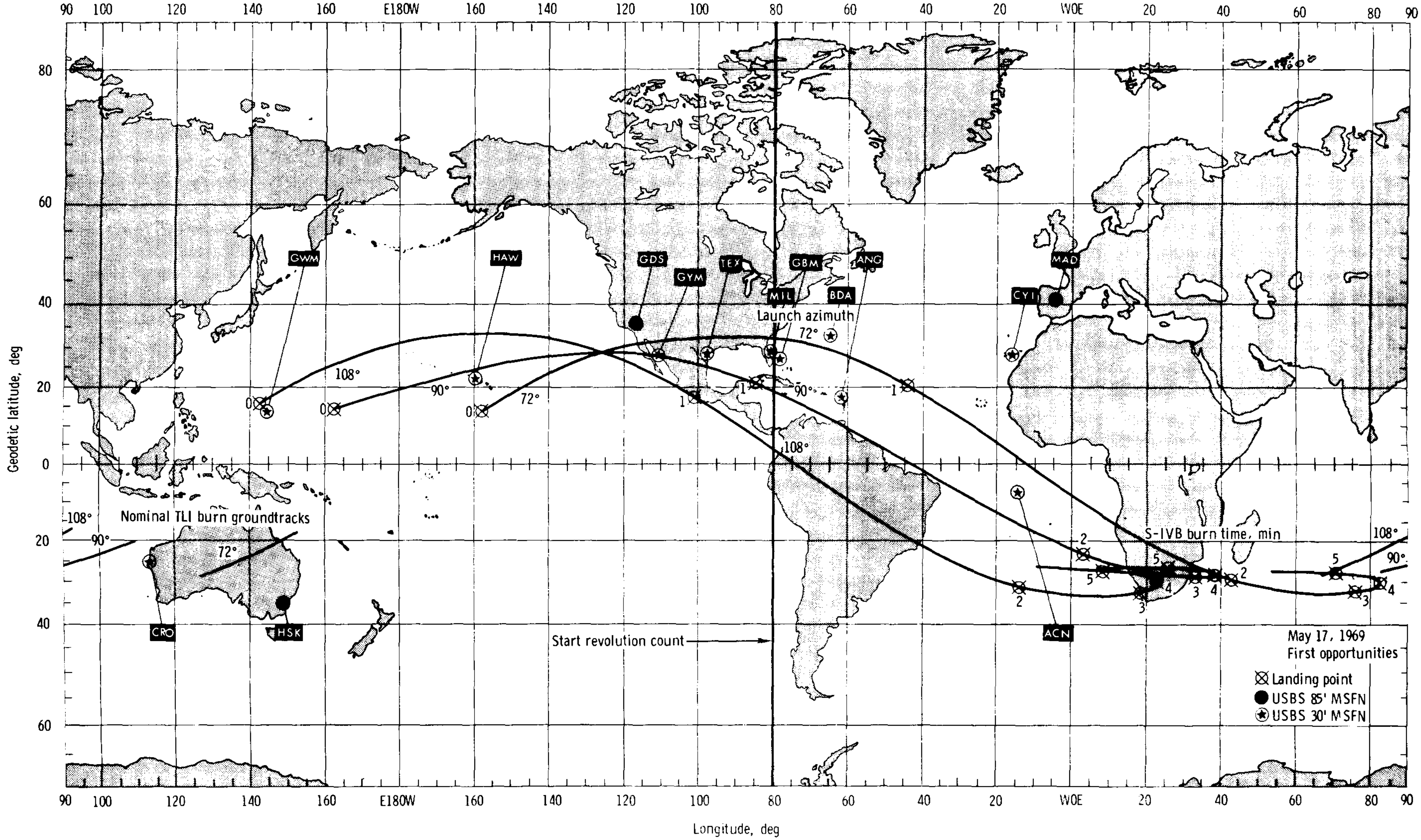
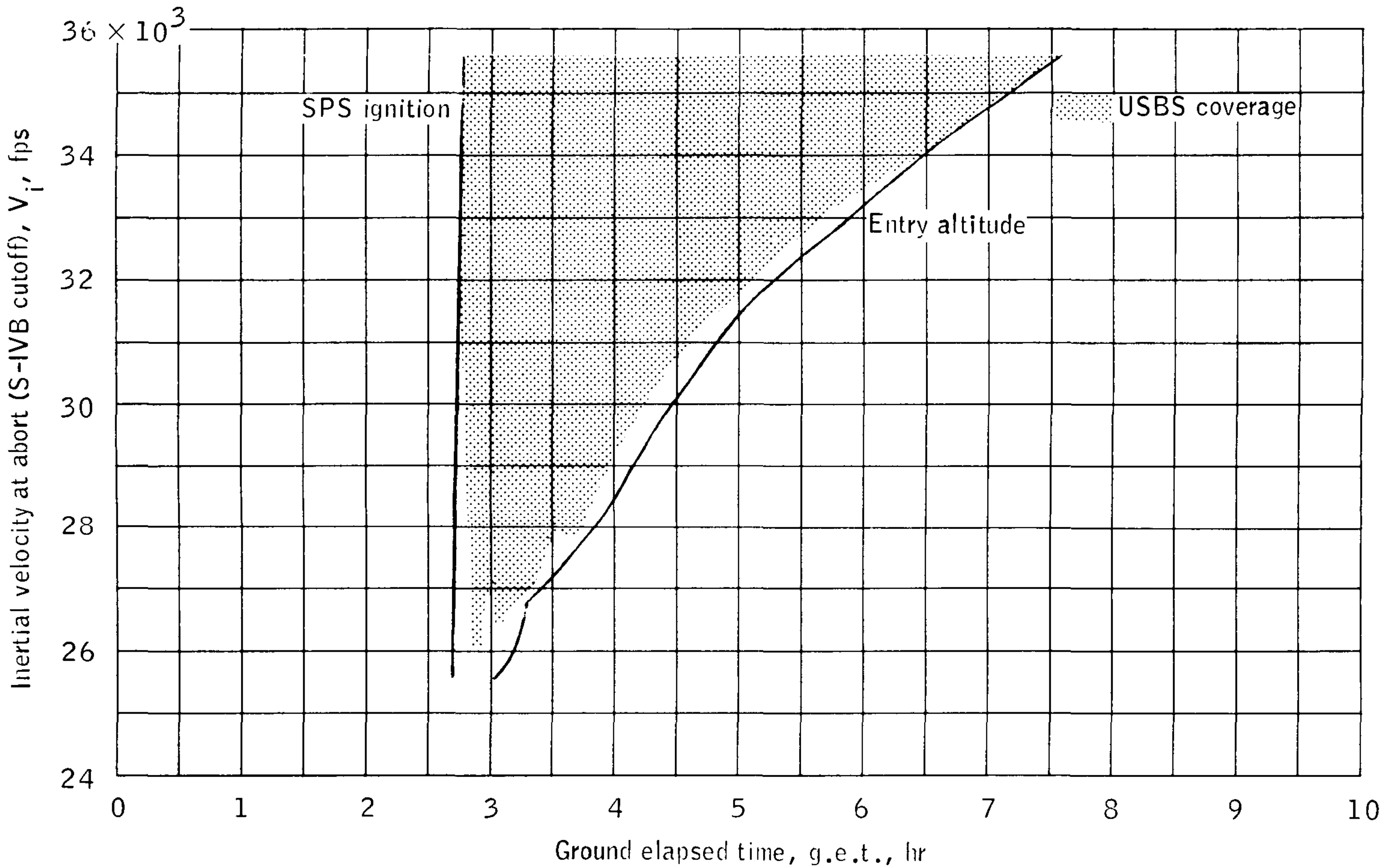
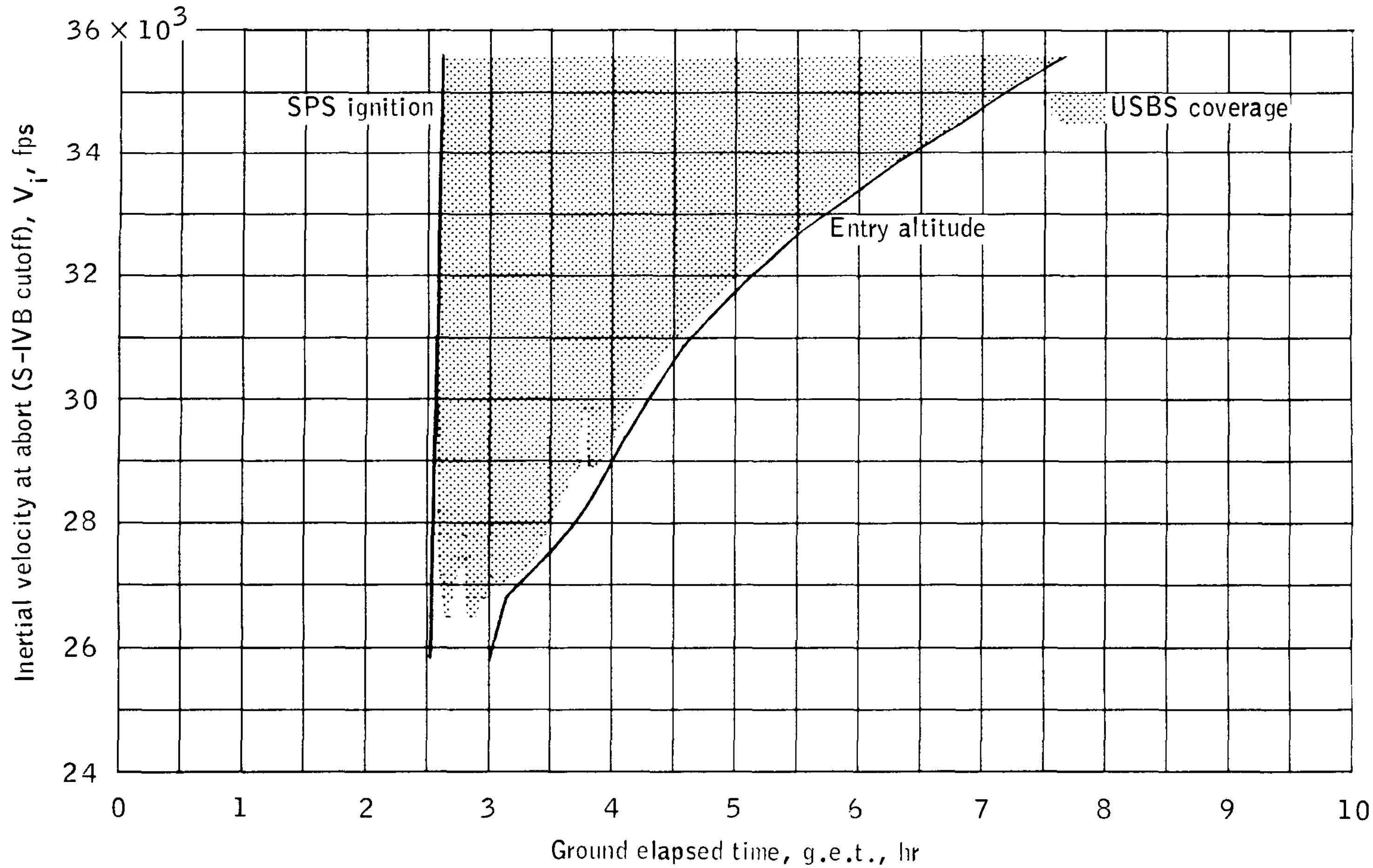


Figure 7-10. - Nominal TLI burn groundtracks and fixed-attitude abort landing point loci.



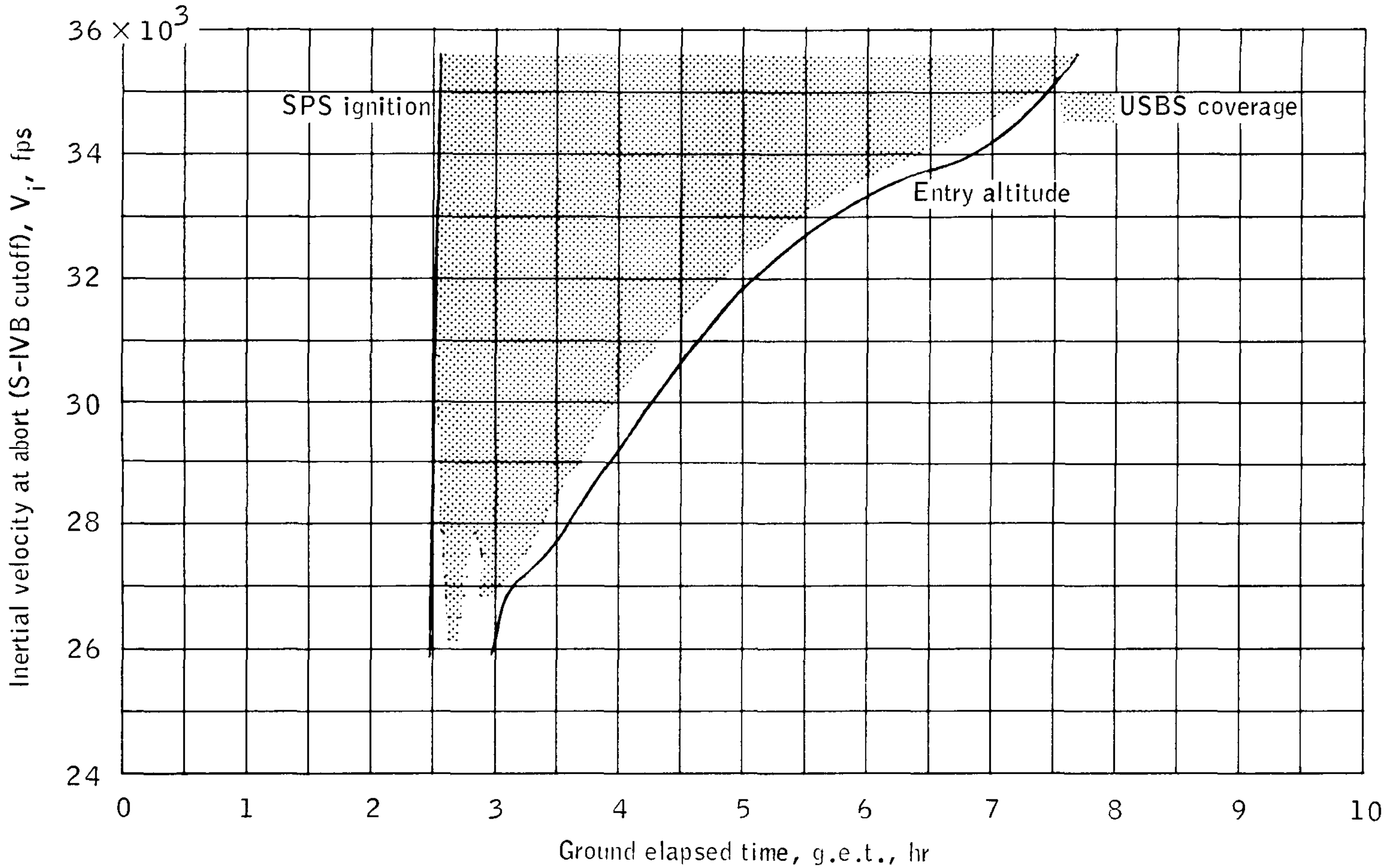
(a) Launch azimuth = 72°, first opportunity.

Figure 7-11.- Ground elapsed time of continuous USBS coverage for fixed-attitude aborts from TLI as a function of inertial velocity at abort.



(b) Launch azimuth = 90°, first opportunity.

Figure 7-11.- Continued.



(c) Launch azimuth = 108°, first opportunity.

Figure 7-11.- Concluded.

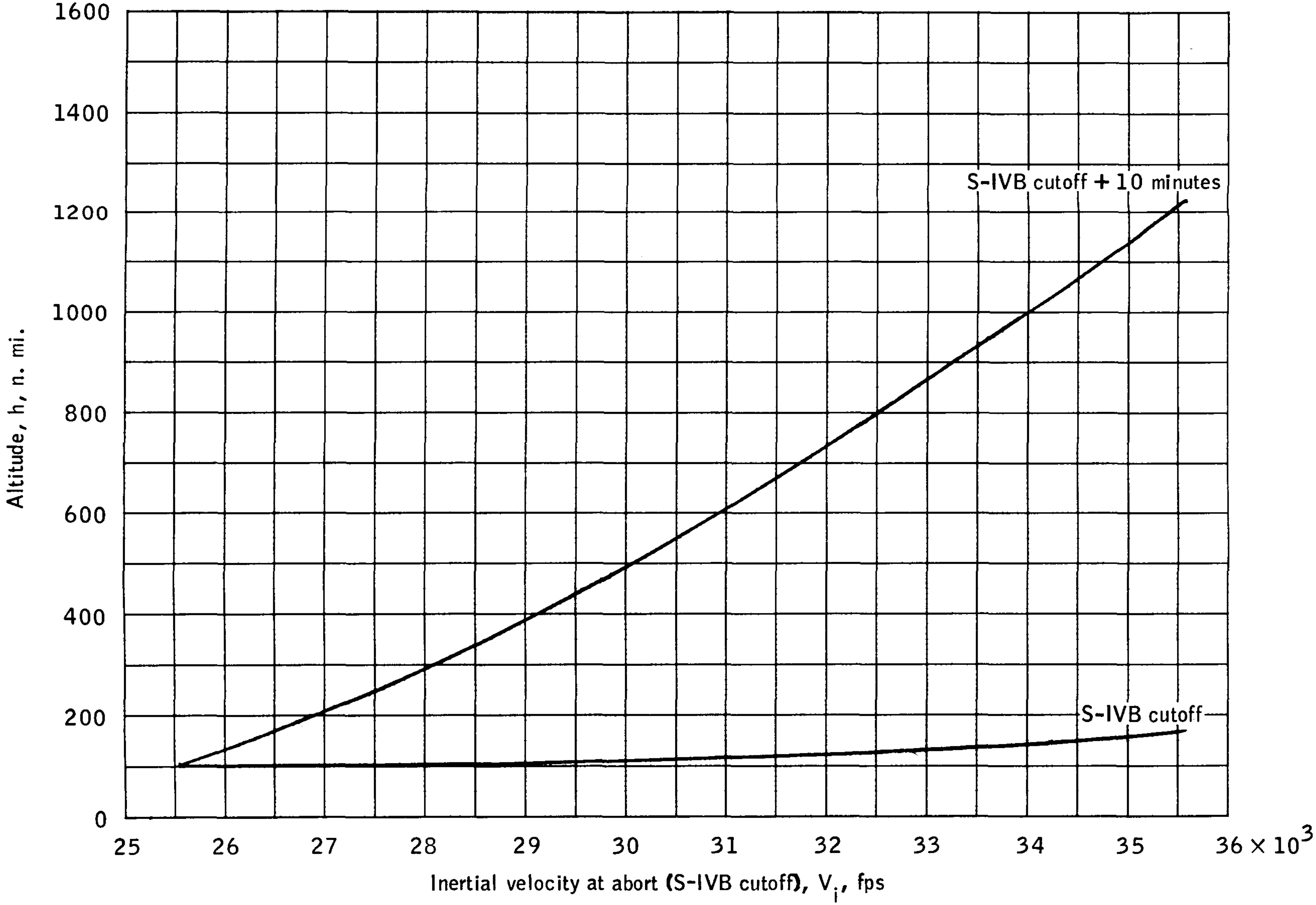


Figure 7-12.- Altitude at S-IVB cutoff and altitude S-IVB cutoff-plus-10-minutes as functions of inertial velocity at abort for fixed-attitude aborts from TLI.

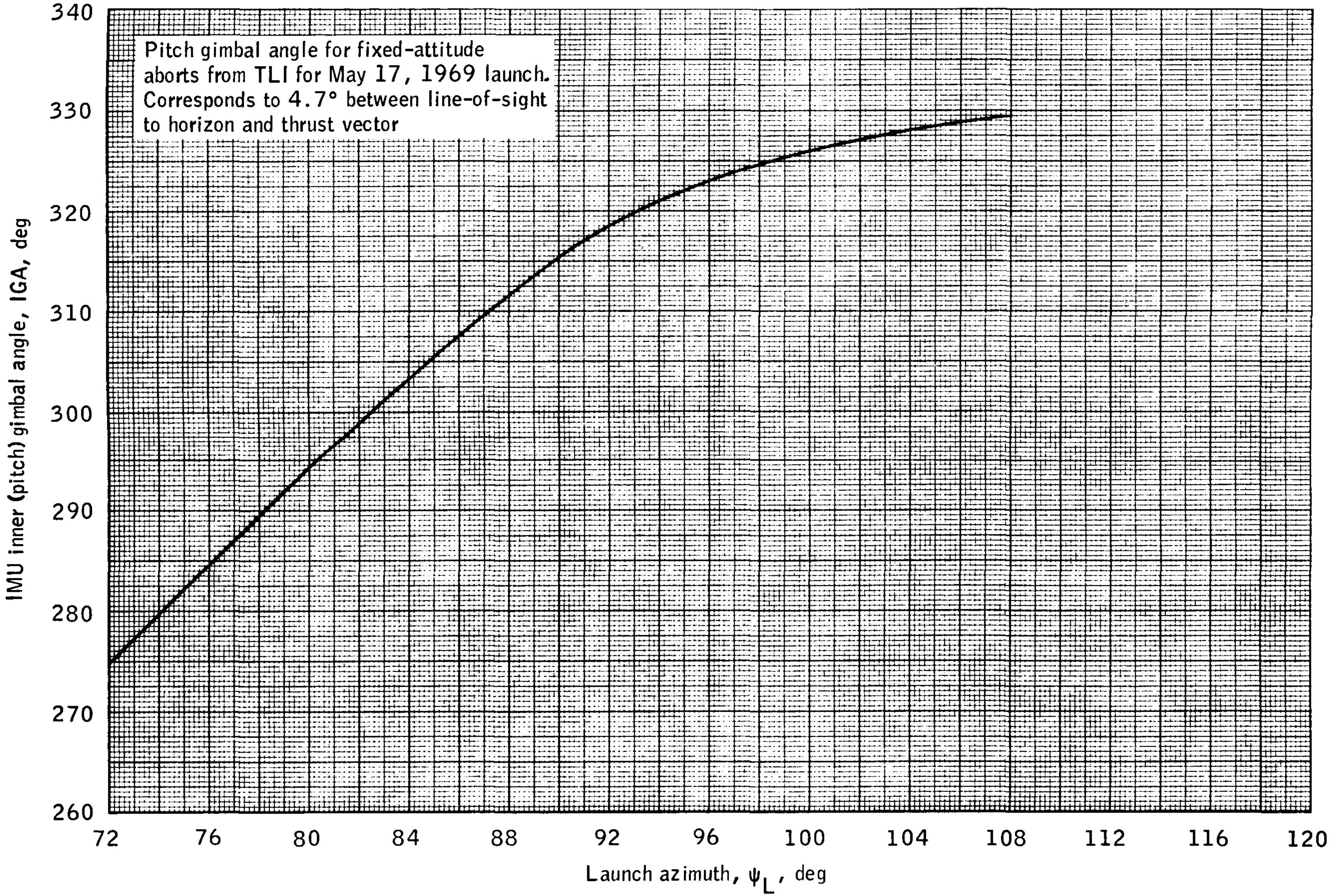


Figure 7-13.- Pitch gimbal angle at S-IVB cutoff-plus-10-minutes as a function of launch azimuth.

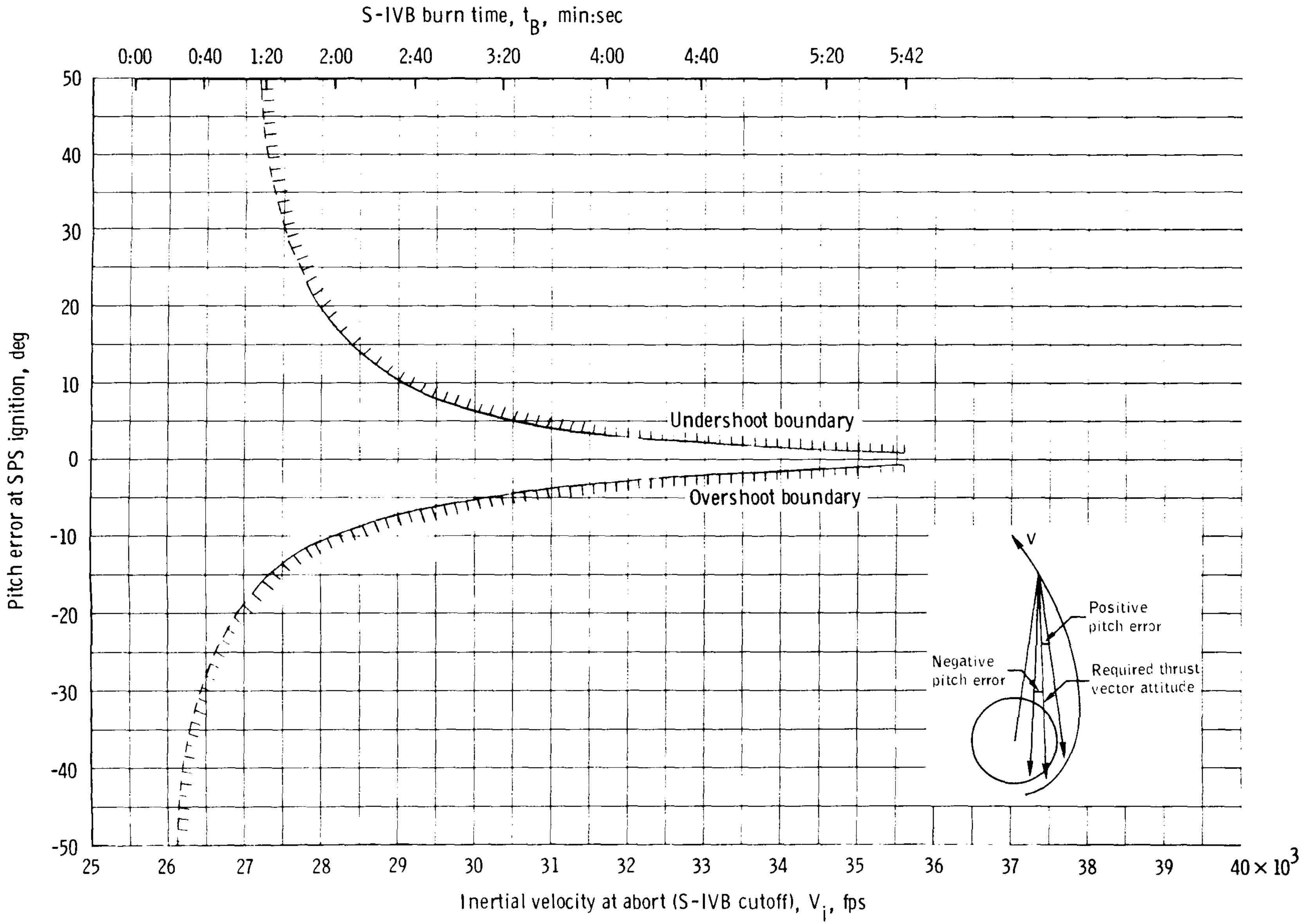


Figure 7-14. - Tolerable pitch errors as a function of inertial velocity at abort for fixed-attitude aborts from TLI.

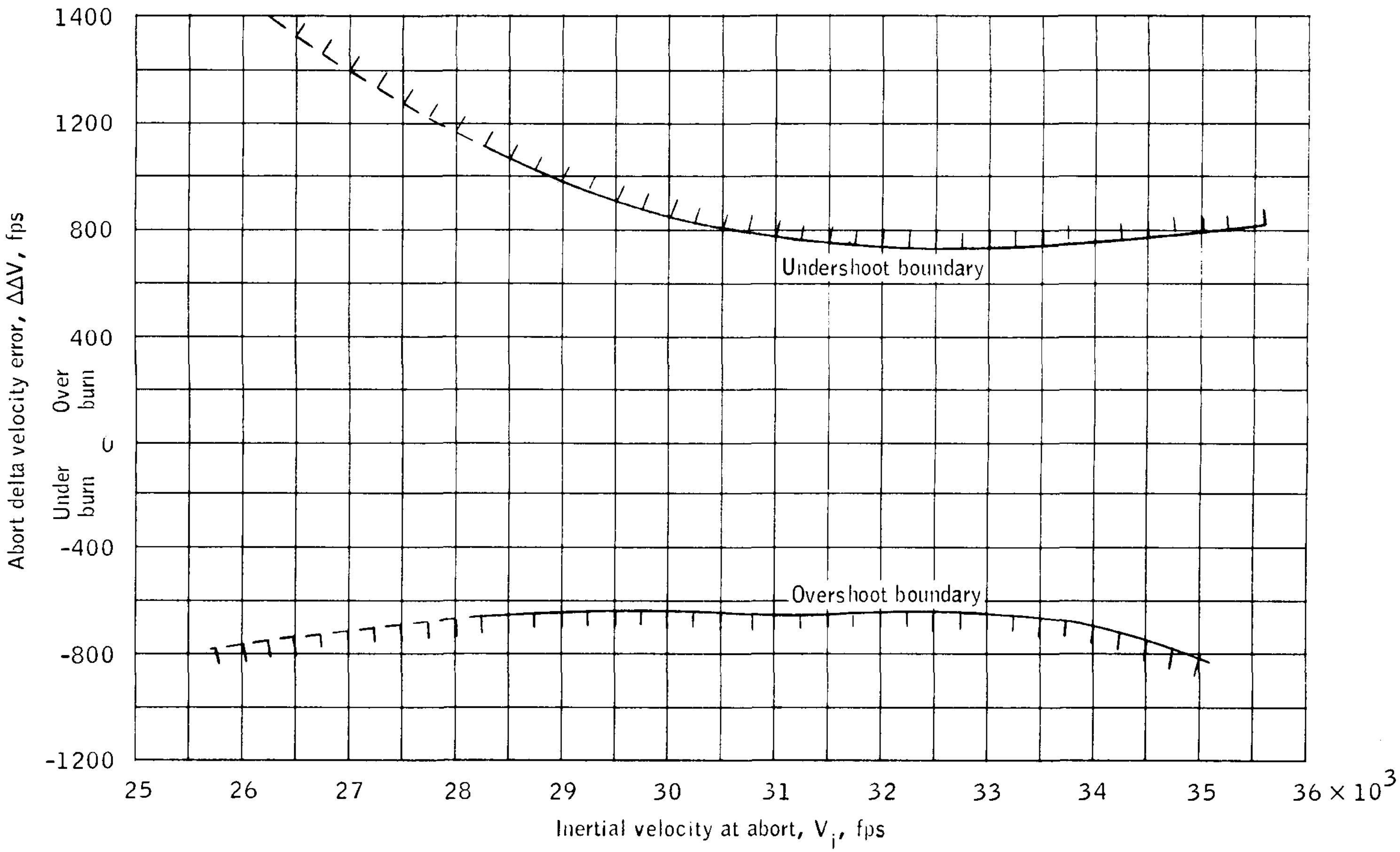


Figure 7-15.- Abort delta velocity error required to achieve overshoot and undershoot entry boundaries for fixed-attitude aborts from TLI.

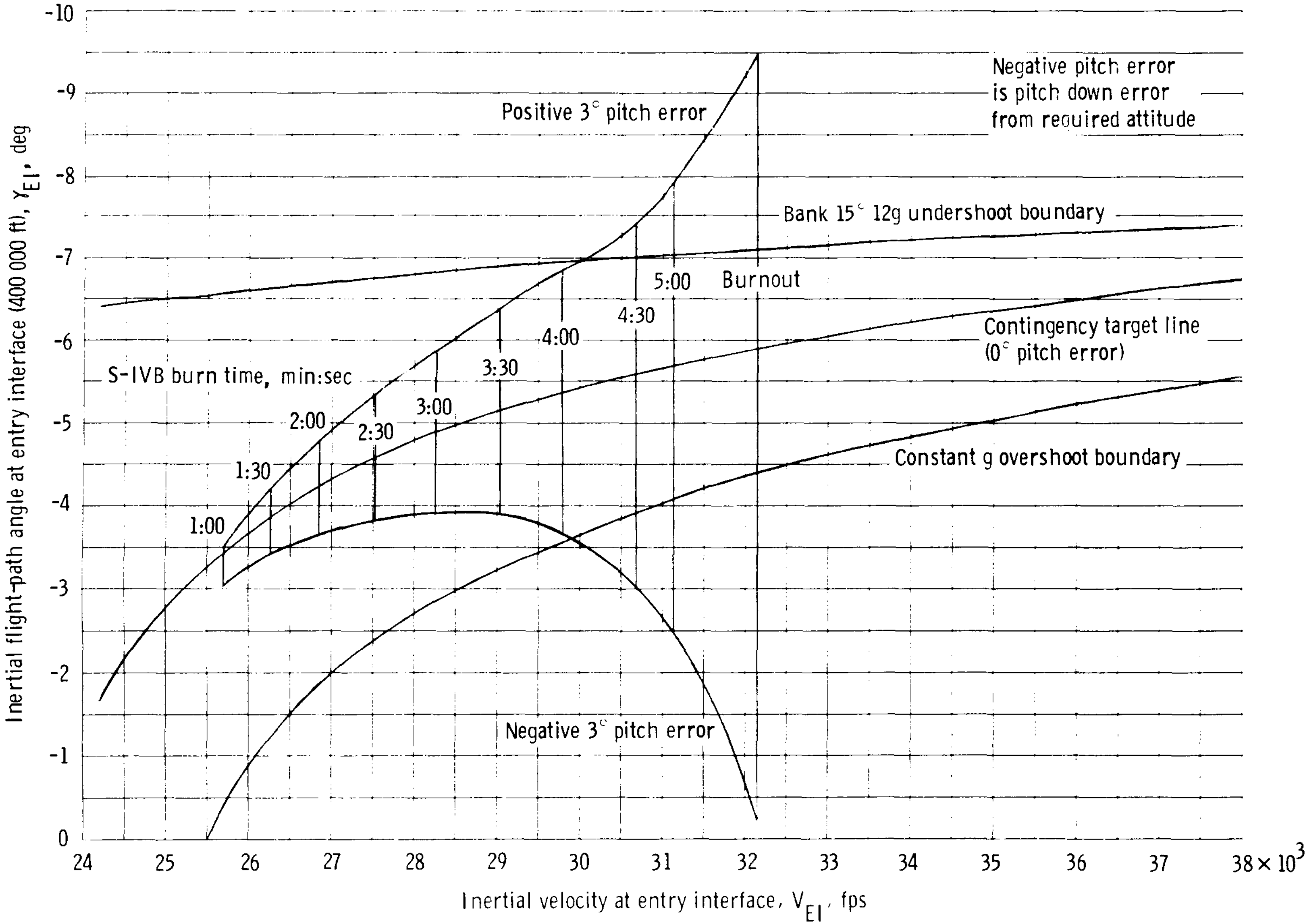


Figure 7-16. - Effect of positive and negative 3° pitch errors on entry vector for fixed-attitude aborts from TLI.

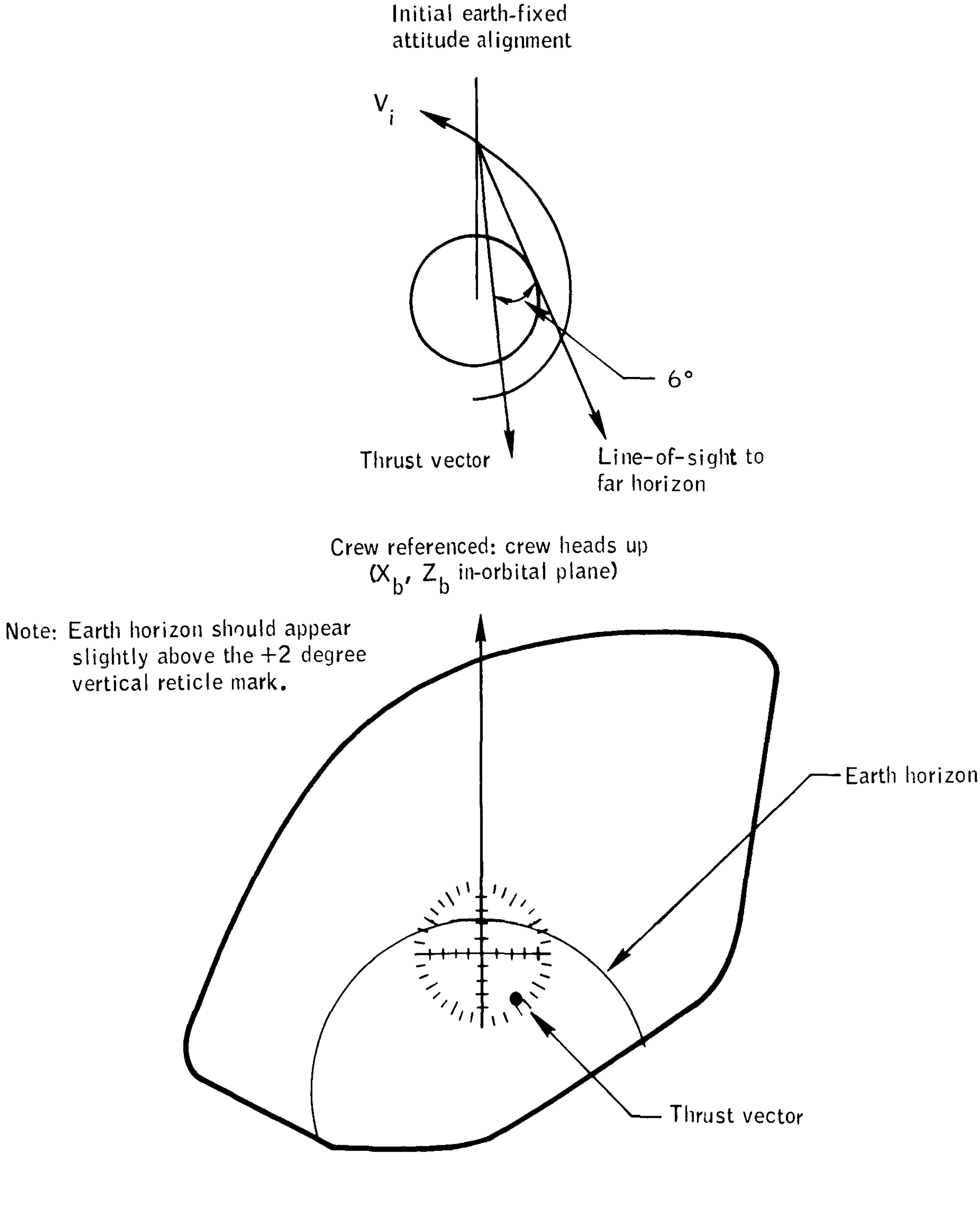


Figure 7-17.- Definition of attitude for TLI-plus-90-minute aborts.

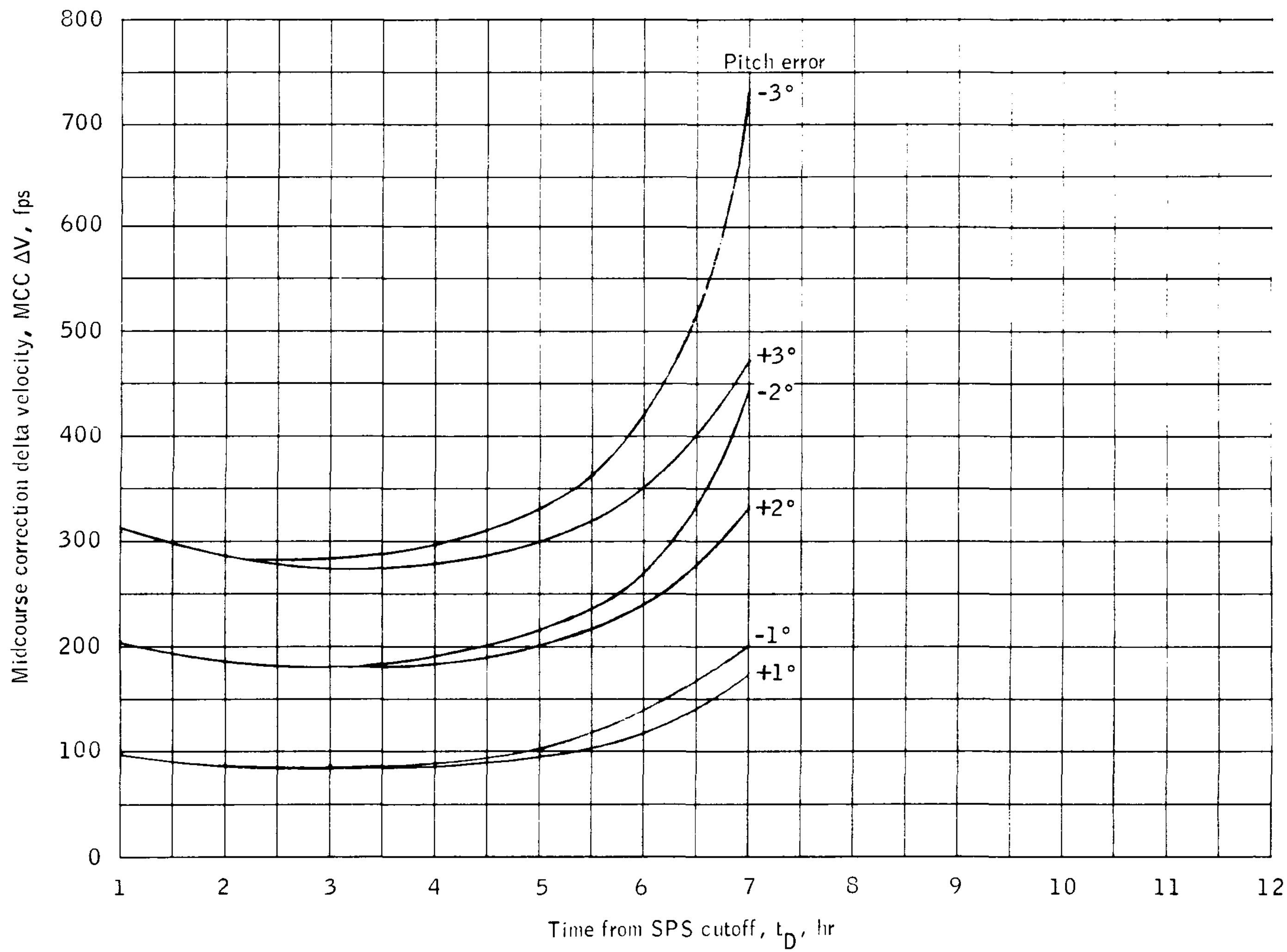


Figure 7-18.- Midcourse correction delta velocities for various pitch pointing errors required to achieve the contingency target line as a function of time from SPS cutoff.

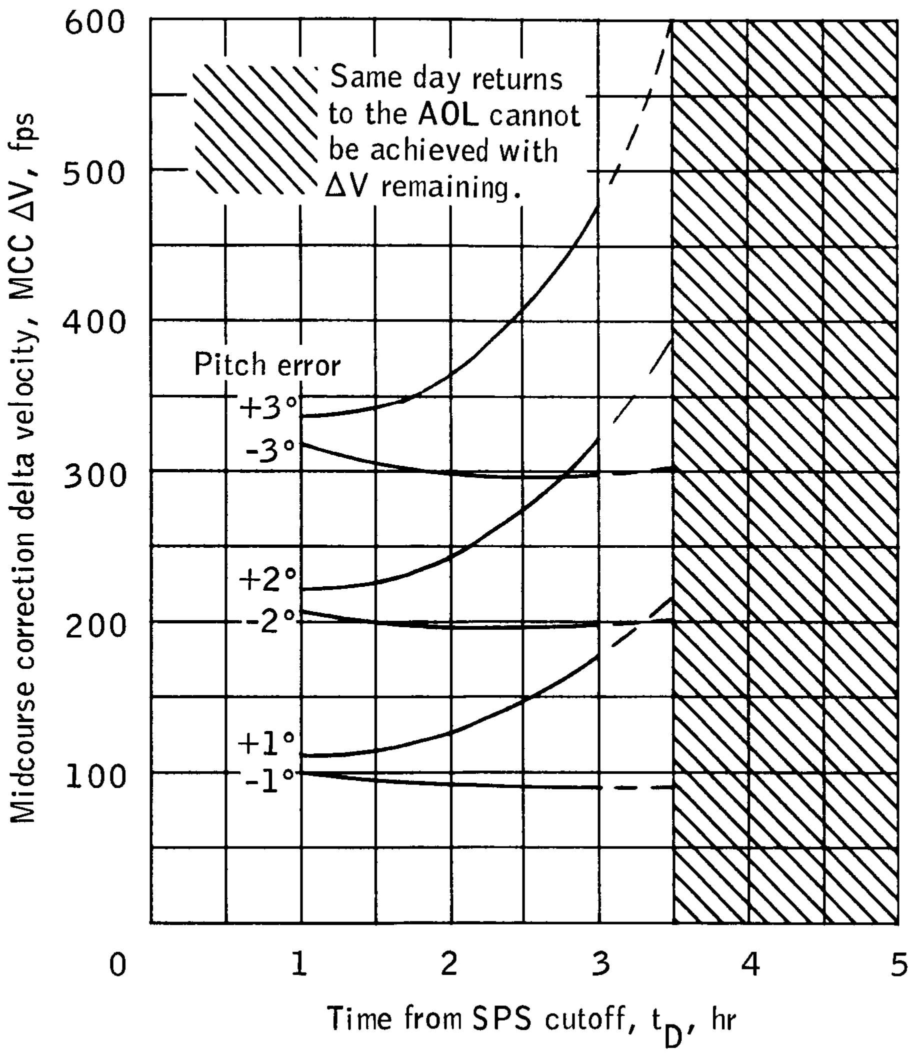


Figure 7-19.- Midcourse correction delta velocities for various pitch pointing errors required to achieve the contingency target line and the Atlantic Ocean Line (AOL) as a function of time from SPS cutoff.

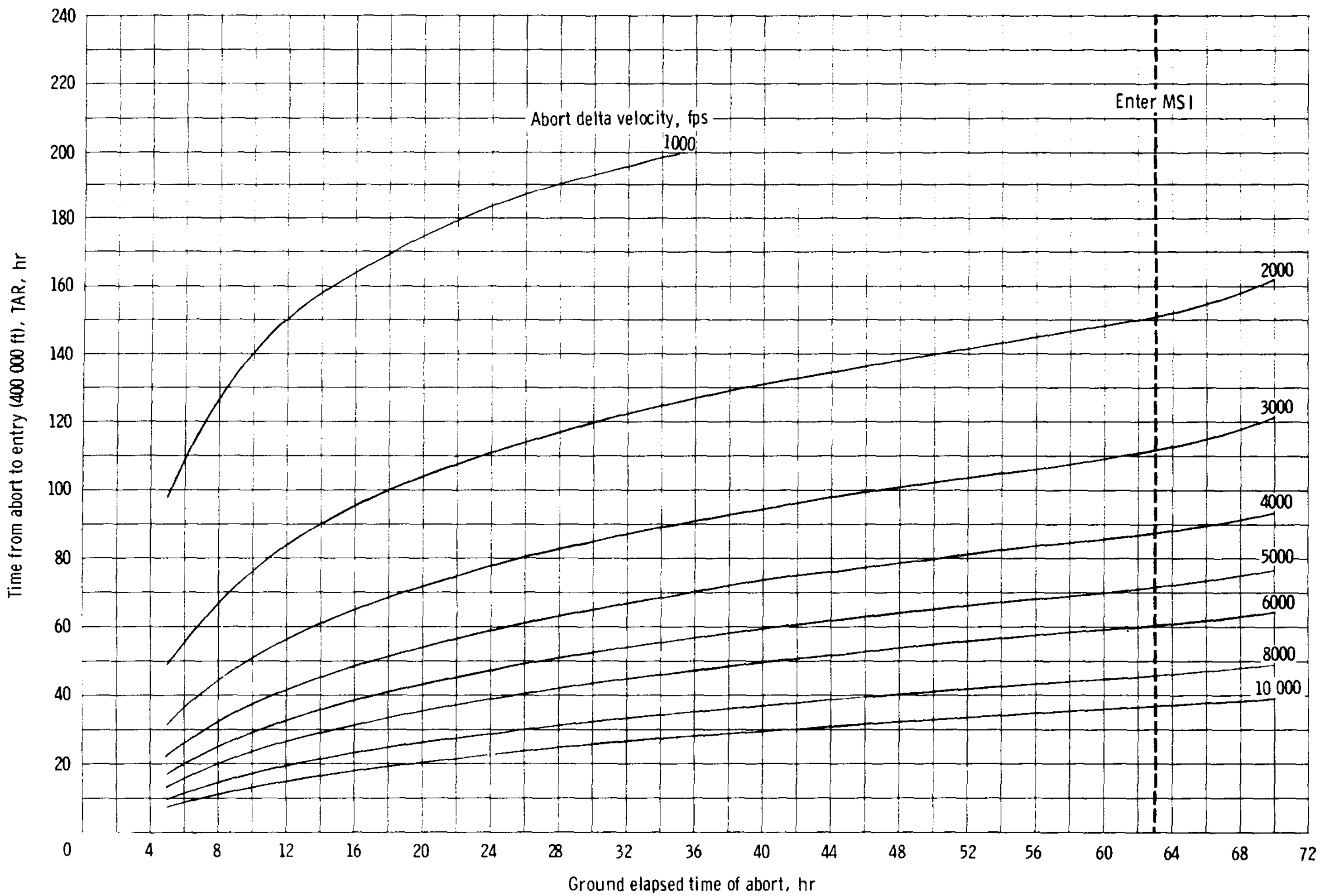


Figure 7-20. - Time from abort to entry as functions of abort ΔV and ground elapsed time of abort for unspecified area aborts from the nominal translunar coast. (May 17, 1969 launch. $\psi_L = 72°$, first opportunity.)

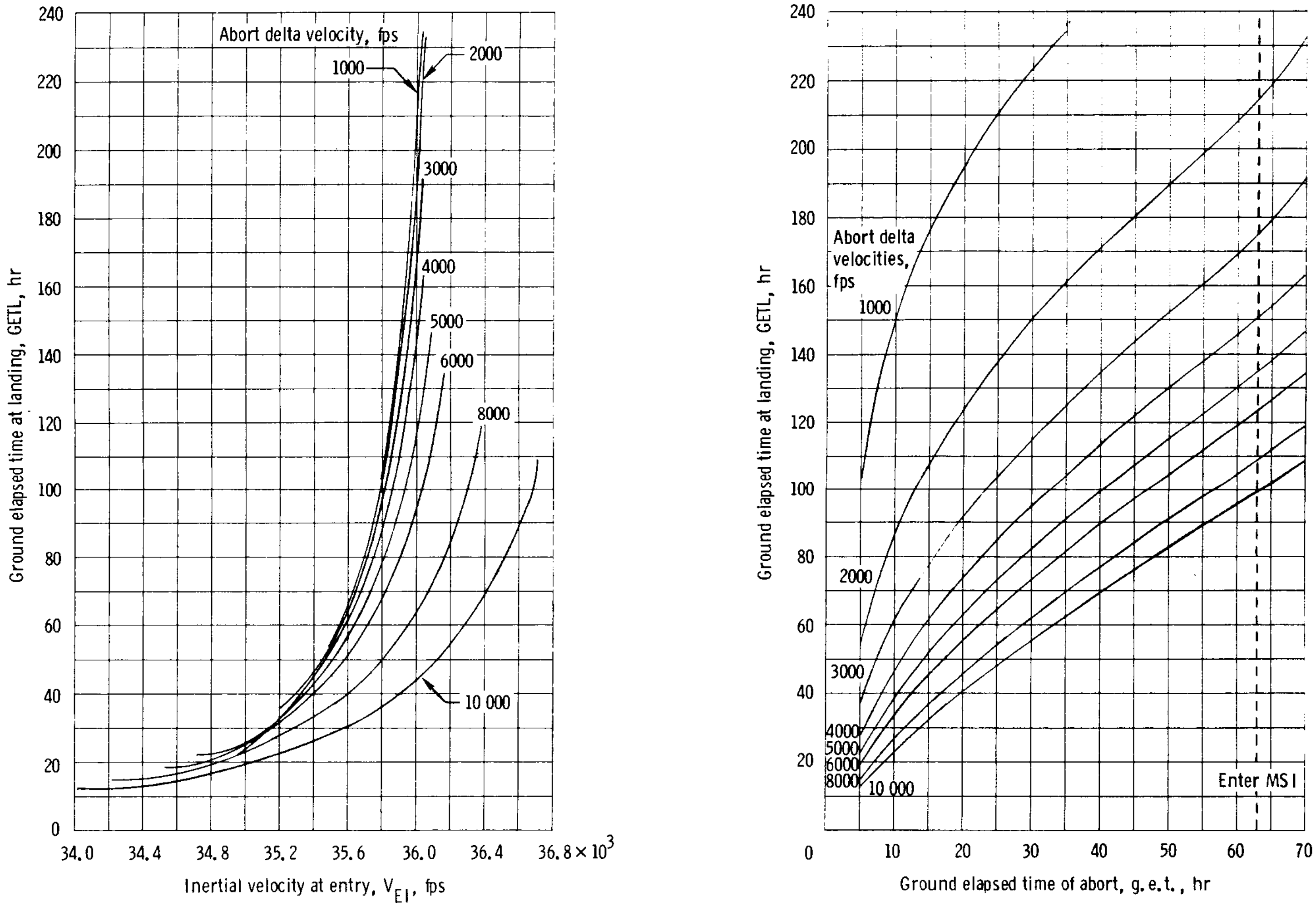


Figure 7-21. - Unspecified area abort analysis during nominal translunar coast. (May 17, 1969 launch. $\psi_L = 72^\circ$, first opportunity.)

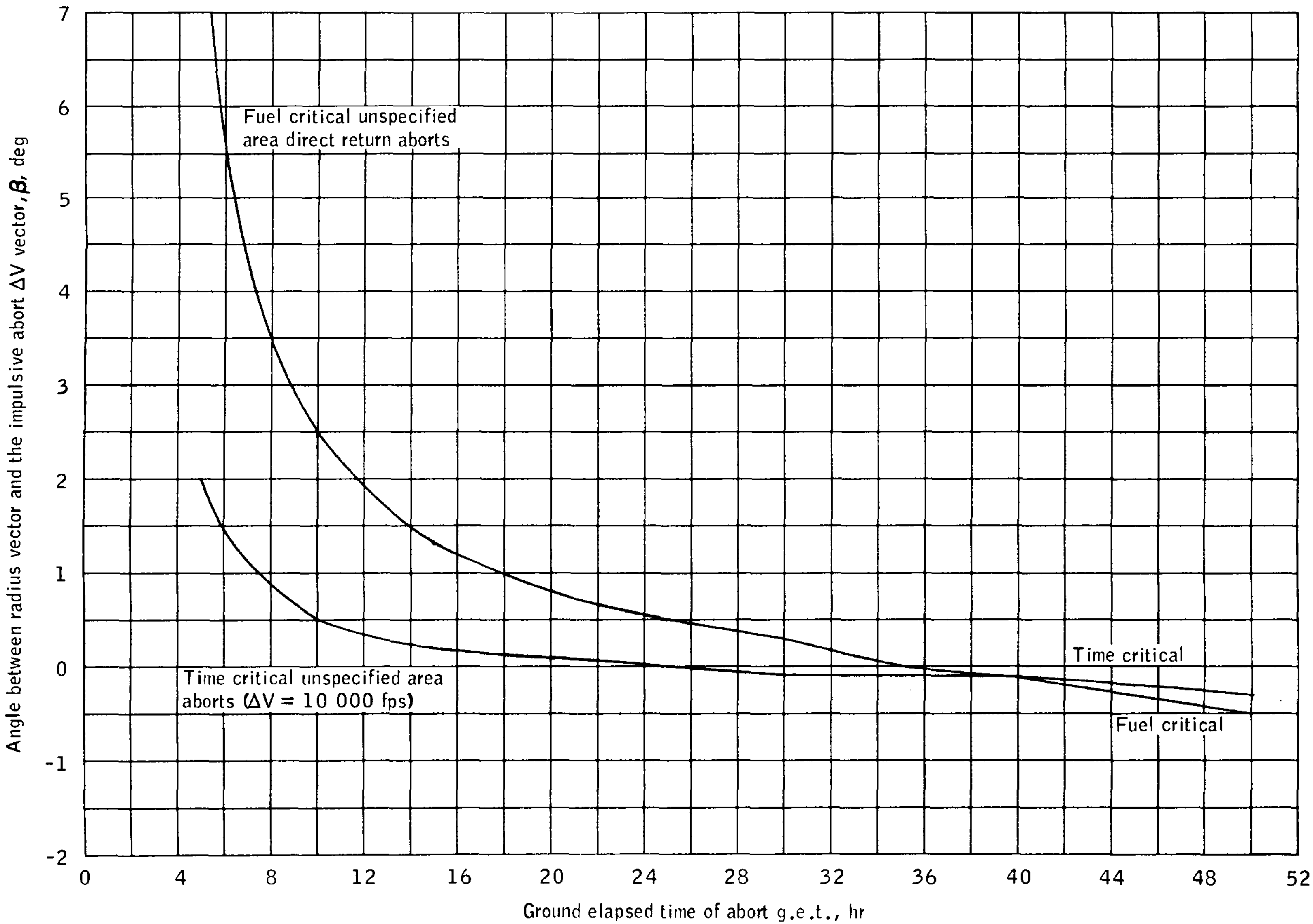
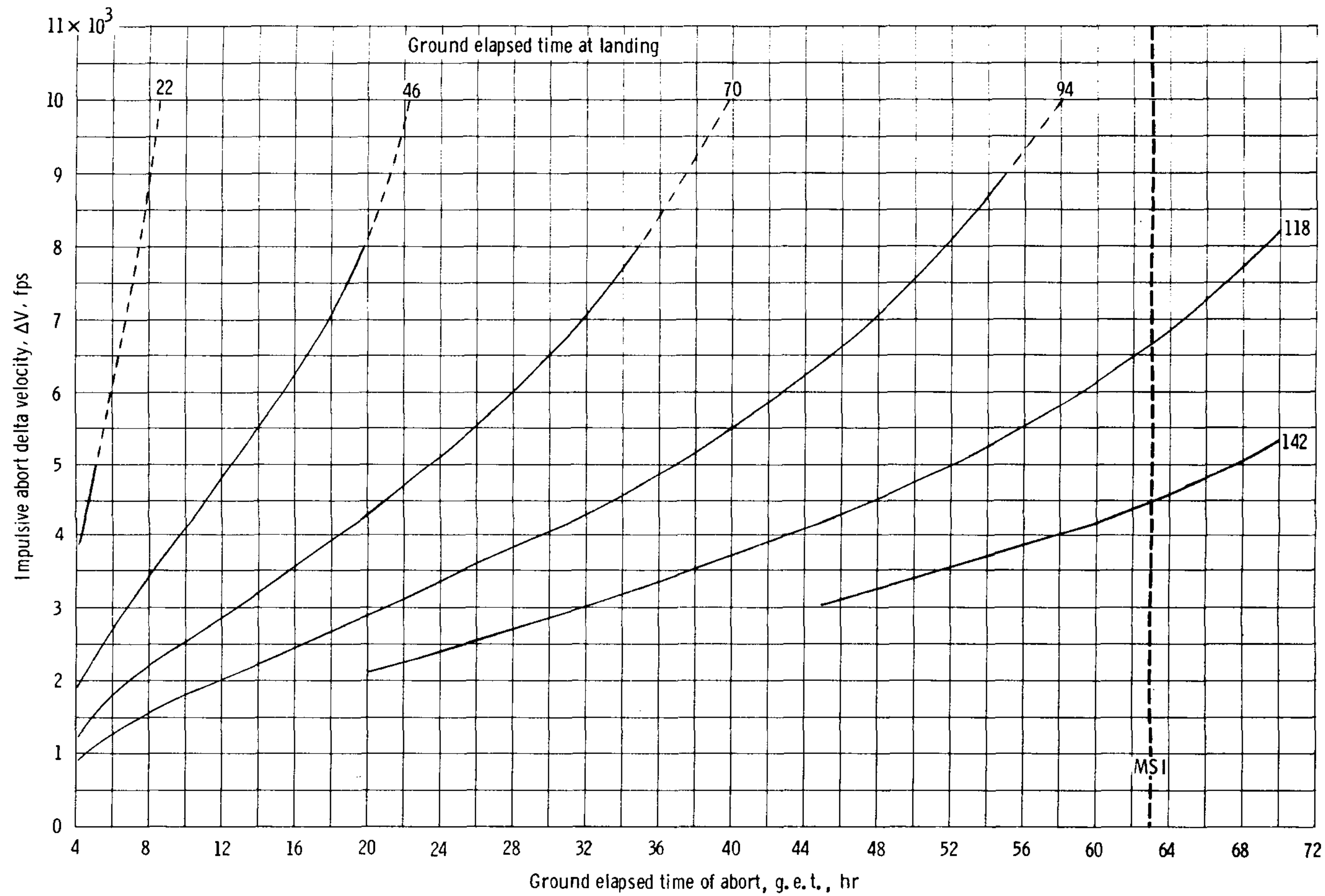
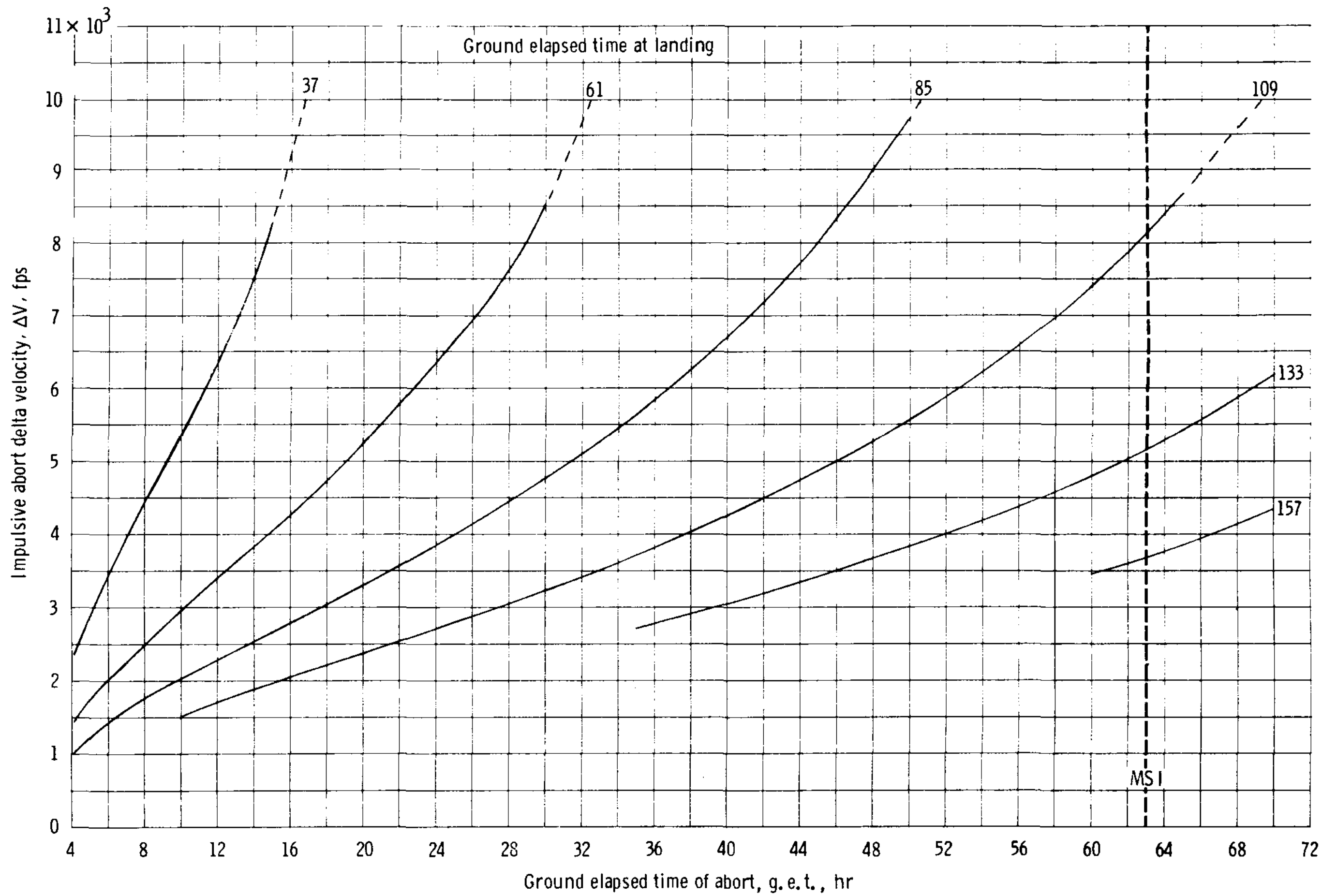


Figure 7-22.- Abort attitude limits for direct return aborts from the nominal translunar coast. (May 17, 1969 launch. $\psi_L = 72°$, first opportunity.)



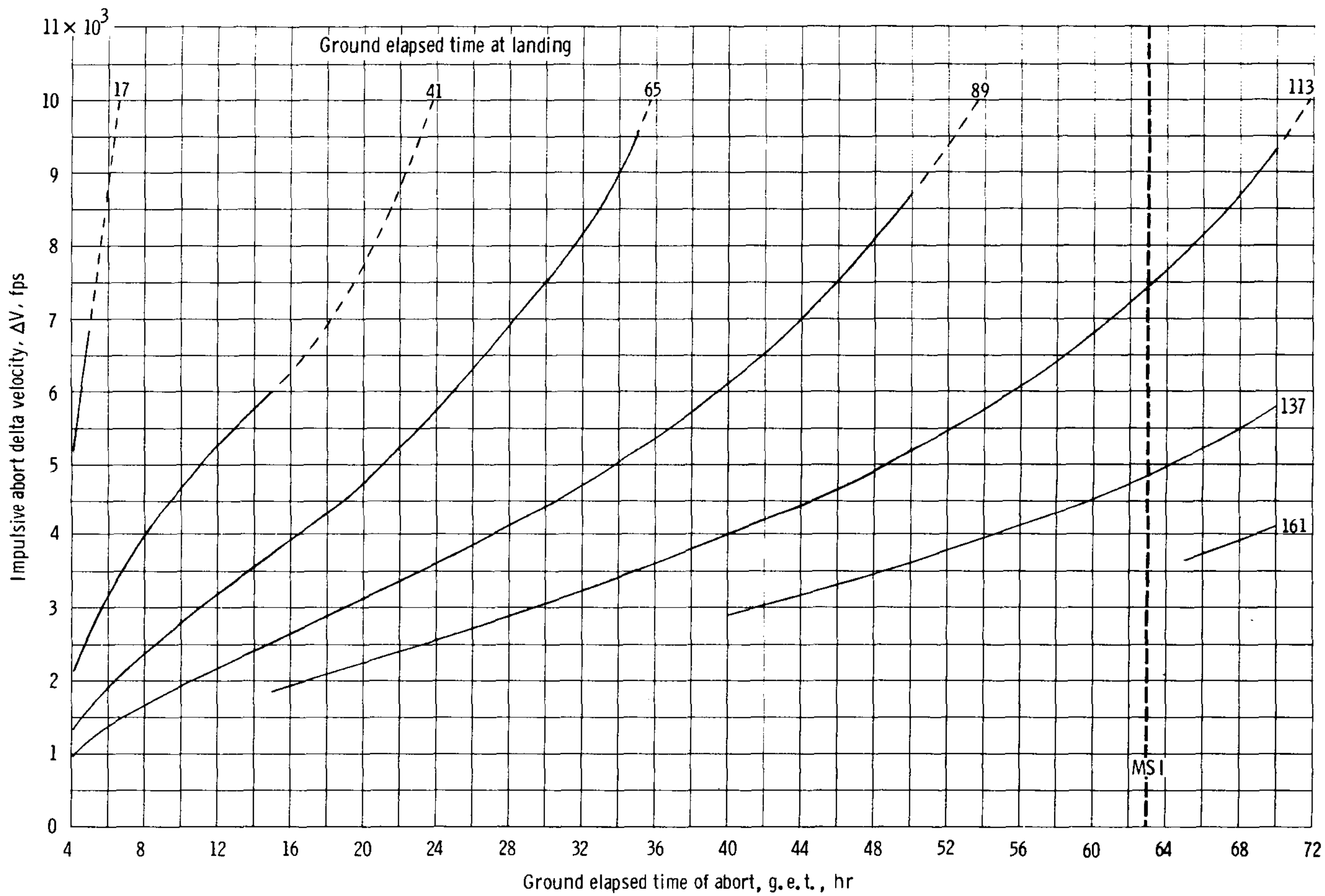
(a) MPL.

Figure 7-23. - Abort ΔV required to achieve ground elapsed time at landing at the contingency landing areas. (May 17, 1969 launch. First injection opportunity, $\psi_L = 72°$.)



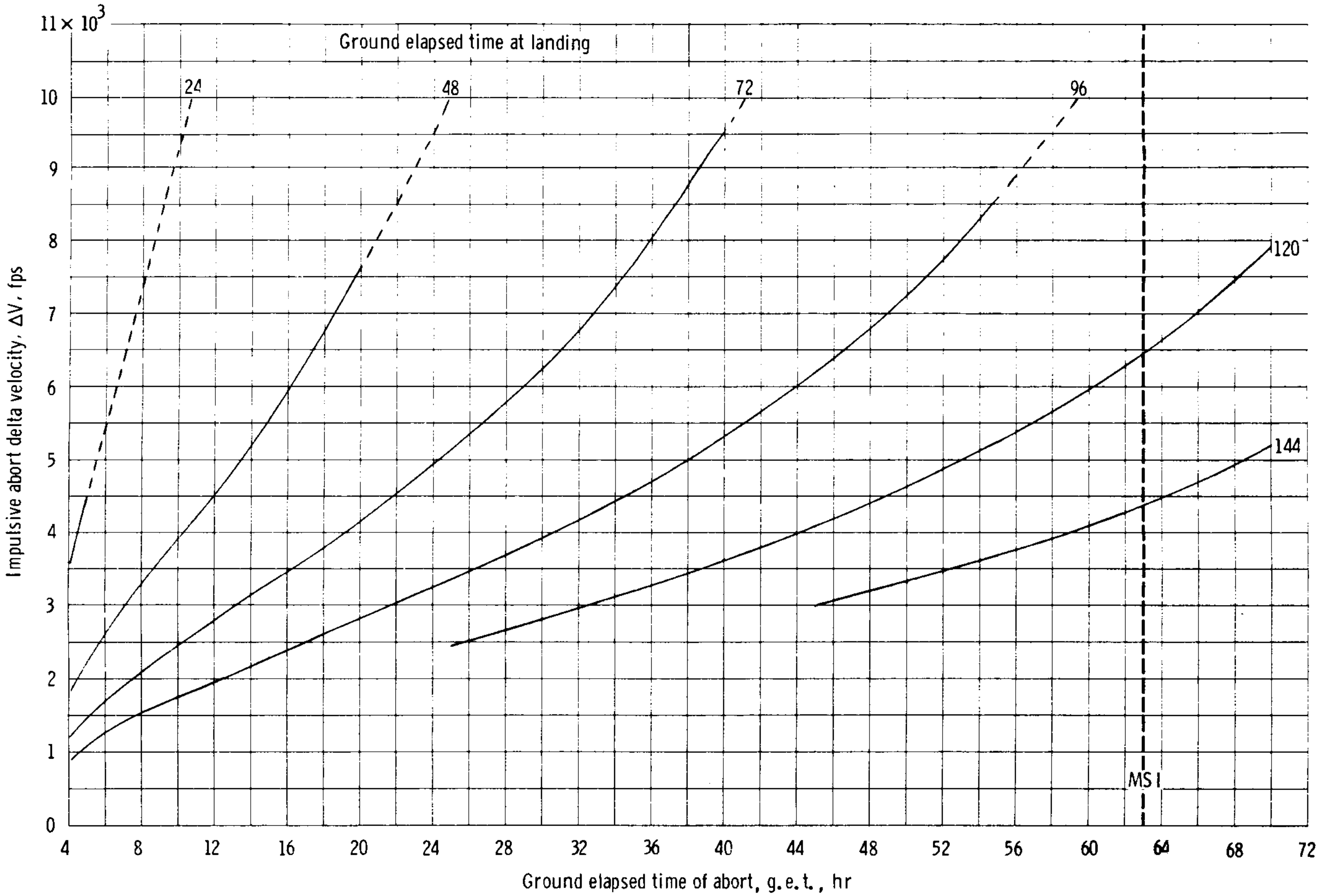
(b) AOL.

Figure 7-23. - Continued.



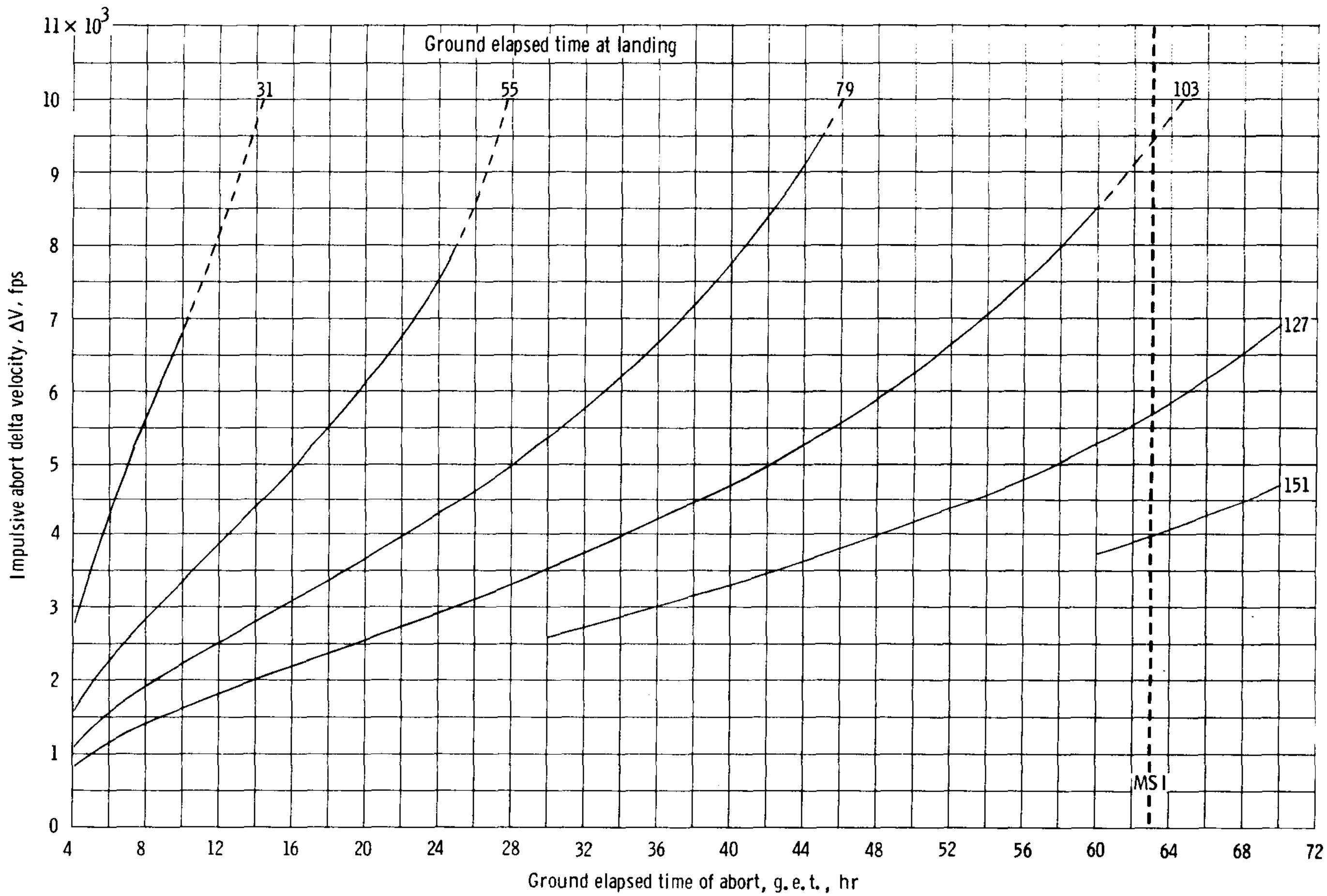
(c) EPL.

Figure 7-23. - Continued.



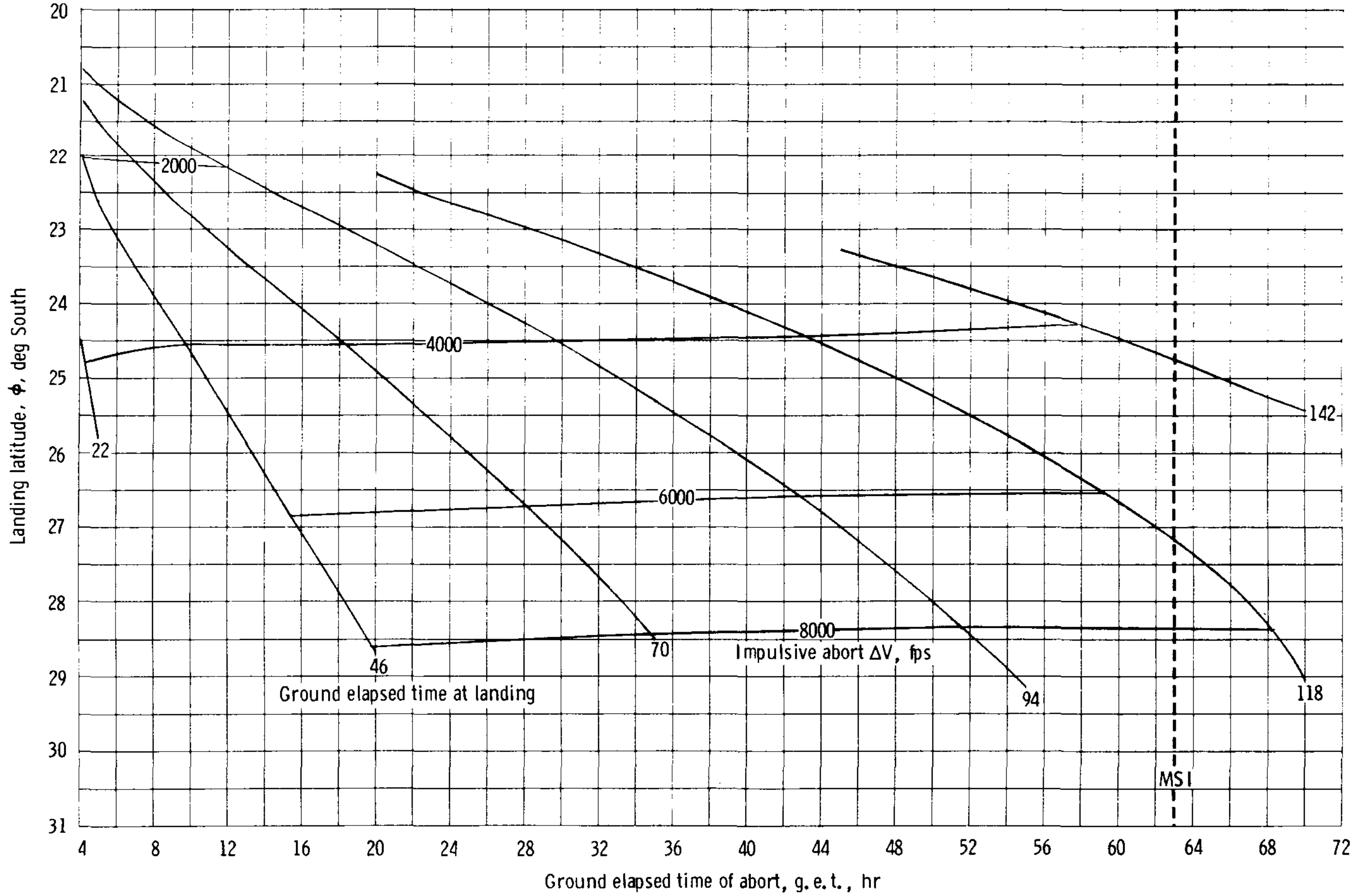
(d) WPL.

Figure 7-23. - Continued.



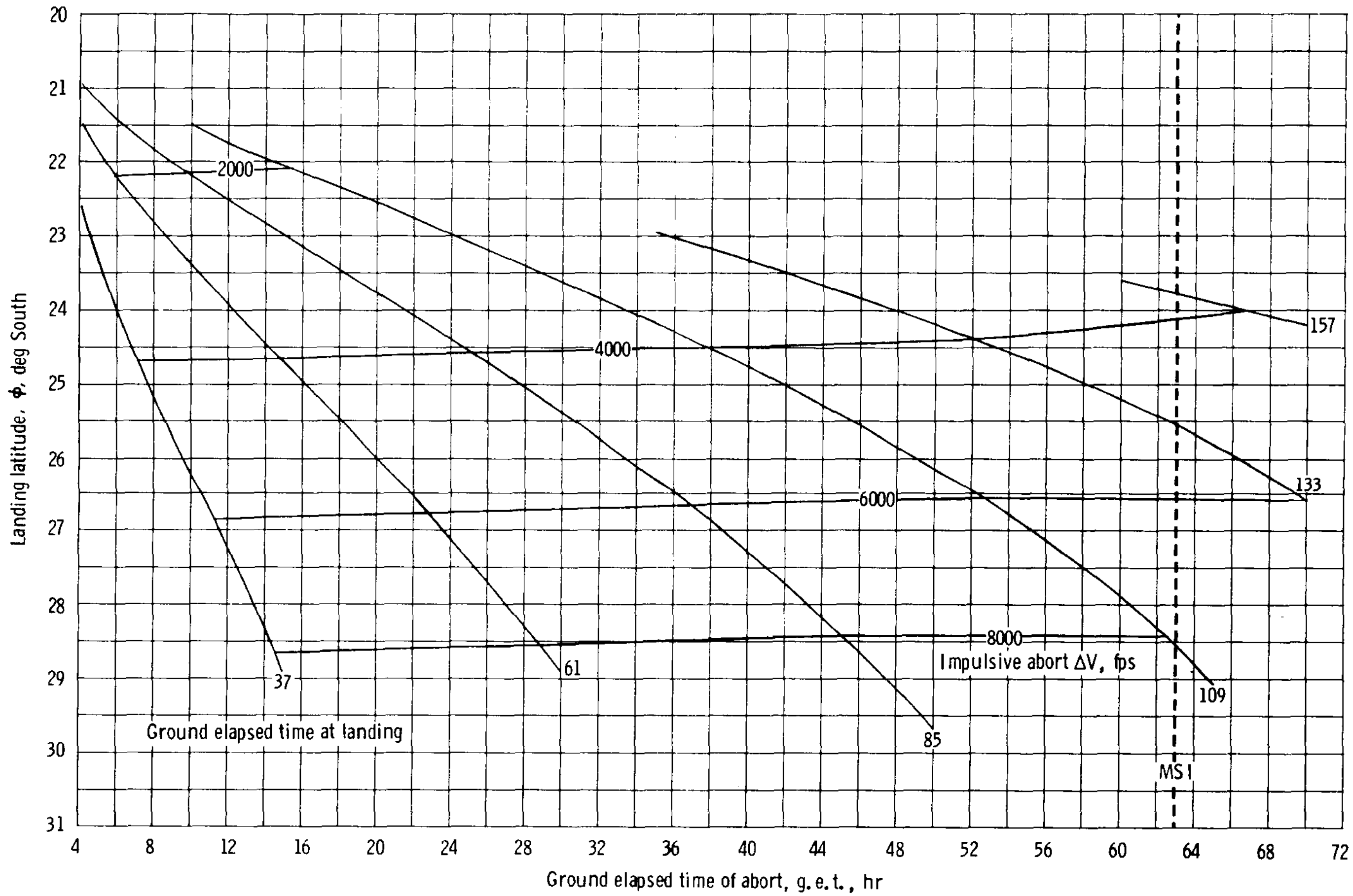
(e) IOL.

Figure 7-23 - Concluded.



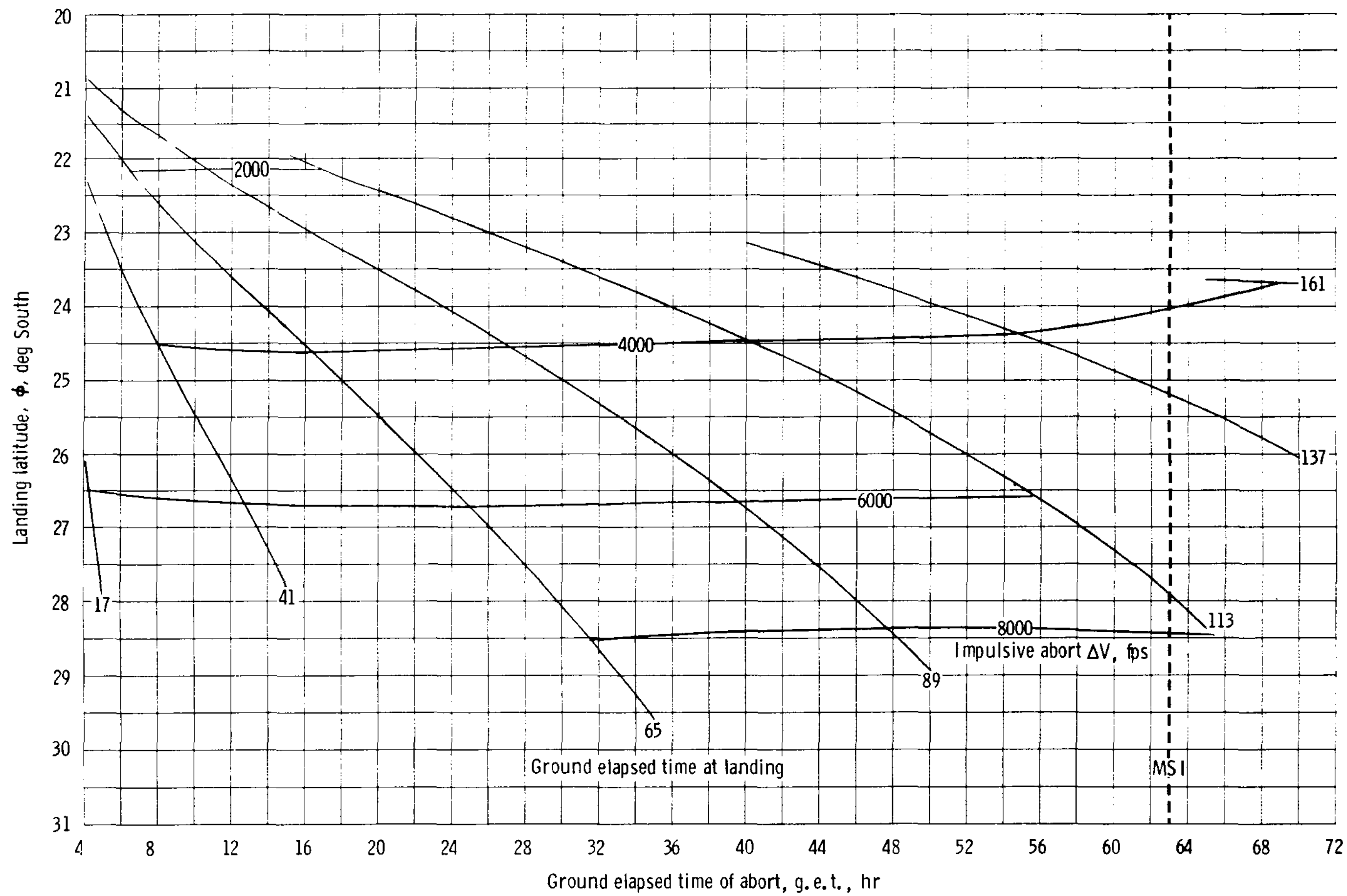
(a) MPL.

Figure 7-24. - Landing latitude as a function of abort ΔV and ground elapsed time at landing at the contingency landing areas. (May 17, 1969 launch. First injection opportunity, $\psi_L = 72°$.)



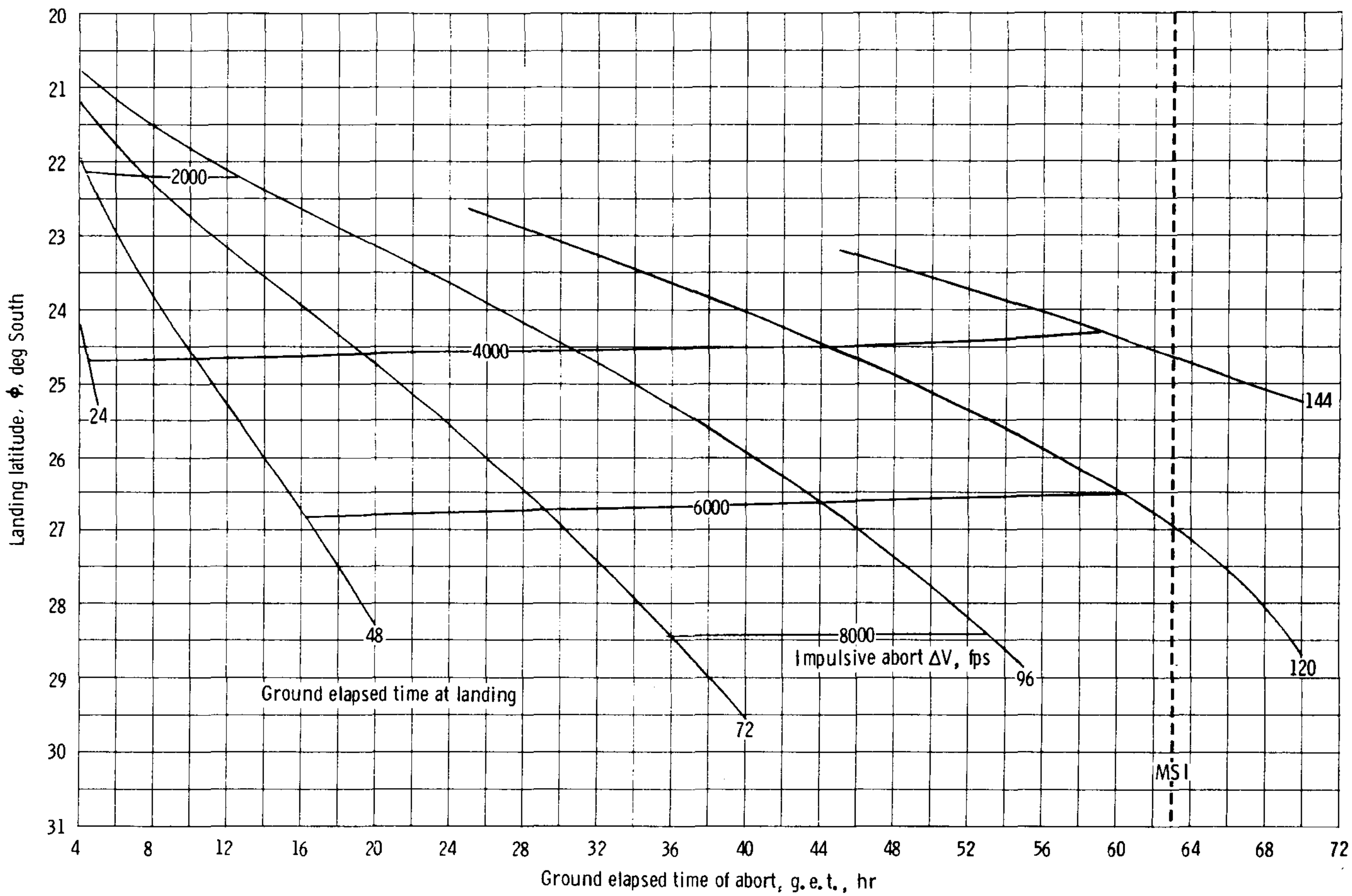
(b) AOL.

Figure 7-24. - Continued.



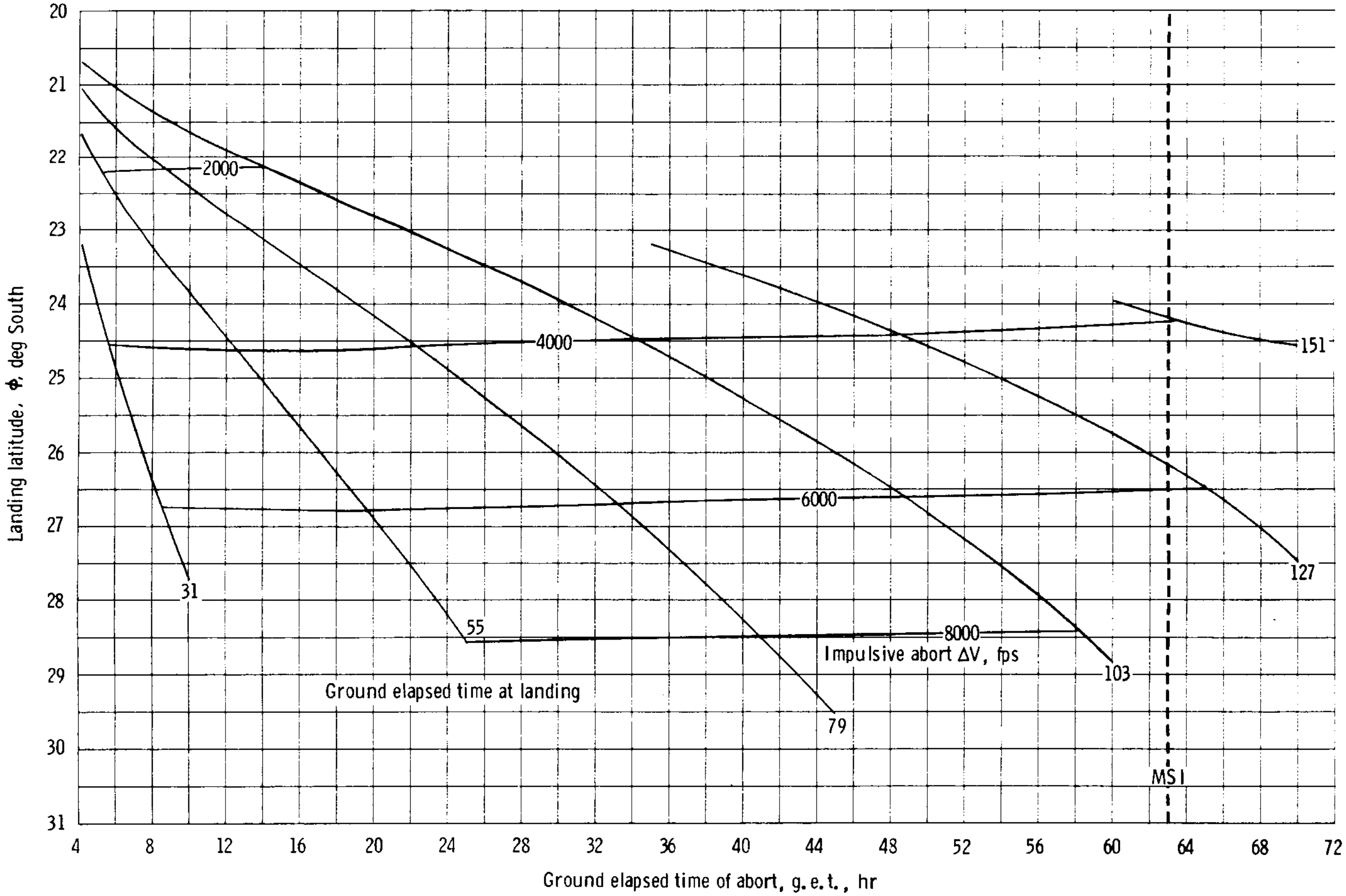
(c) EPL.

Figure 7-24. - Continued.



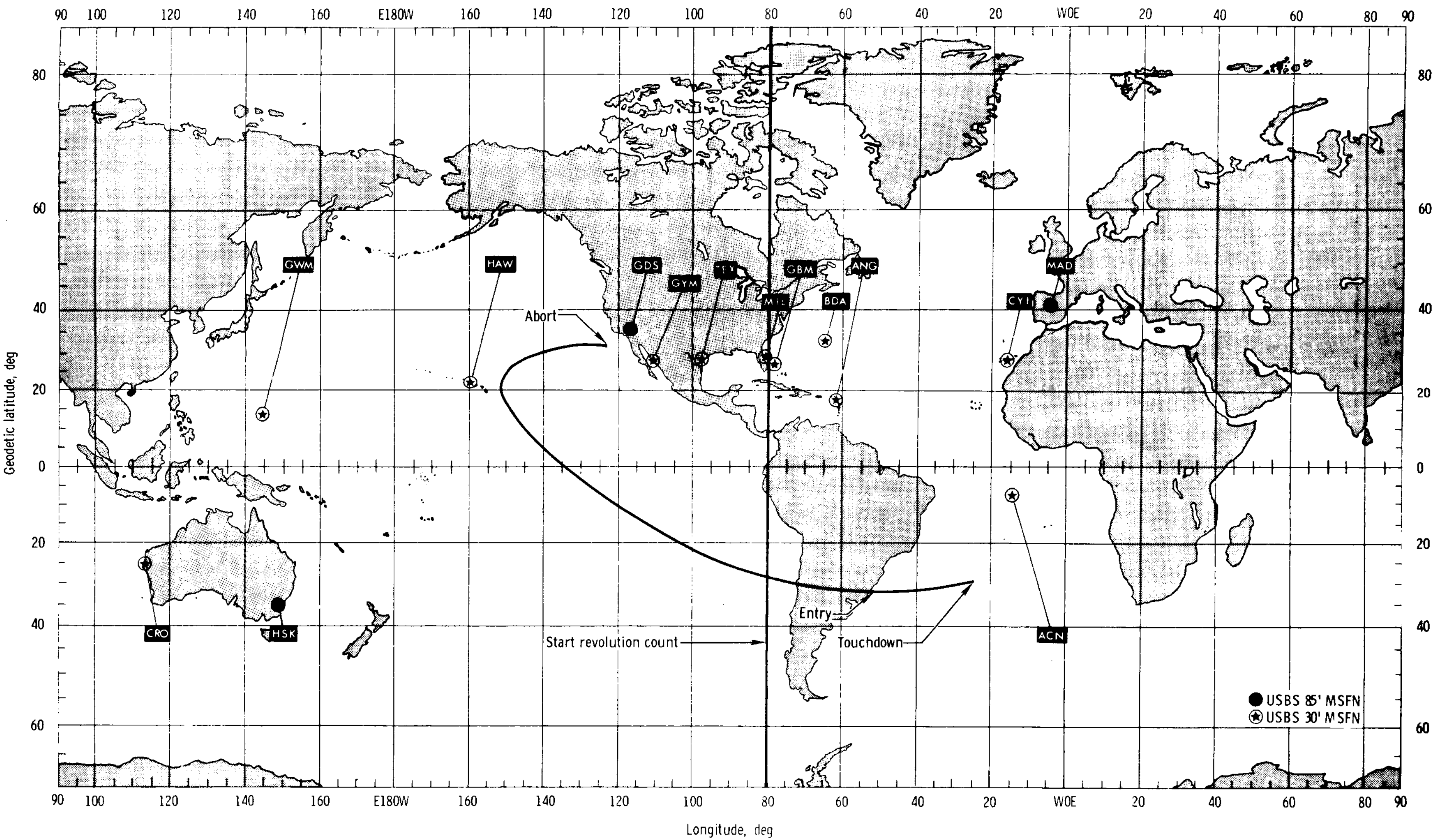
(d) WPL.

Figure 7-24. - Continued.



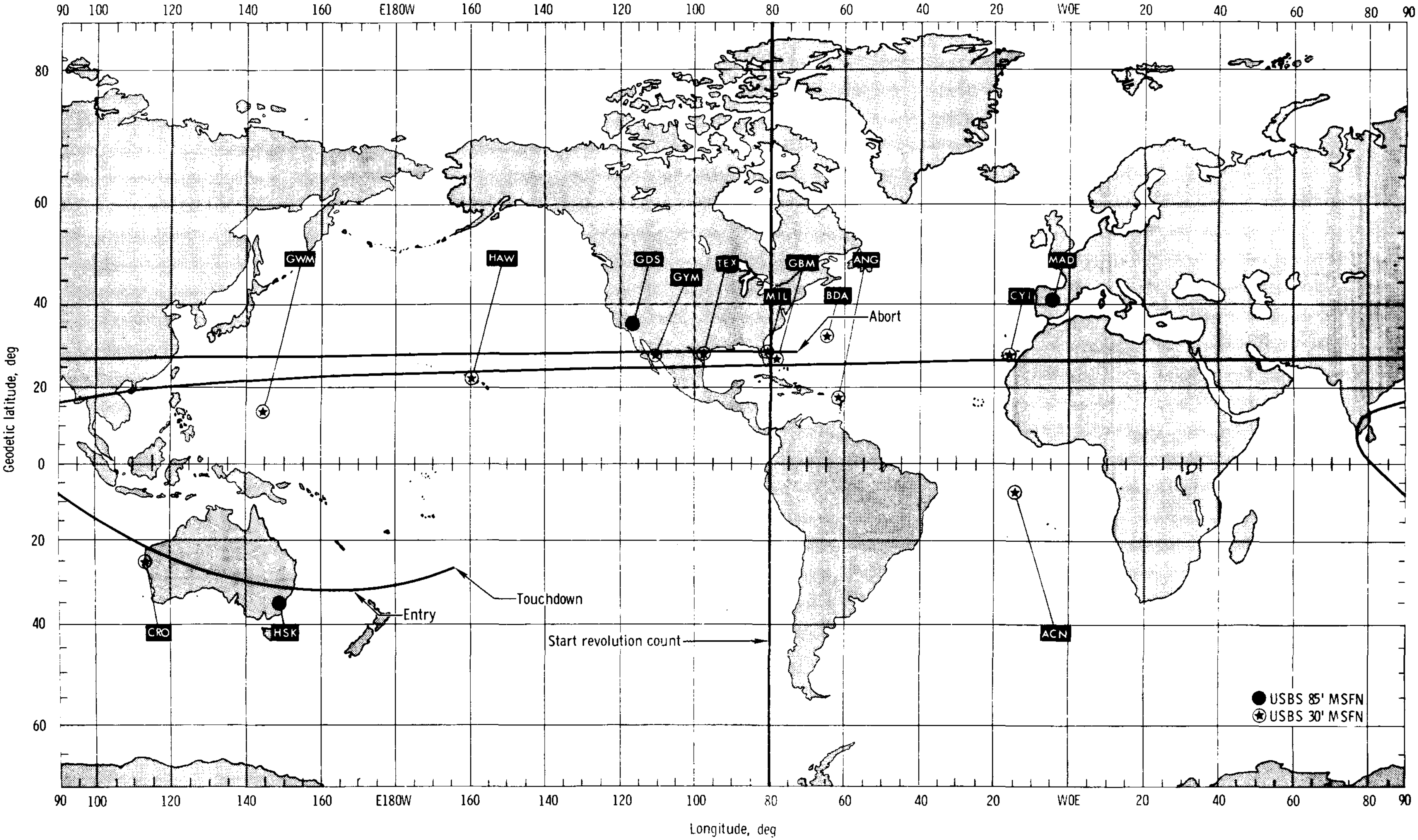
(e) 10L.

Figure 7-24. - Concluded.



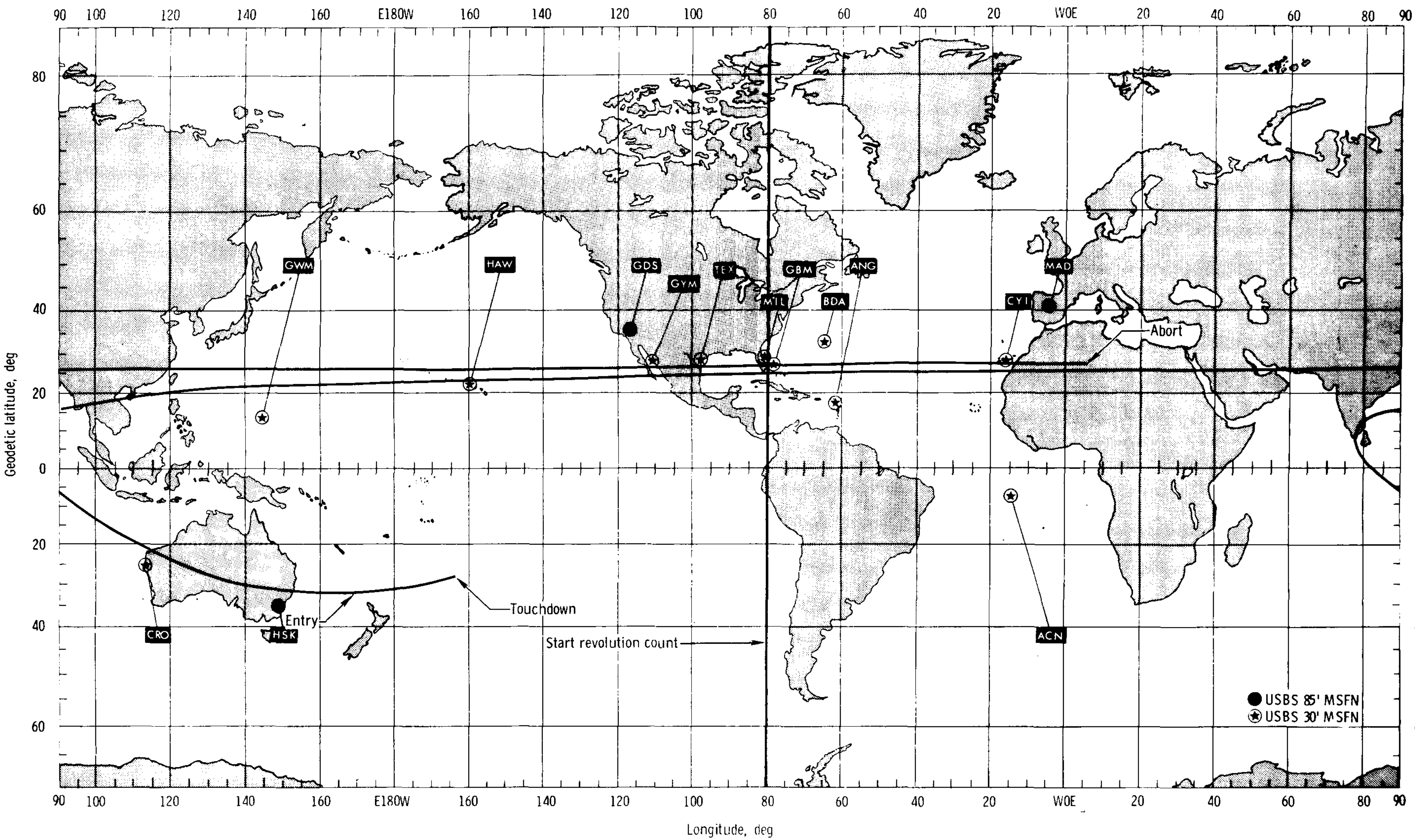
(a) 90 minute abort

Figure 7-25. - Postabort groundtracks for various abort times during TLC.



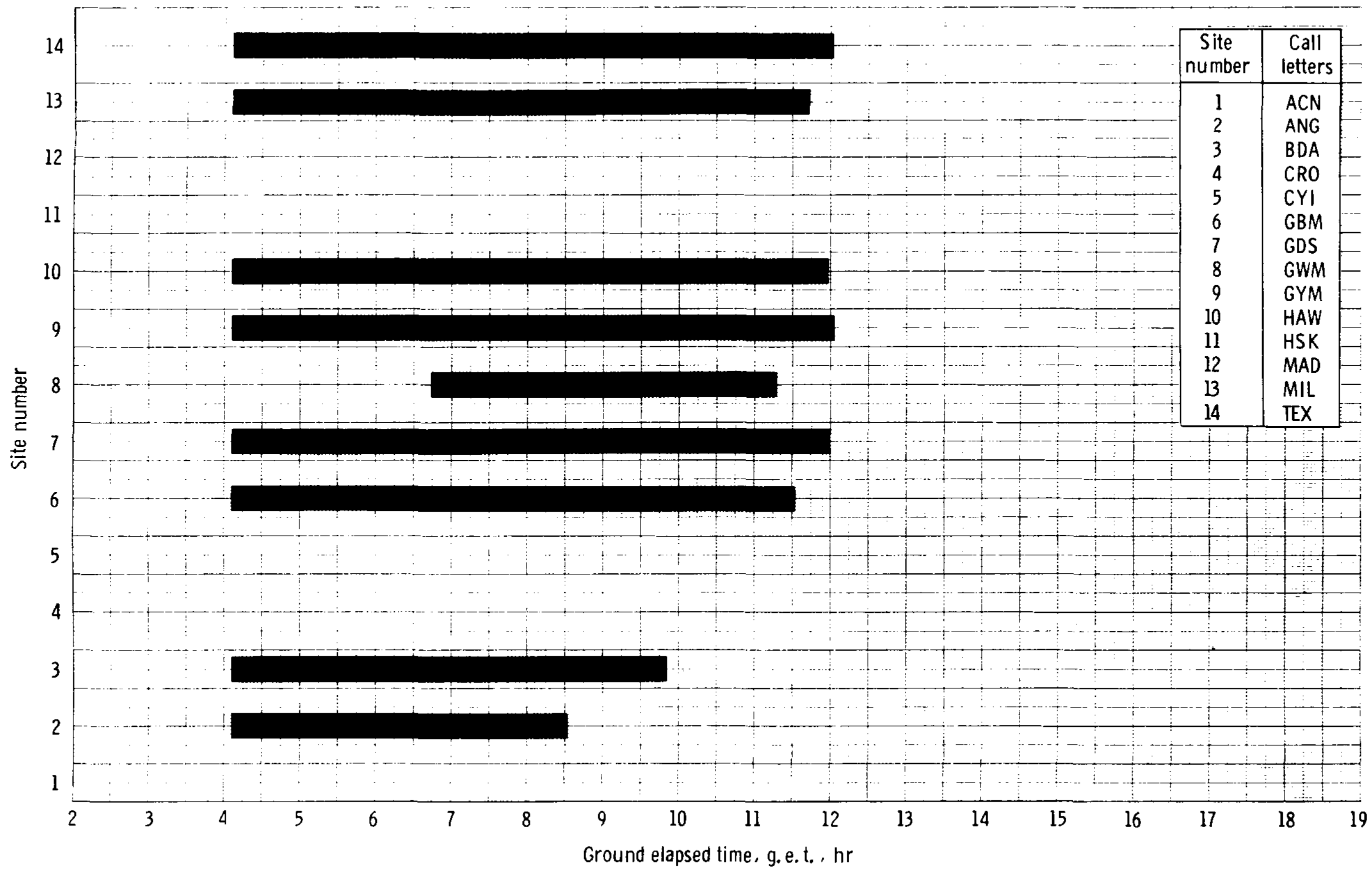
(b) 27 hour 30 minute abort.

Figure 7-25. - Continued.



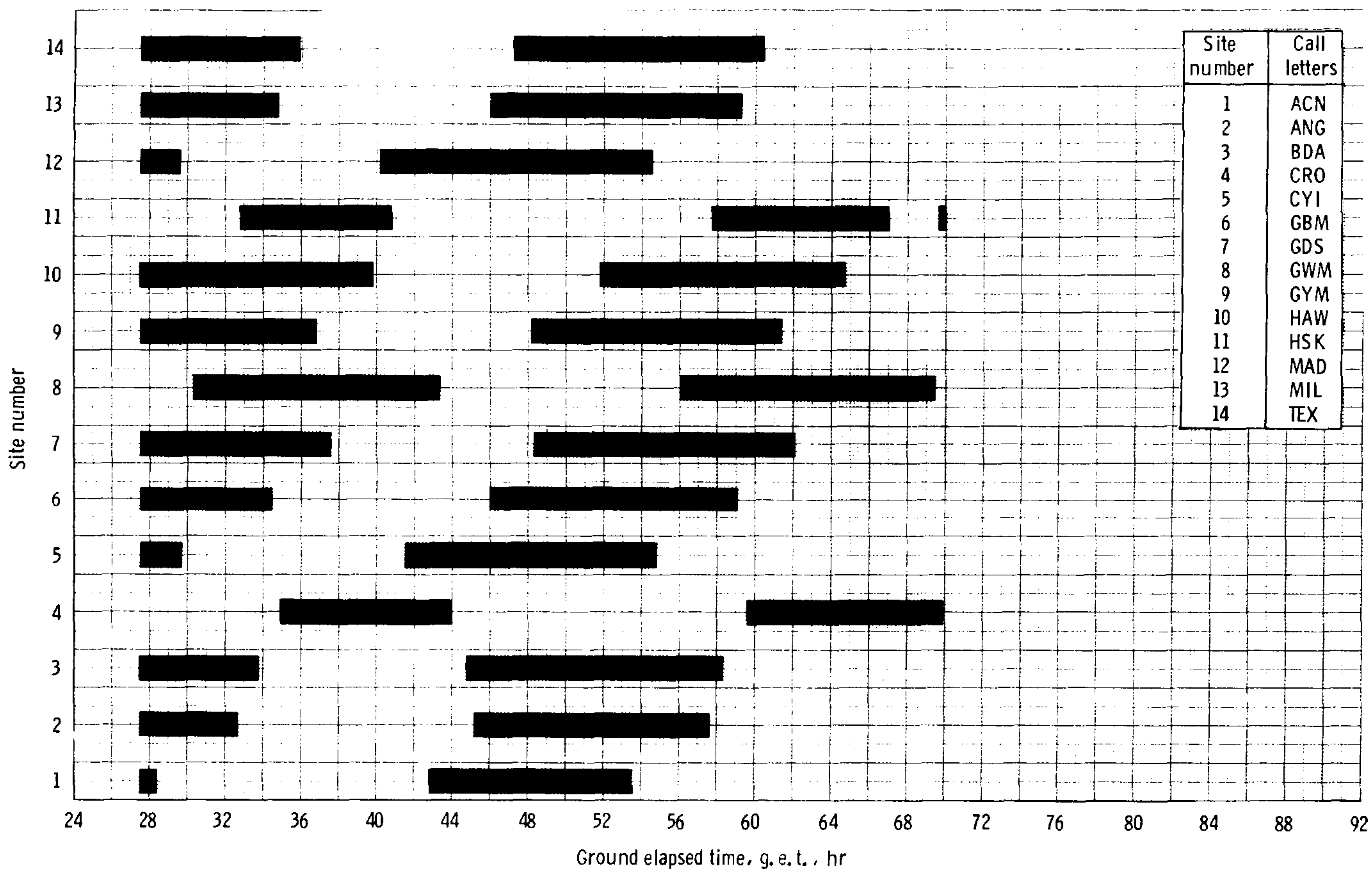
(c) 46 hour 30 minute abort.

Figure 7-25. - Concluded.



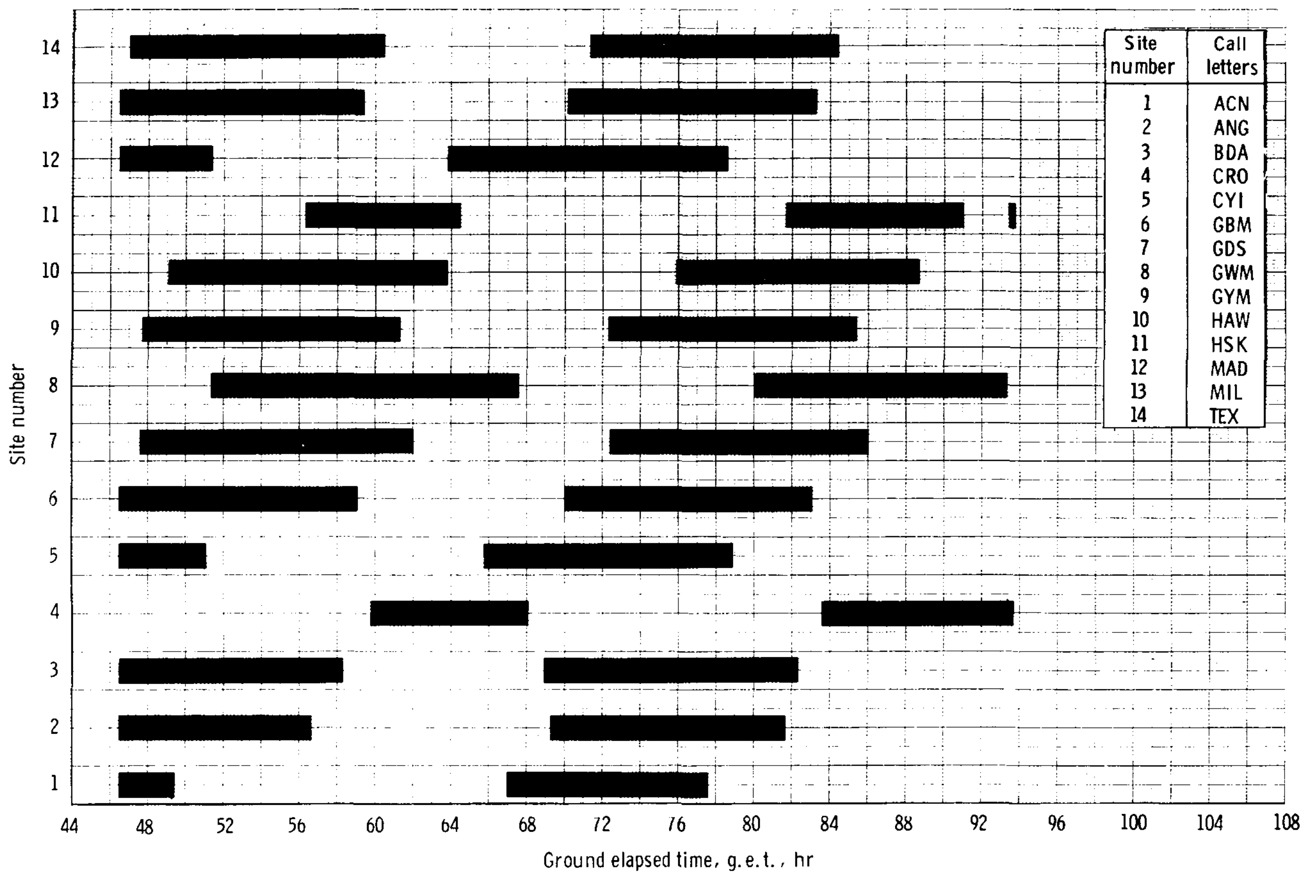
(a) 90 minute abort.

Figure 7-26. - Postabort radar tracking for 5° elevation.



(b) 27 hour 30 minute abort.

Figure 7-26. - Continued.



(c) 46 hour 30 minute abort.

Figure 7-26. - Concluded.

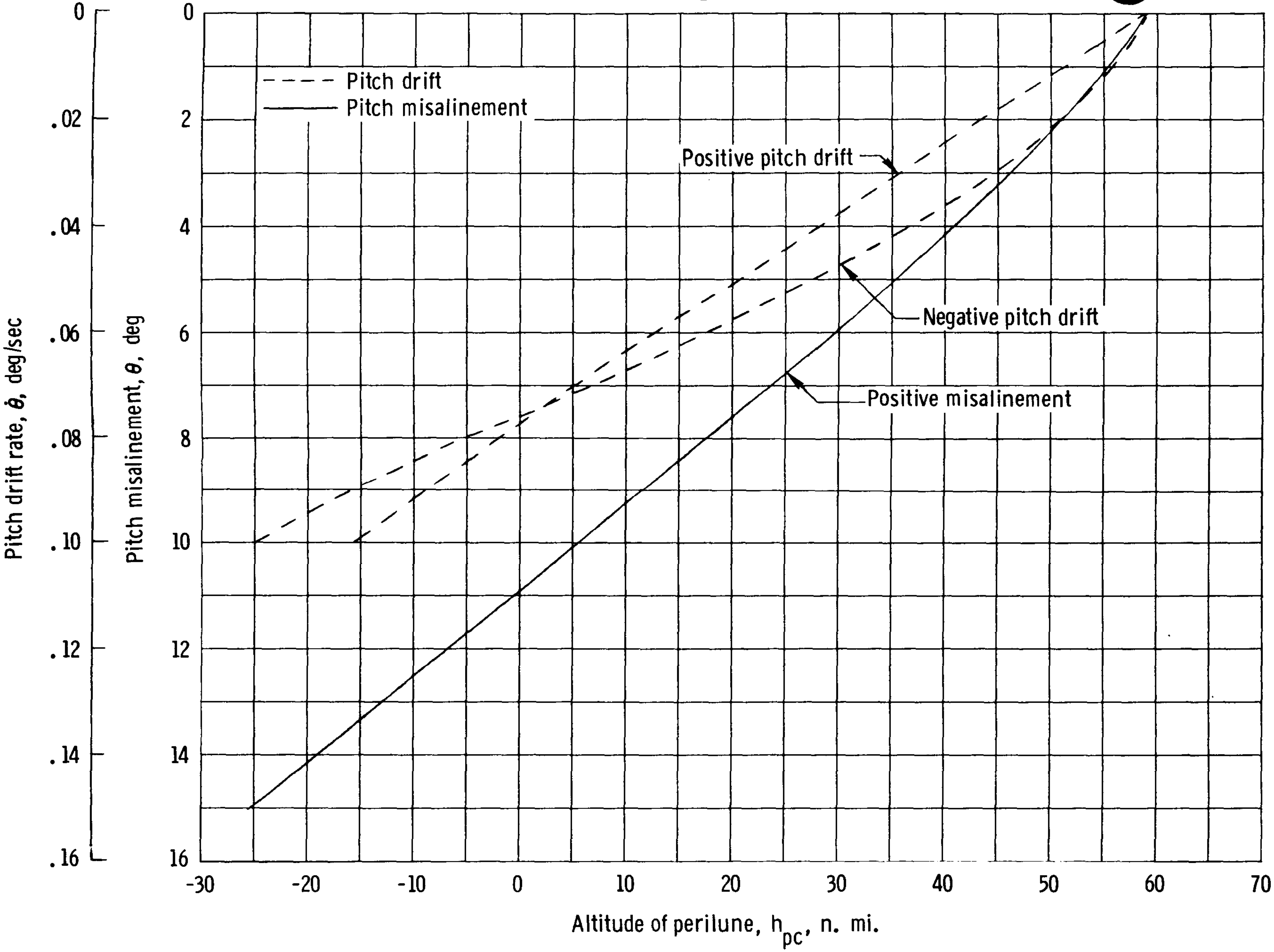


Figure 8-1.- Perilune altitude for simulated IMU pitch drifts and misalinements during LOI-1.

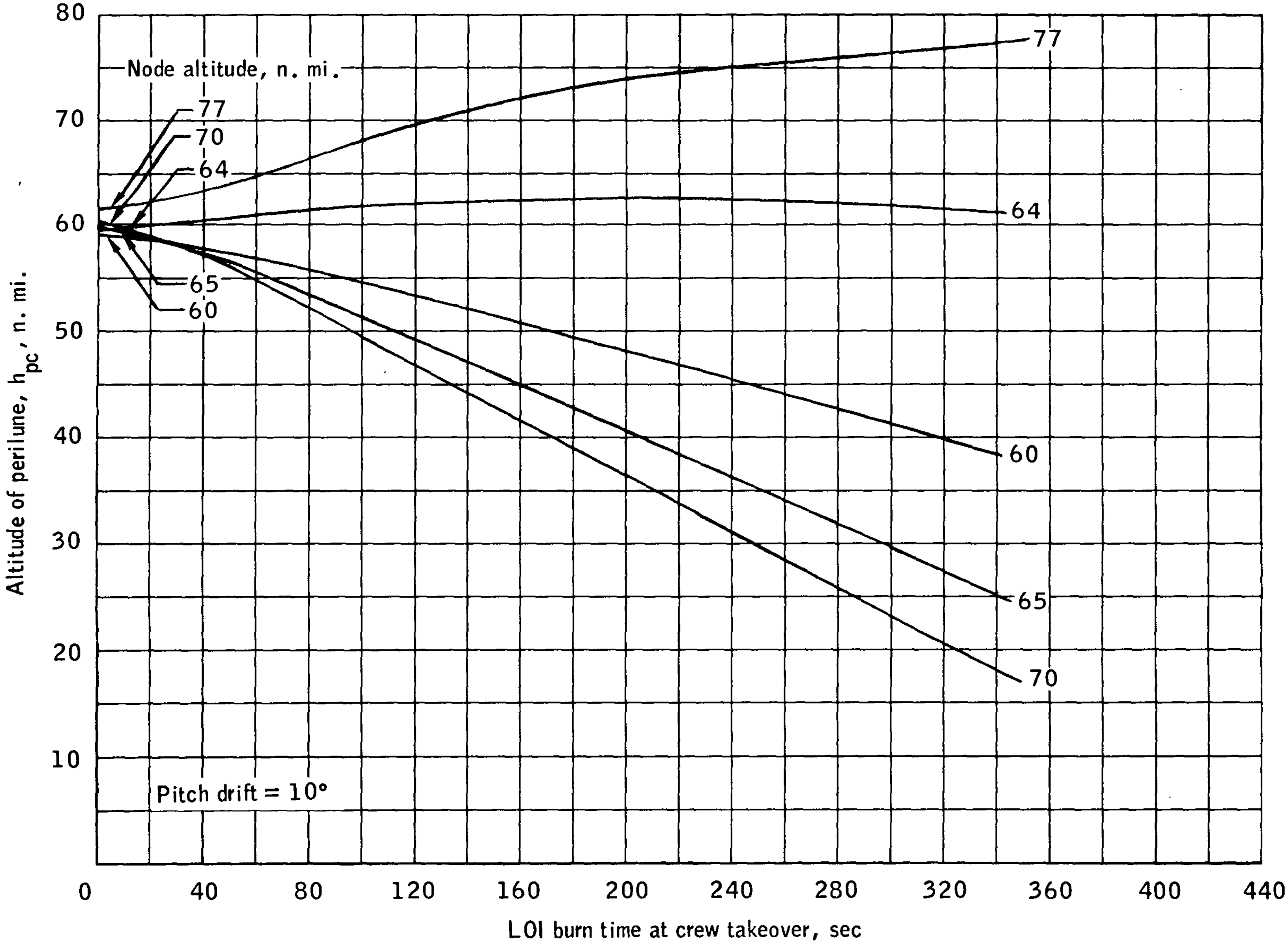


Figure 8-2.- Altitude of perilune as a function of LOI burn time at crew takeover for various nodal altitudes for pitch drift = 10°.

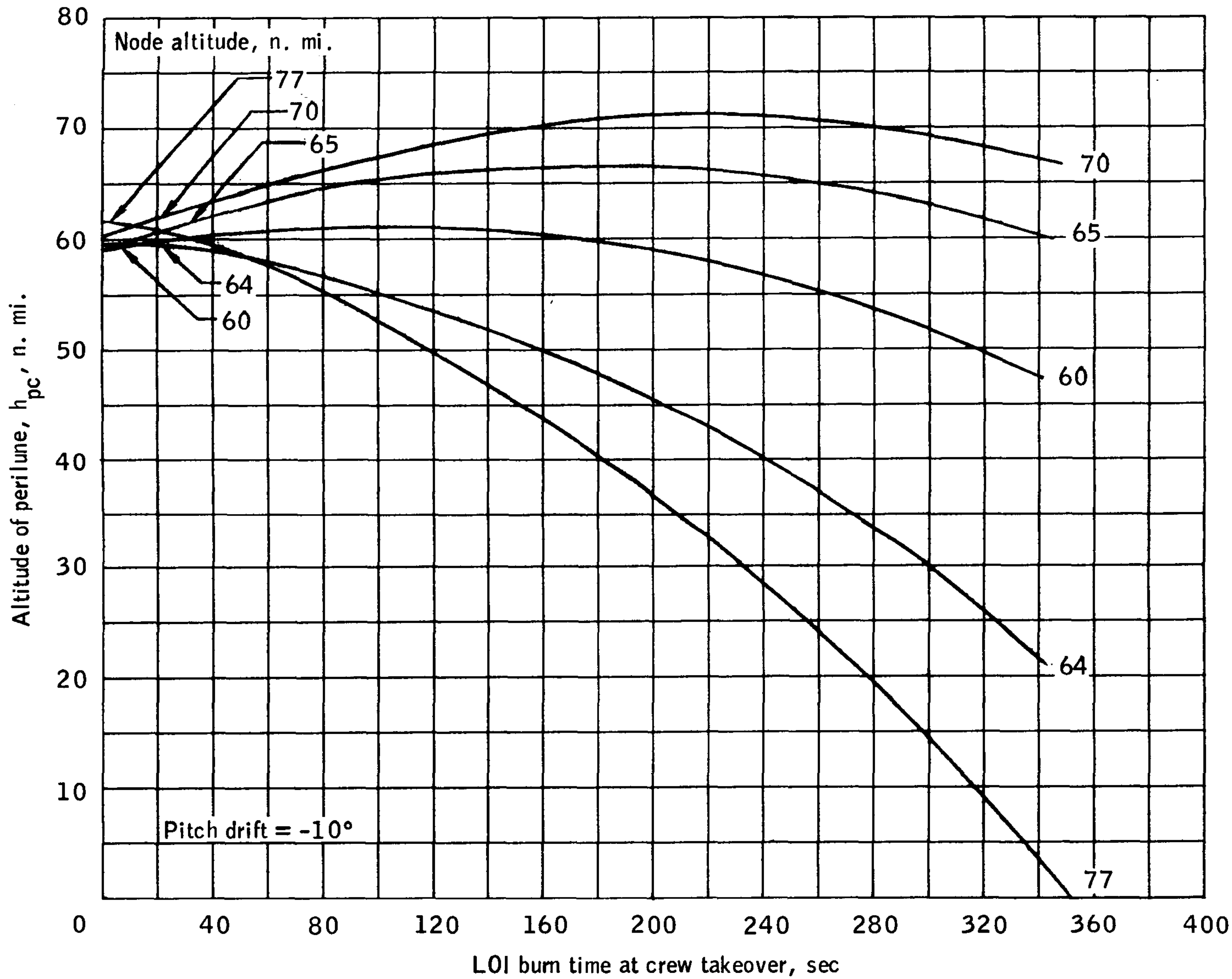


Figure 8-3.- Altitude of perilune as a function of LOI burn time at crew takeover for various nodal altitudes for pitch drift = -10°.

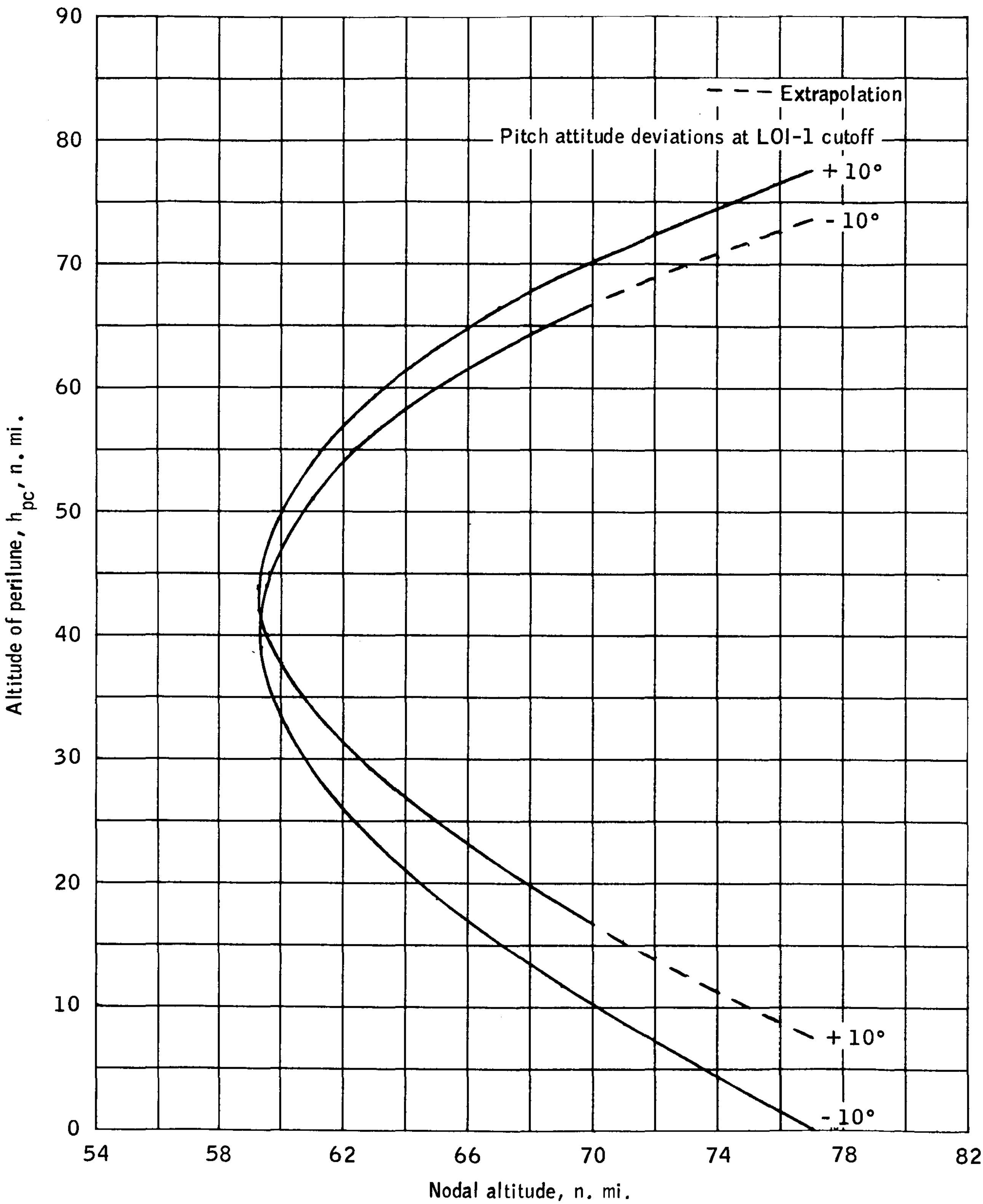


Figure 8-4.- Altitude of perilune as a function of nodal altitude after IMU pitch drifts at ±10°.

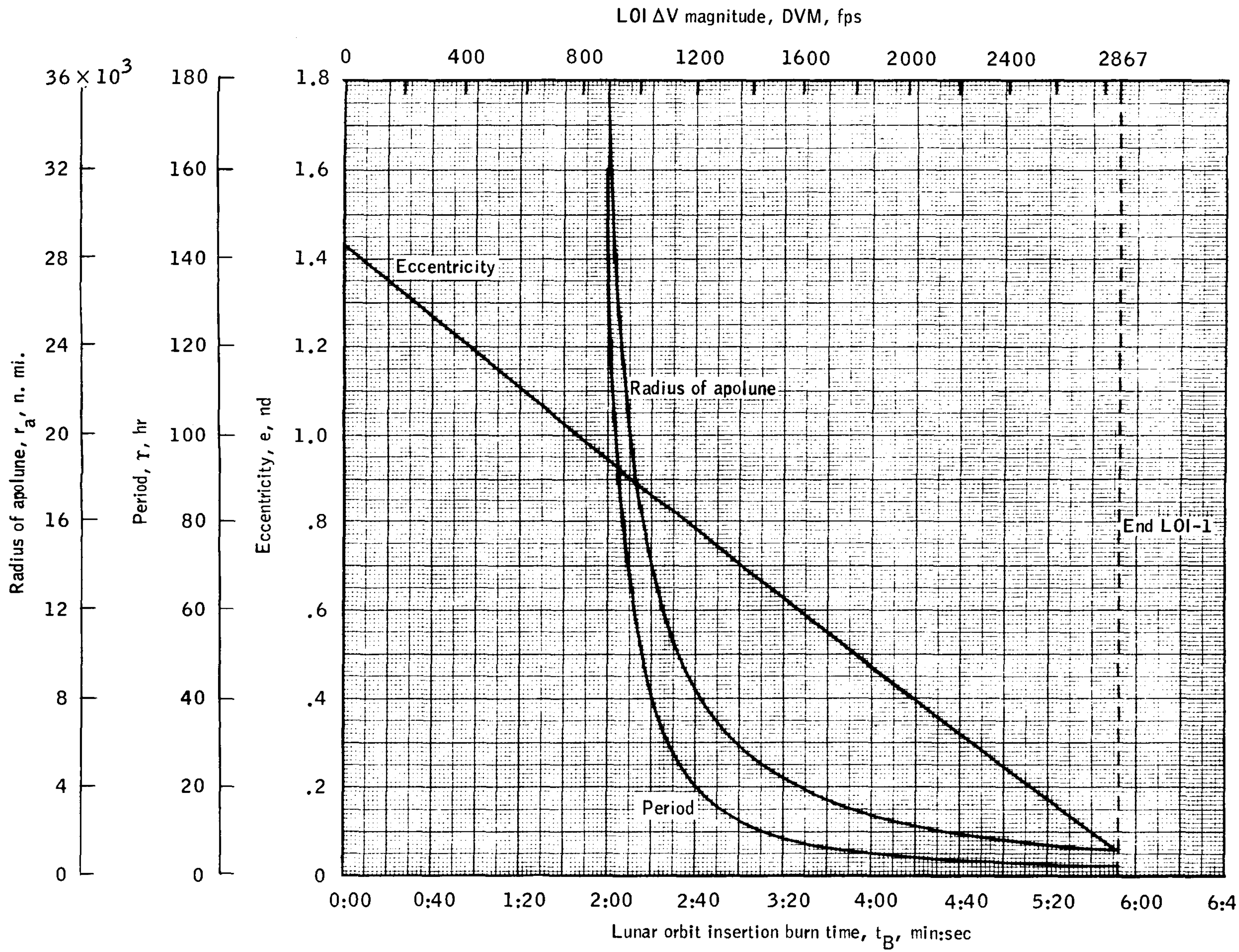


Figure 8-5.- Conic parameters as a function of SPS burn time during the LOI burn.

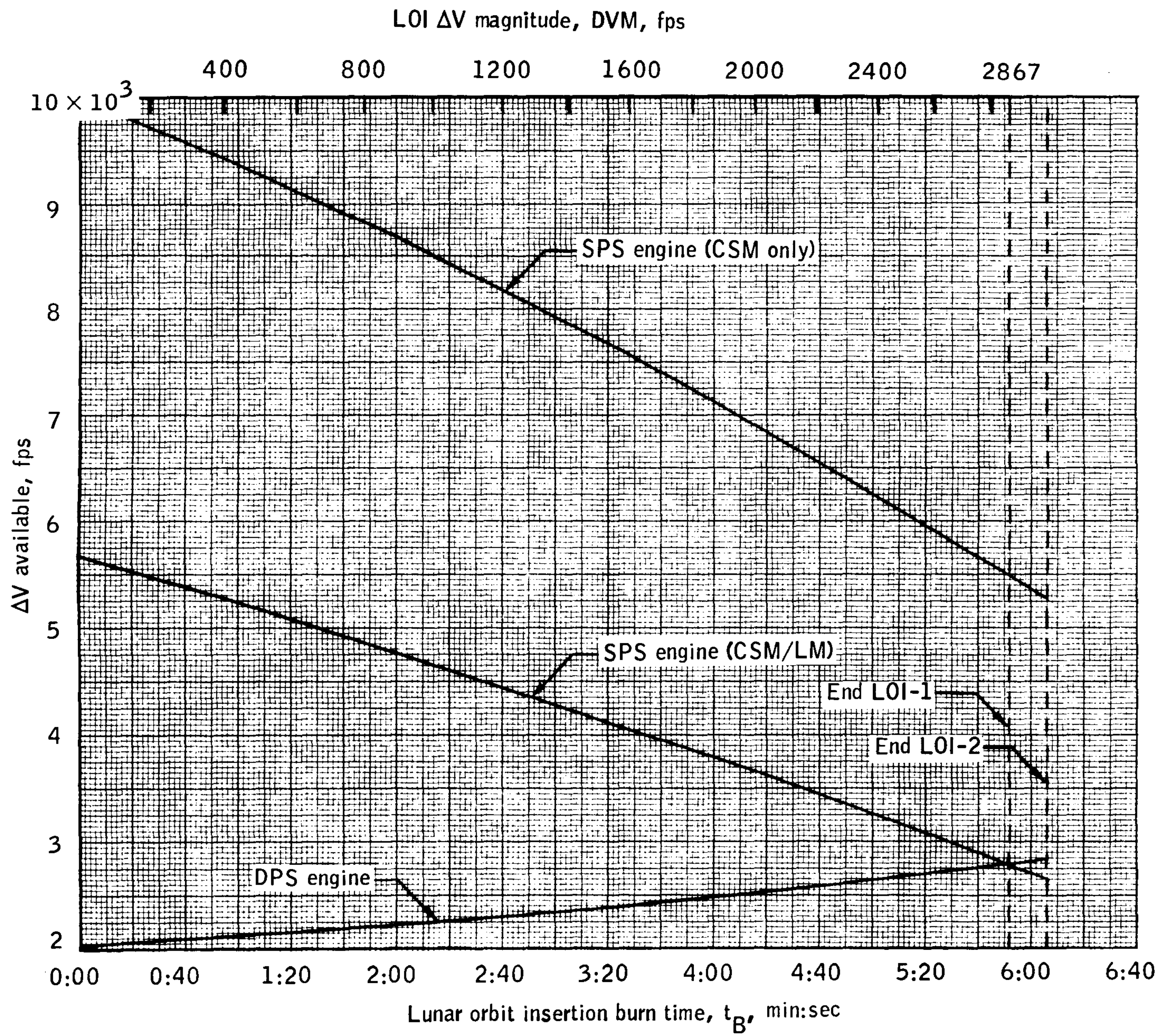
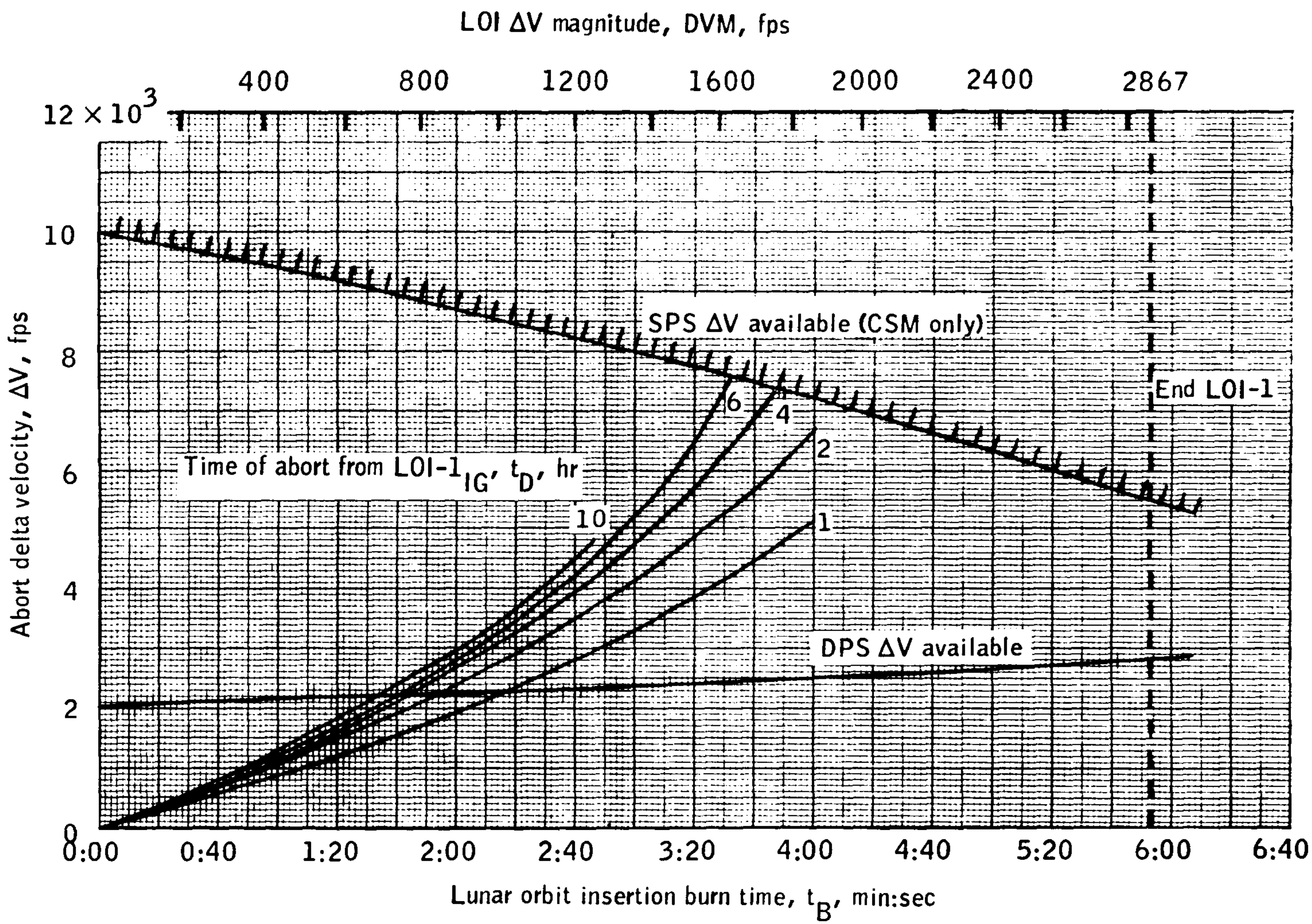
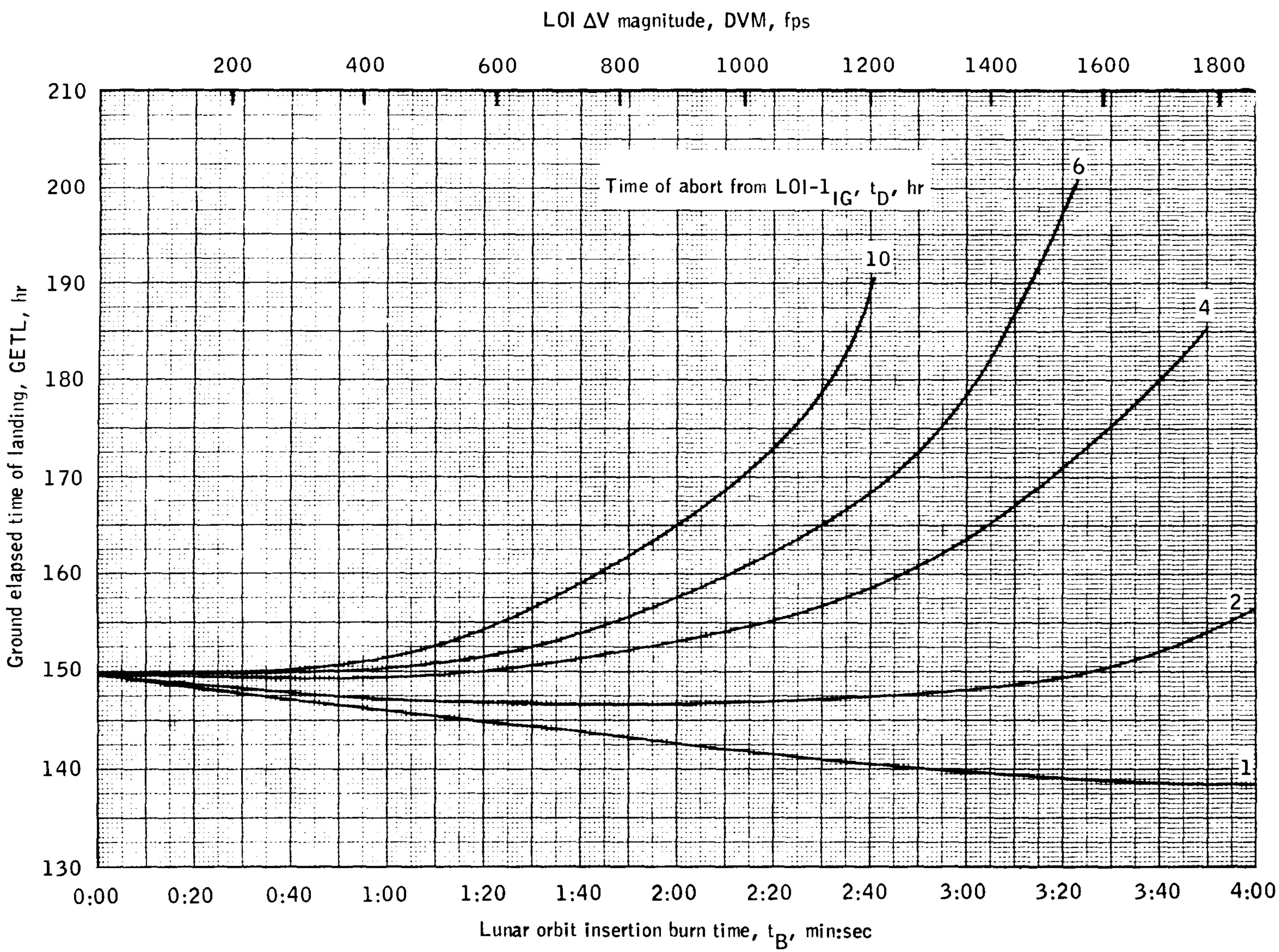


Figure 8-6.- Abort ΔV available following a premature SPS shutdown during the LOI burn.



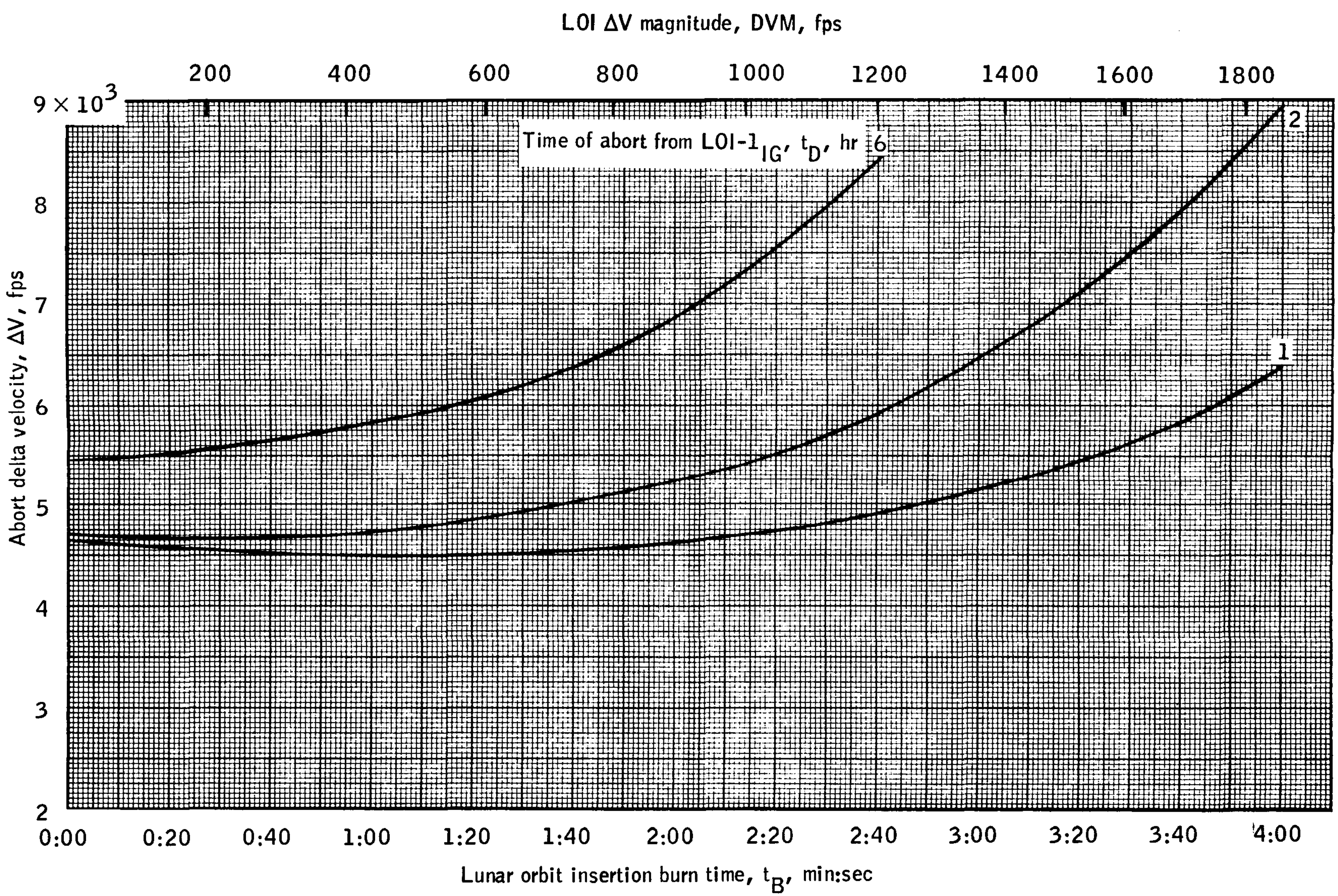
(a) Abort ΔV required for various times of ignition.

Figure 8-7.- Mode I unspecified area abort analysis as a function of LOI burn time.



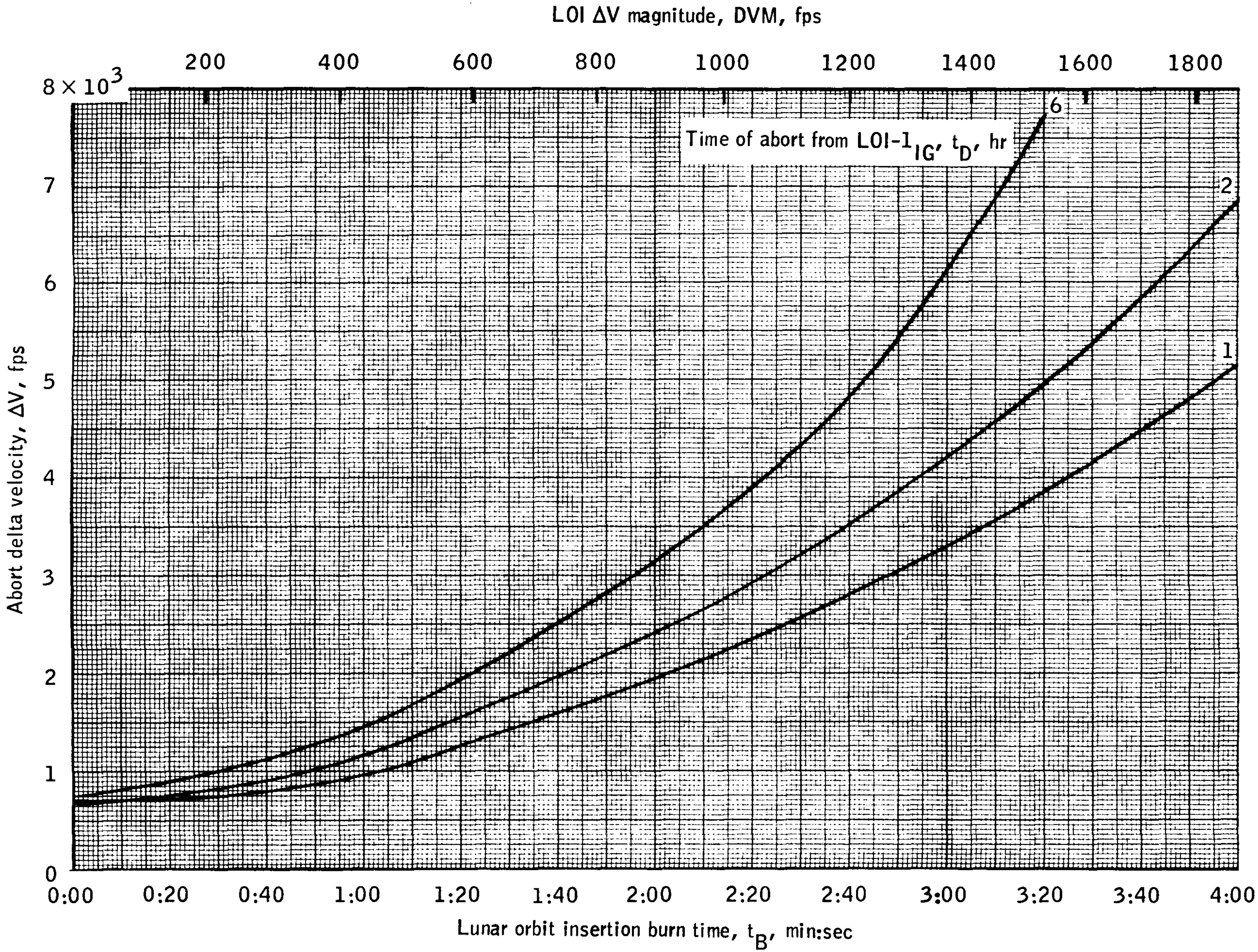
(b) Ground elapsed time of landing for various times of ignition.

Figure 8-7.- Concluded.



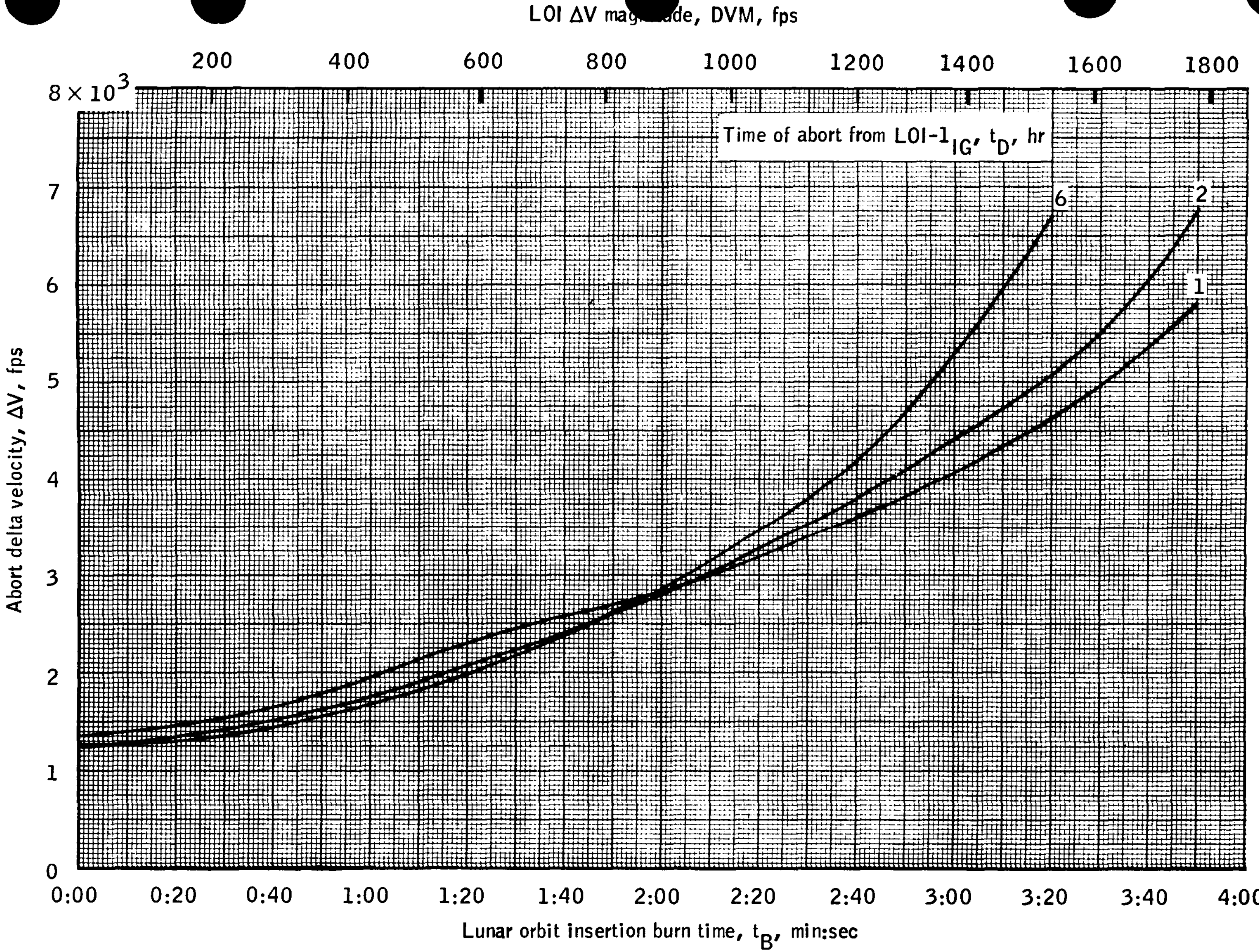
(a) Abort ΔV for GETL = 118 hours.

Figure 8-8.- Mode I MPL returns as a function of LOI burn time.



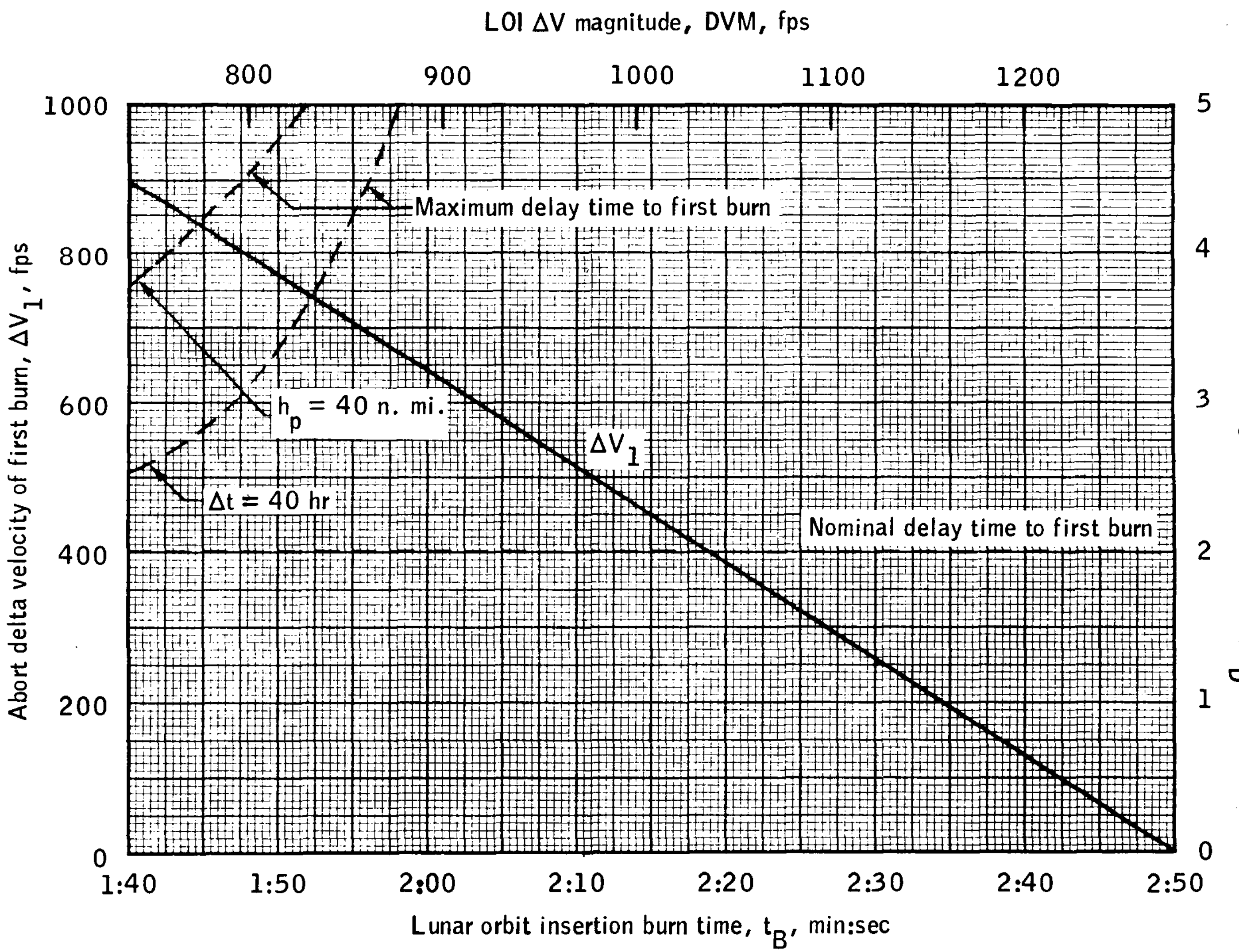
(b) Abort ΔV for GETL = 142 hours.

Figure 8-8.- Continued.



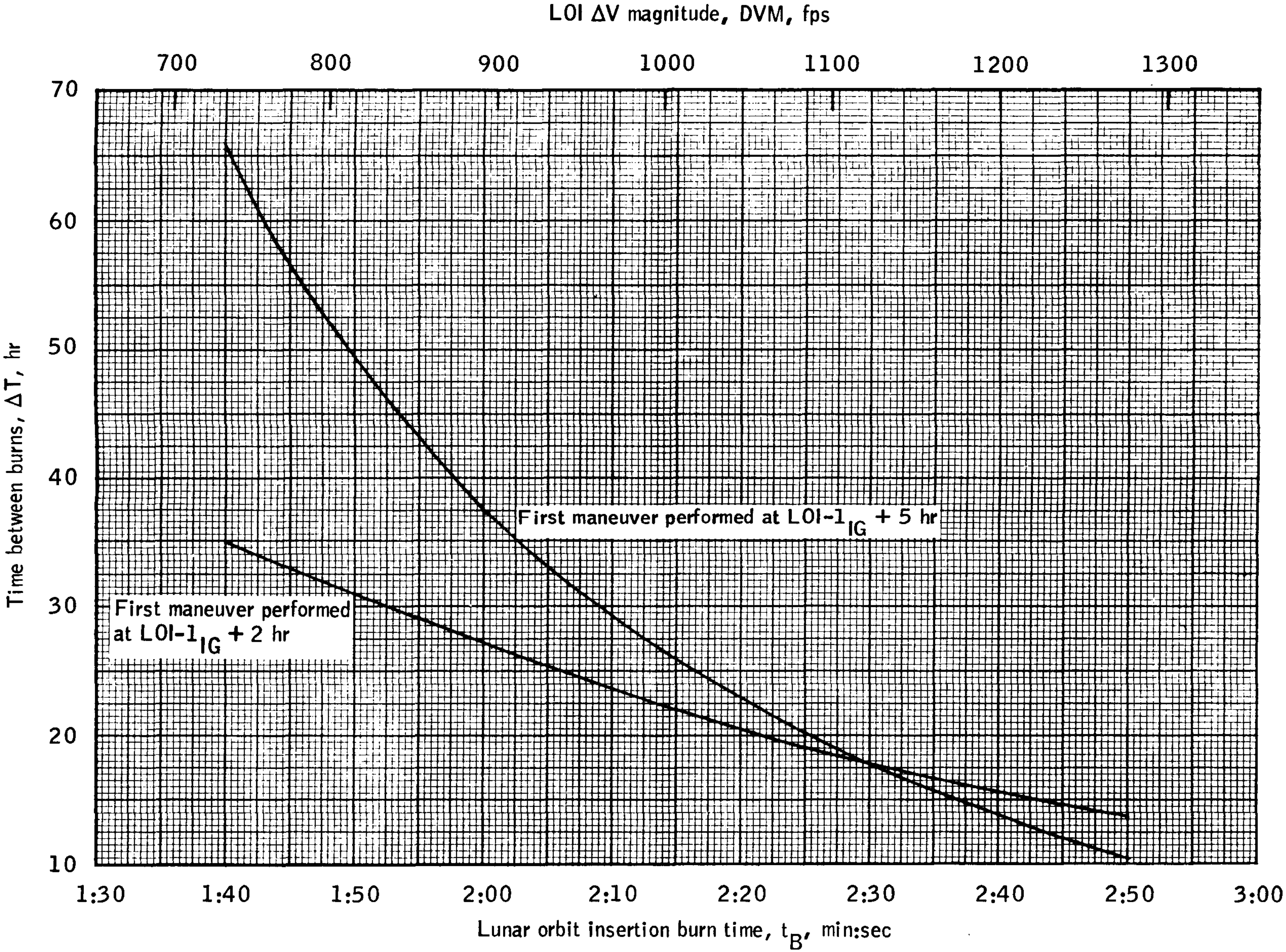
(c) Abort ΔV for GETL = 166 hours.

Figure 8-8.- Concluded.



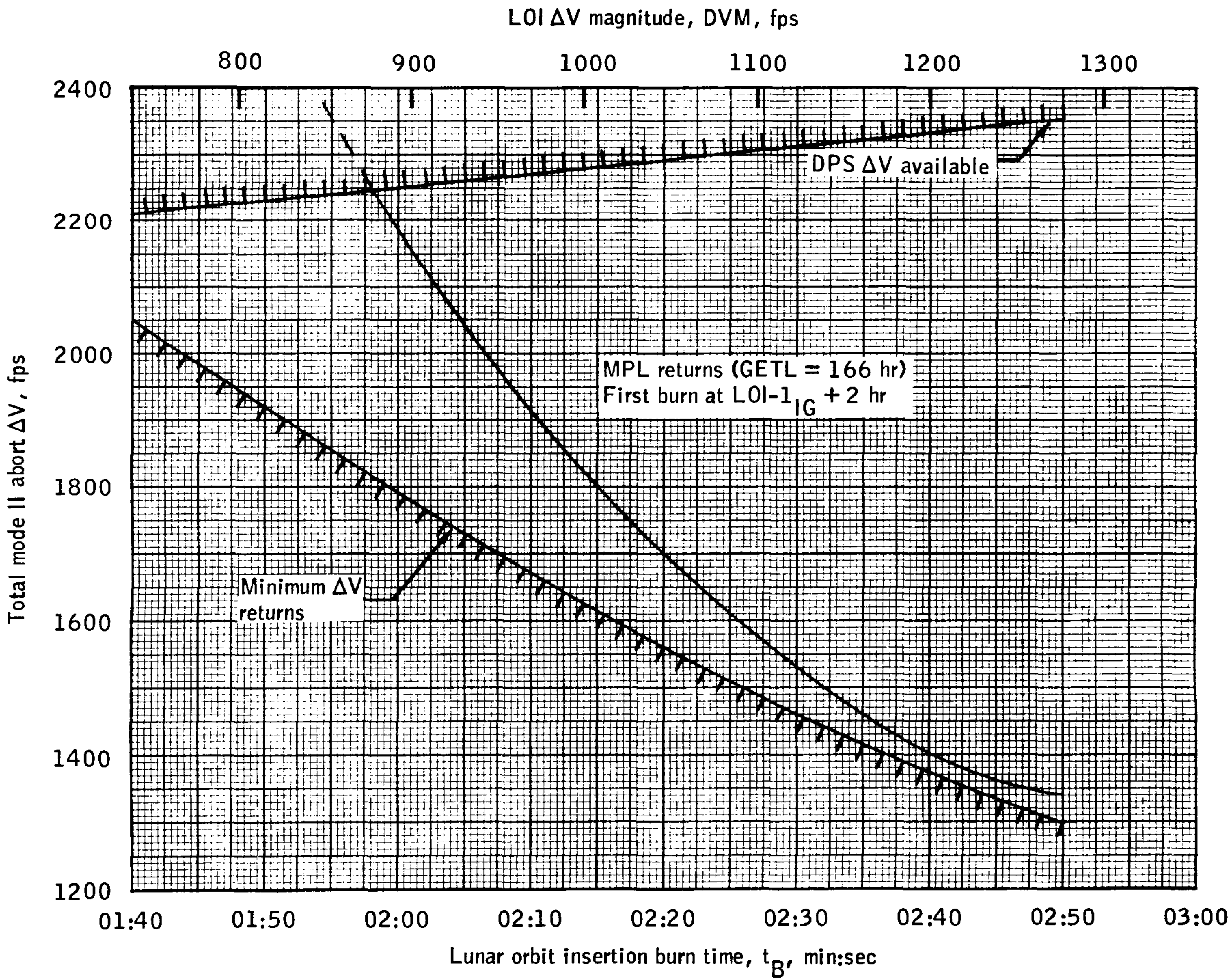
(a) First burn ΔV magnitude and allowable delay time to ignition.

Figure 8-9.- Mode II abort analysis as a function of LOI burn time.



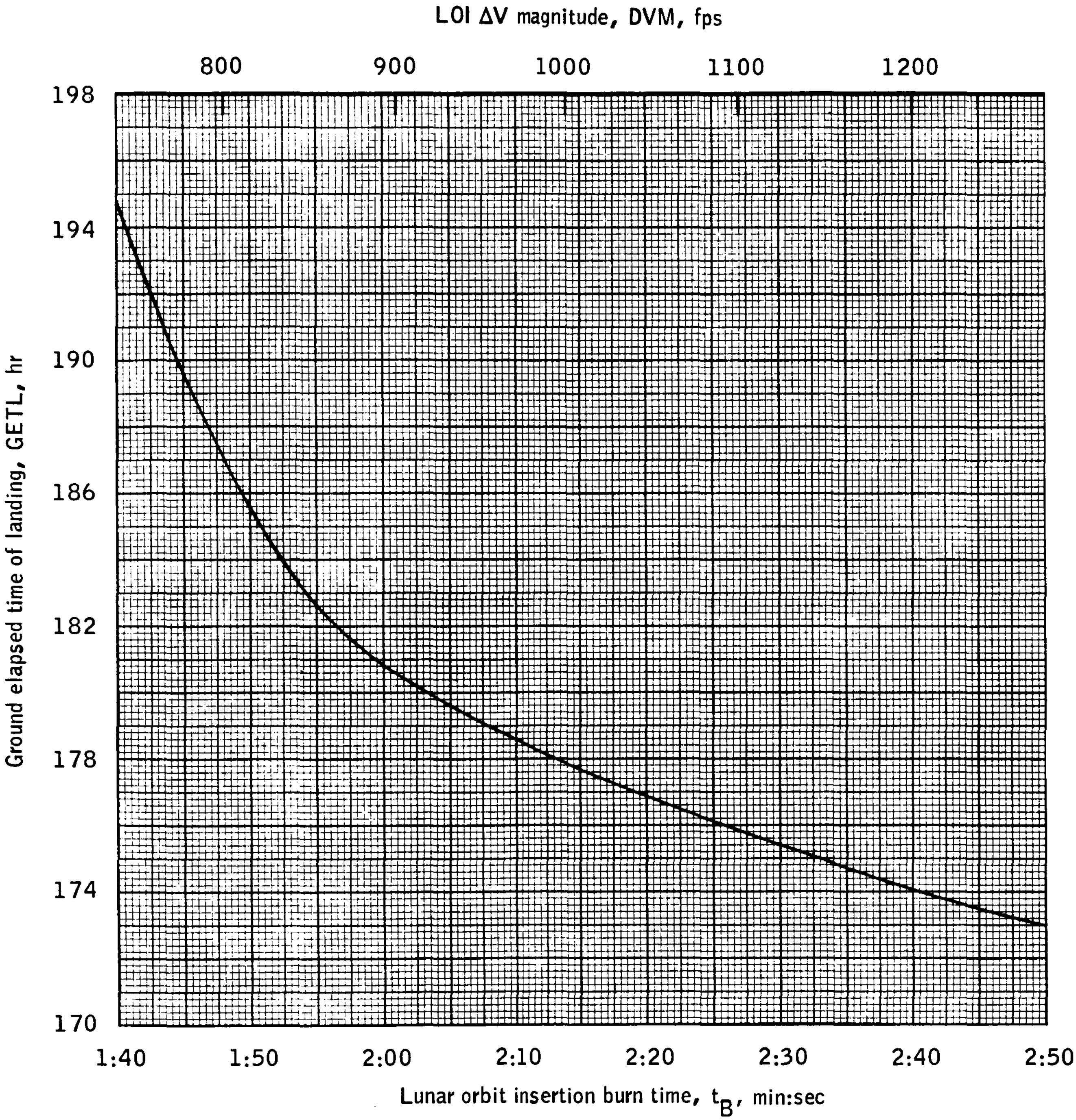
(b) Time between burns following nominal first burn.

Figure 8-9.- Continued.



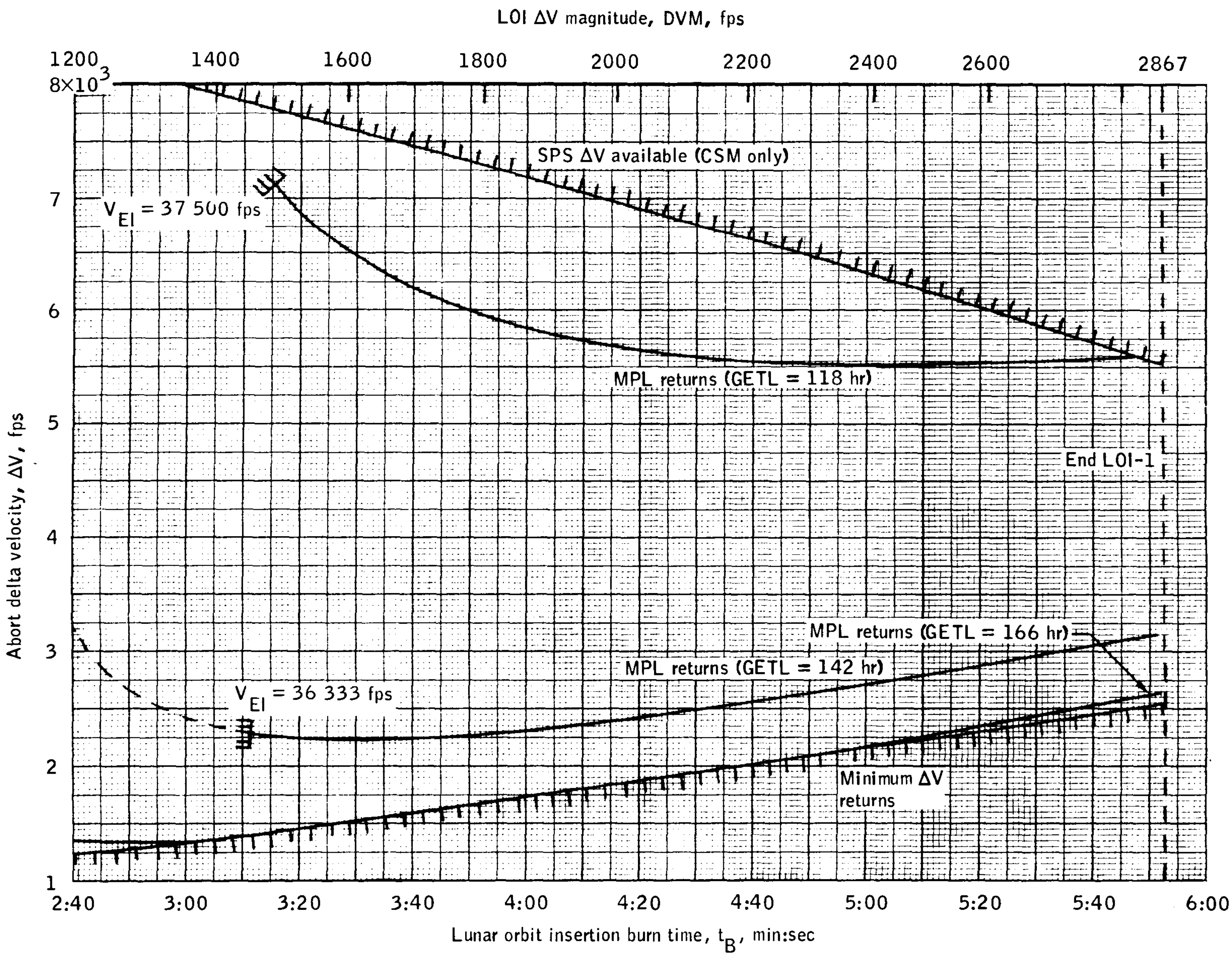
(c) Total abort ΔV required for FCUA and MPL returns $(\Delta V_1 + \Delta V_2)$.

Figure 8-9.- Continued.



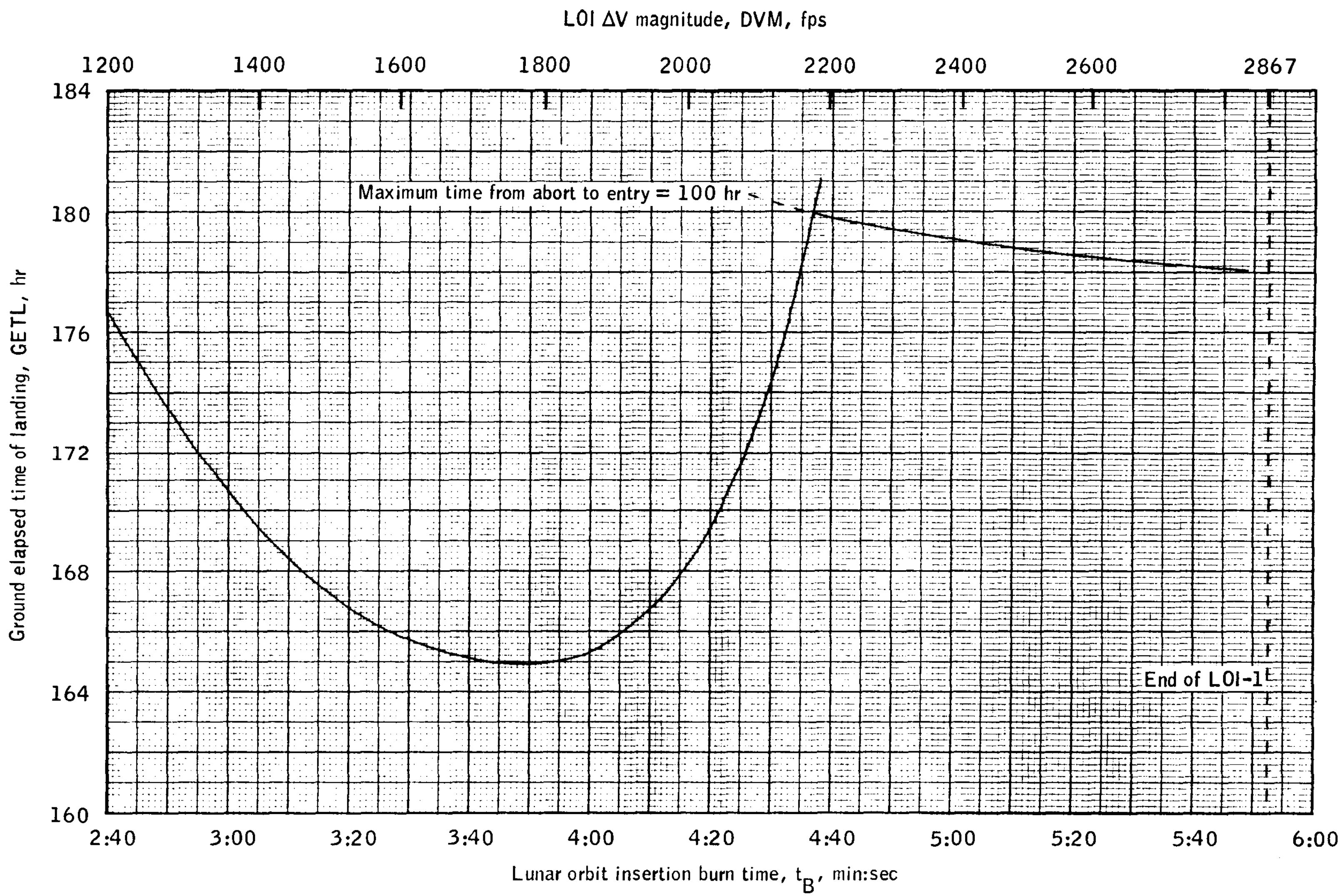
(d) Ground elapsed time of landing for FCUA returns.

Figure 8-9.- Concluded.



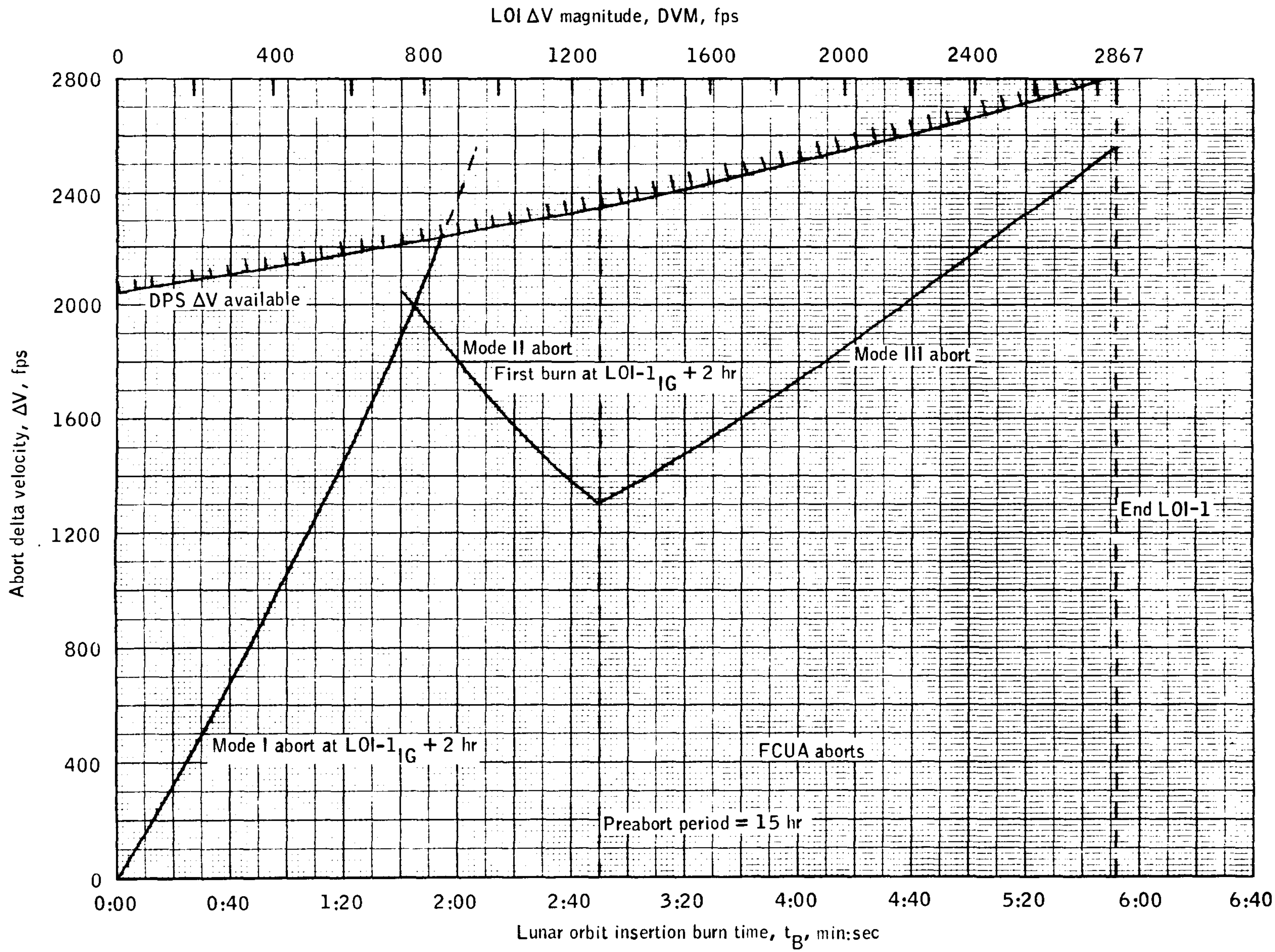
(a) Abort ΔV for FCUA and MPL returns.

Figure 8-10.- Mode III abort analysis as a function of LOI burn time.



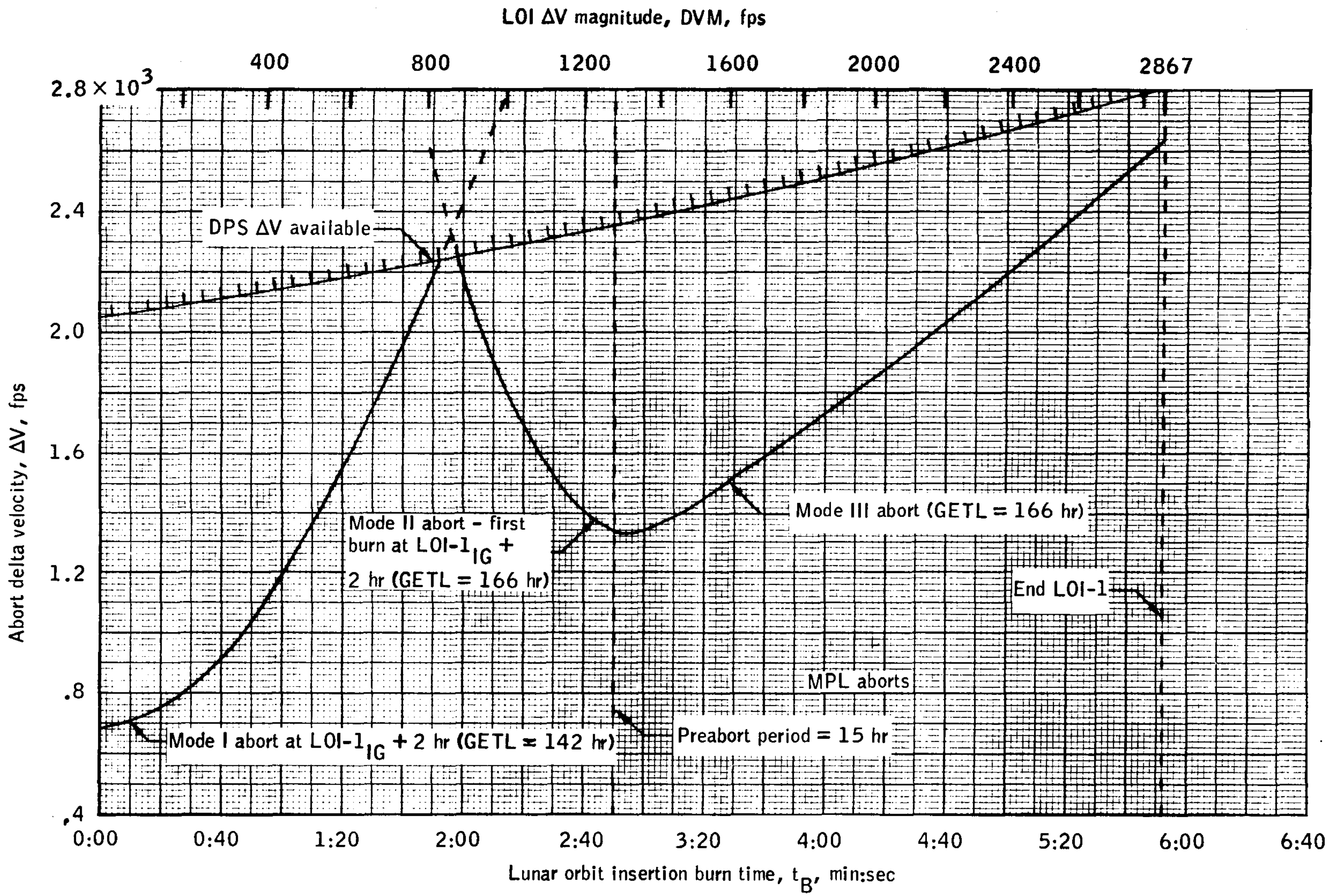
(b) Ground elapsed time of landing for FCUA returns.

Figure 8-10.- Concluded.



(a) Abort ΔV for FCUA returns.

Figure 8-11.- Summary of DPS abort capability as a function of LOI burn time.



(b) Abort ΔV for MPL returns.

Figure 8-11.- Concluded.

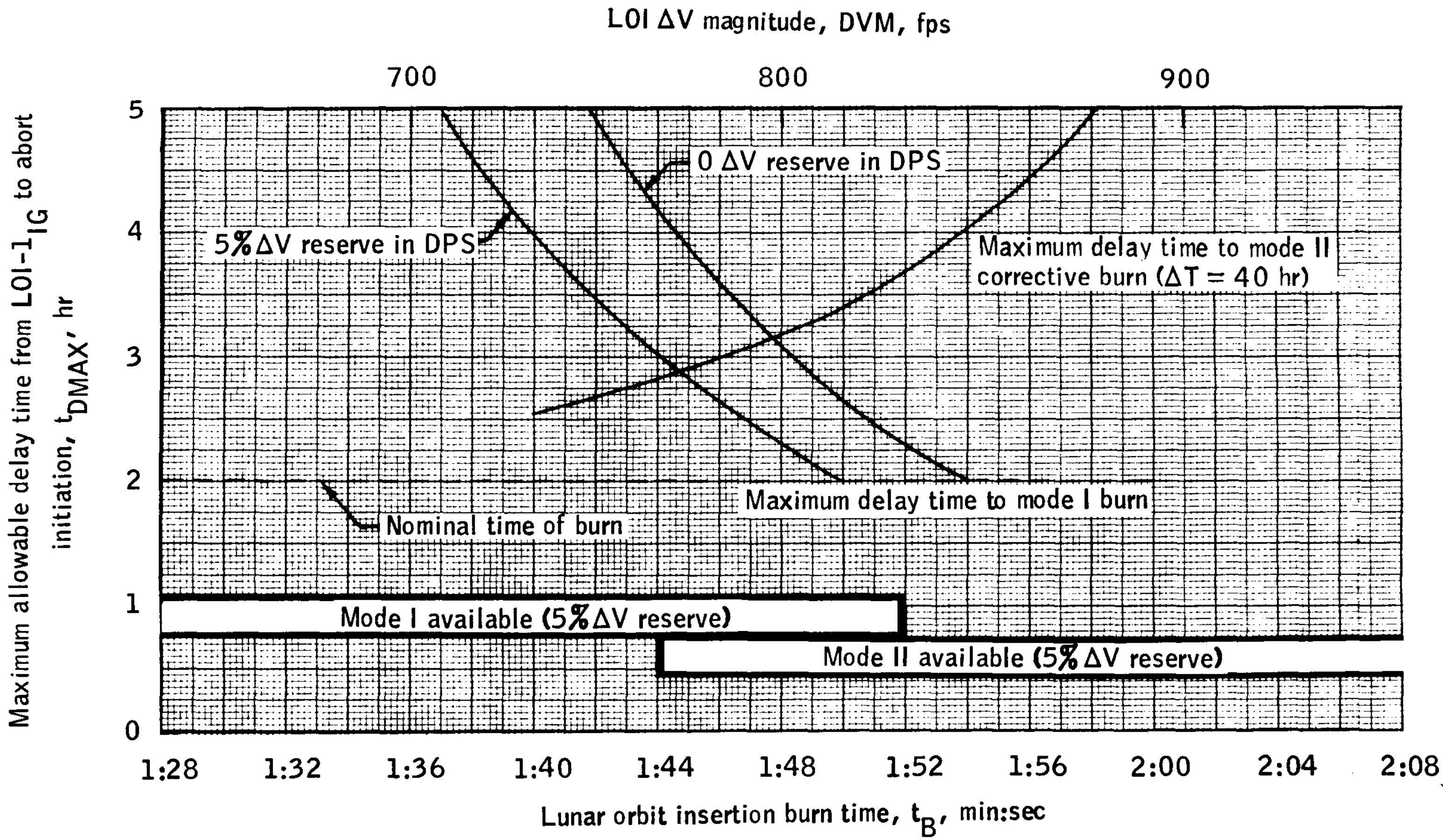


Figure 8-12.- Maximum allowable delay time from $LOI\text{-}1_{IG}$ to abort for premature LOI shutdowns (FCUA returns).

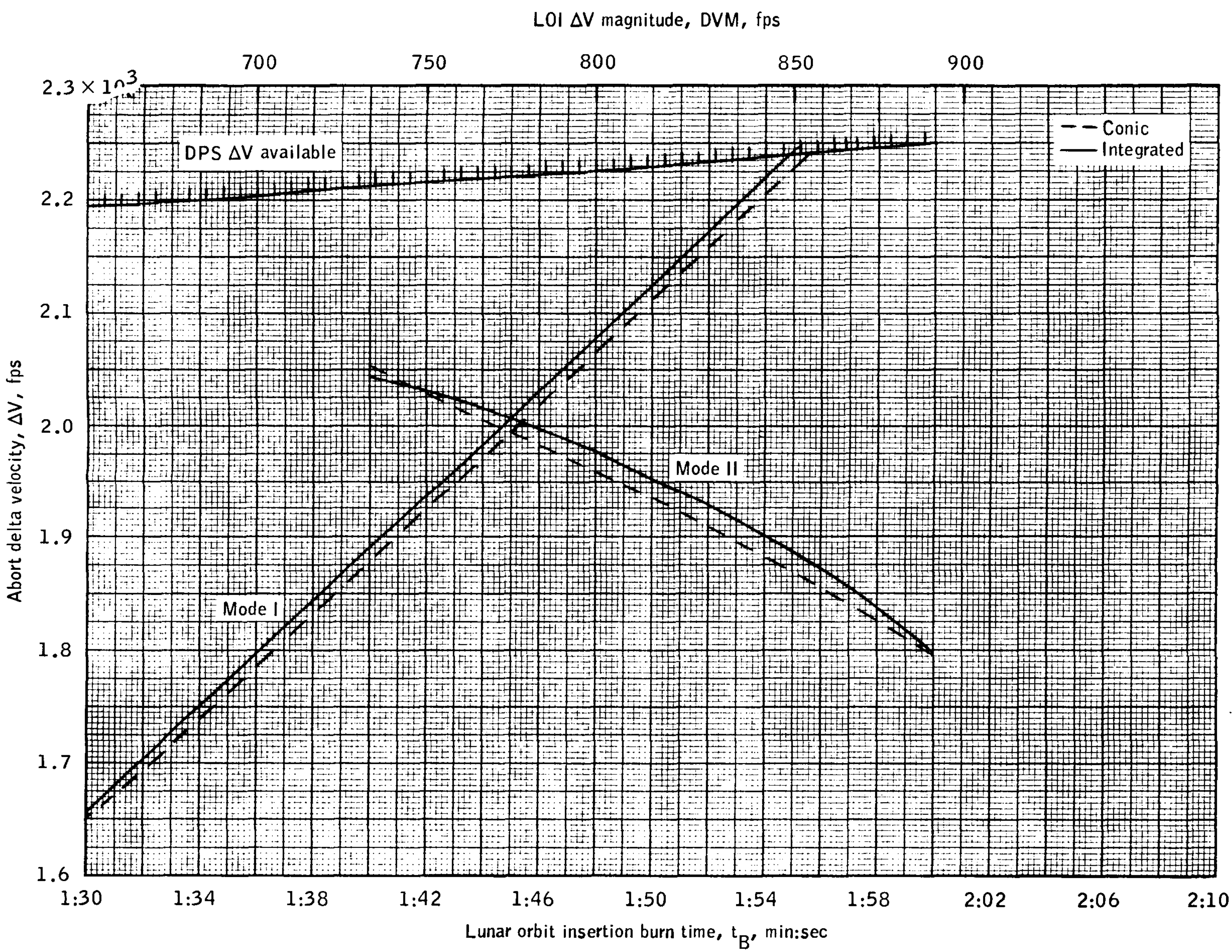


Figure 8-13.- Comparison of patched conic and integrated transearth coast on abort ΔV requirements in mode I/mode II overlap region (impulsive burns) forFCUA returns.

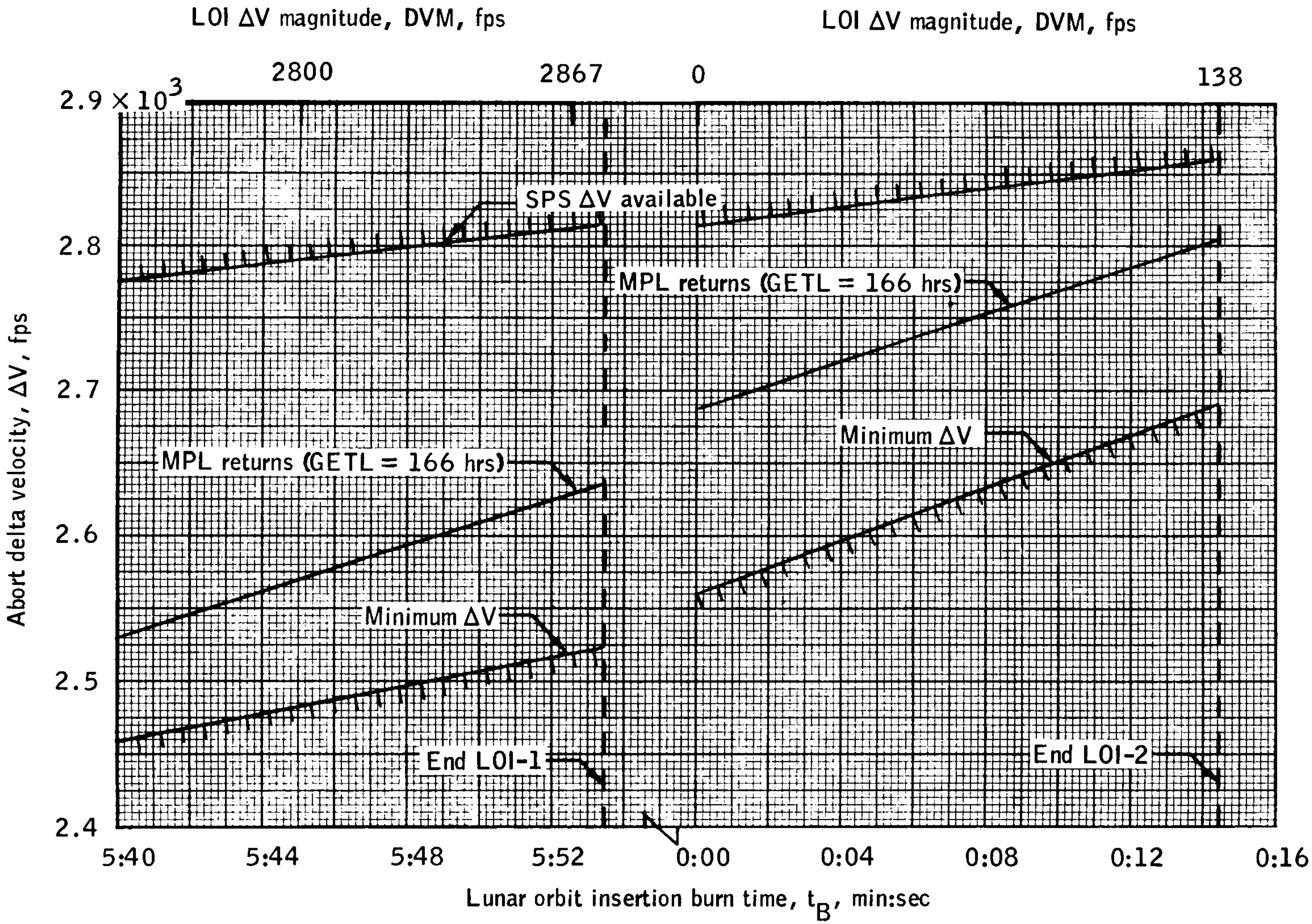


Figure 8-14.- Mode III abort requirements using integrated transearth coasts (impulsive burns) for FCUA returns.

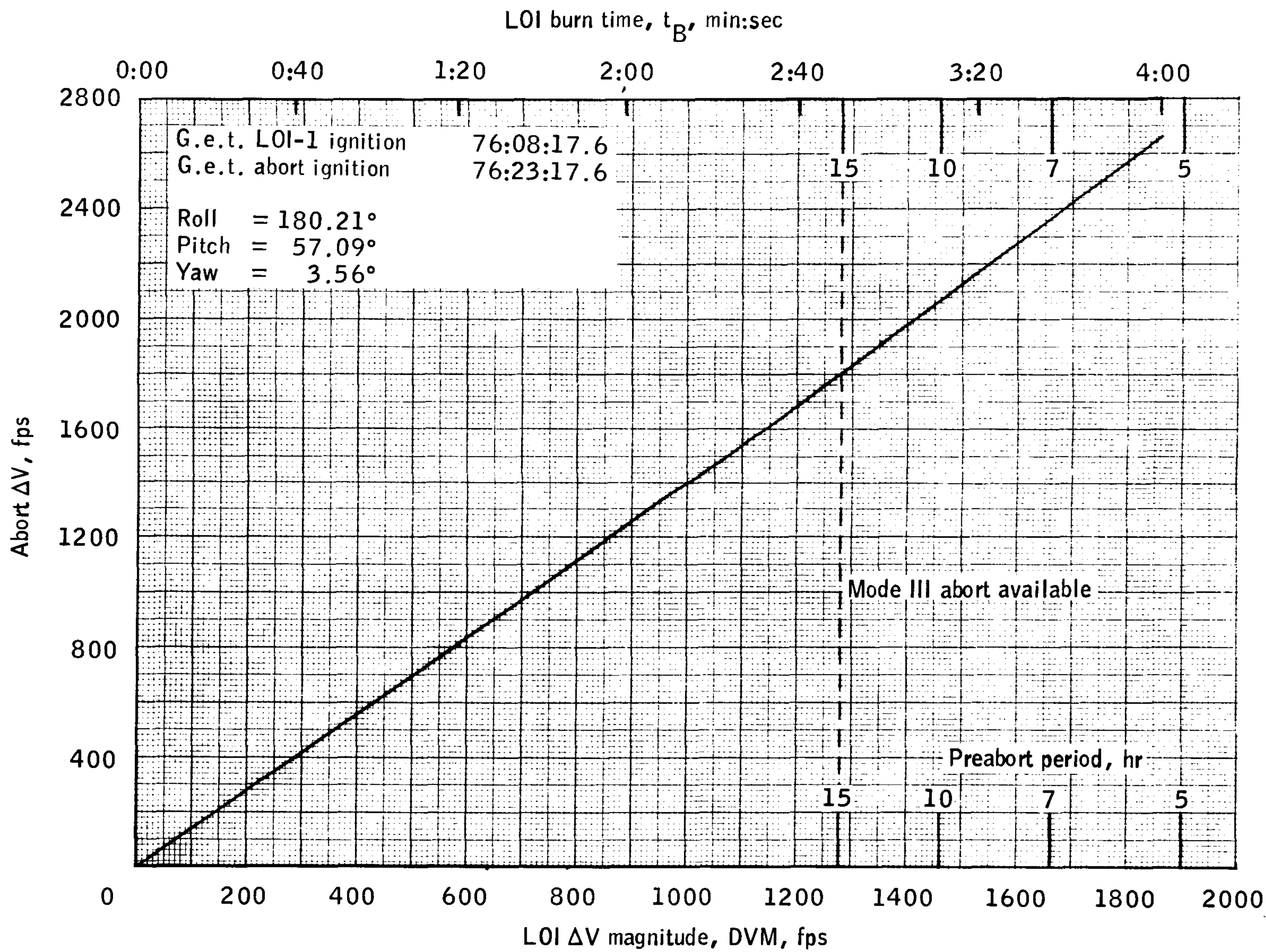
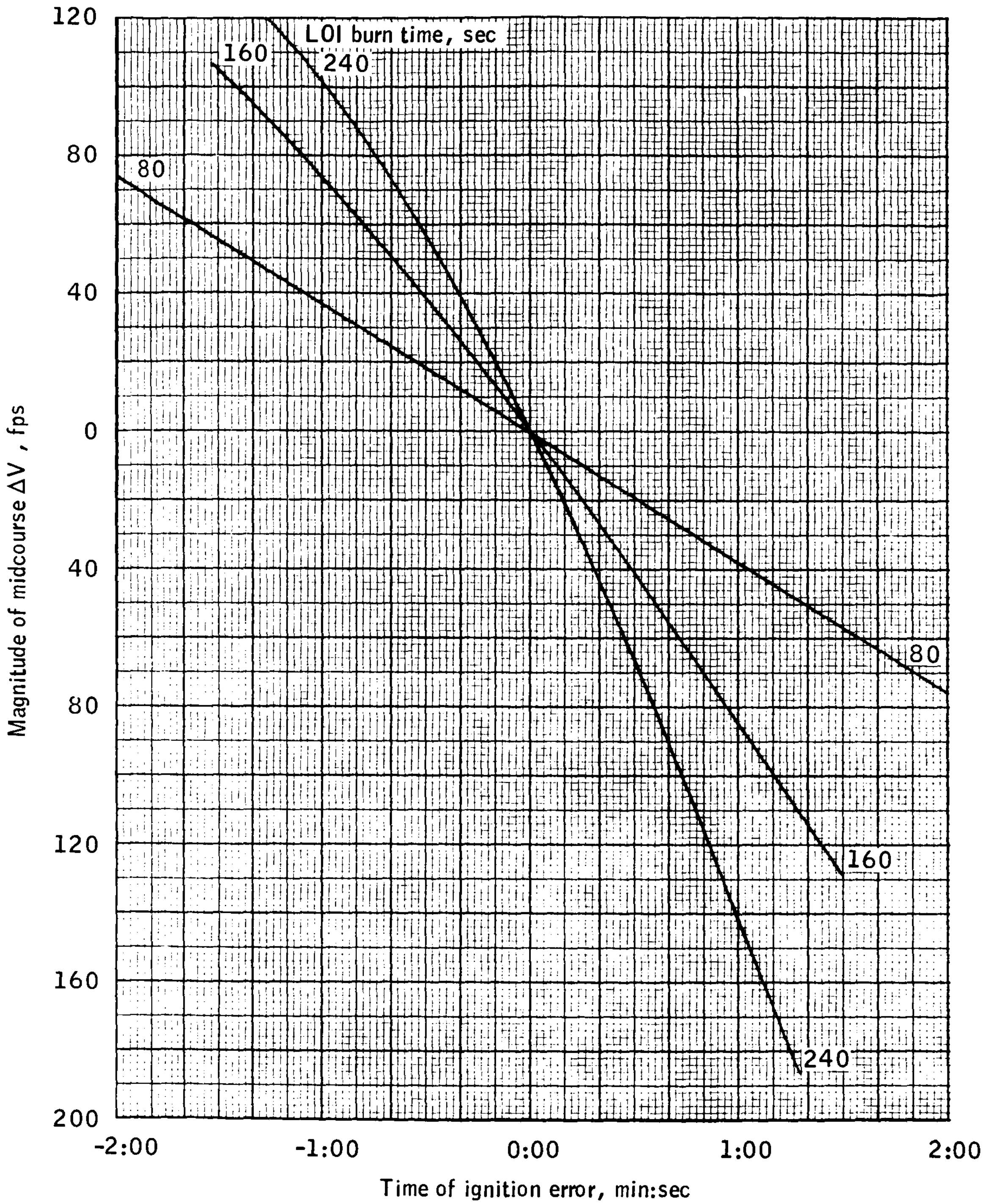
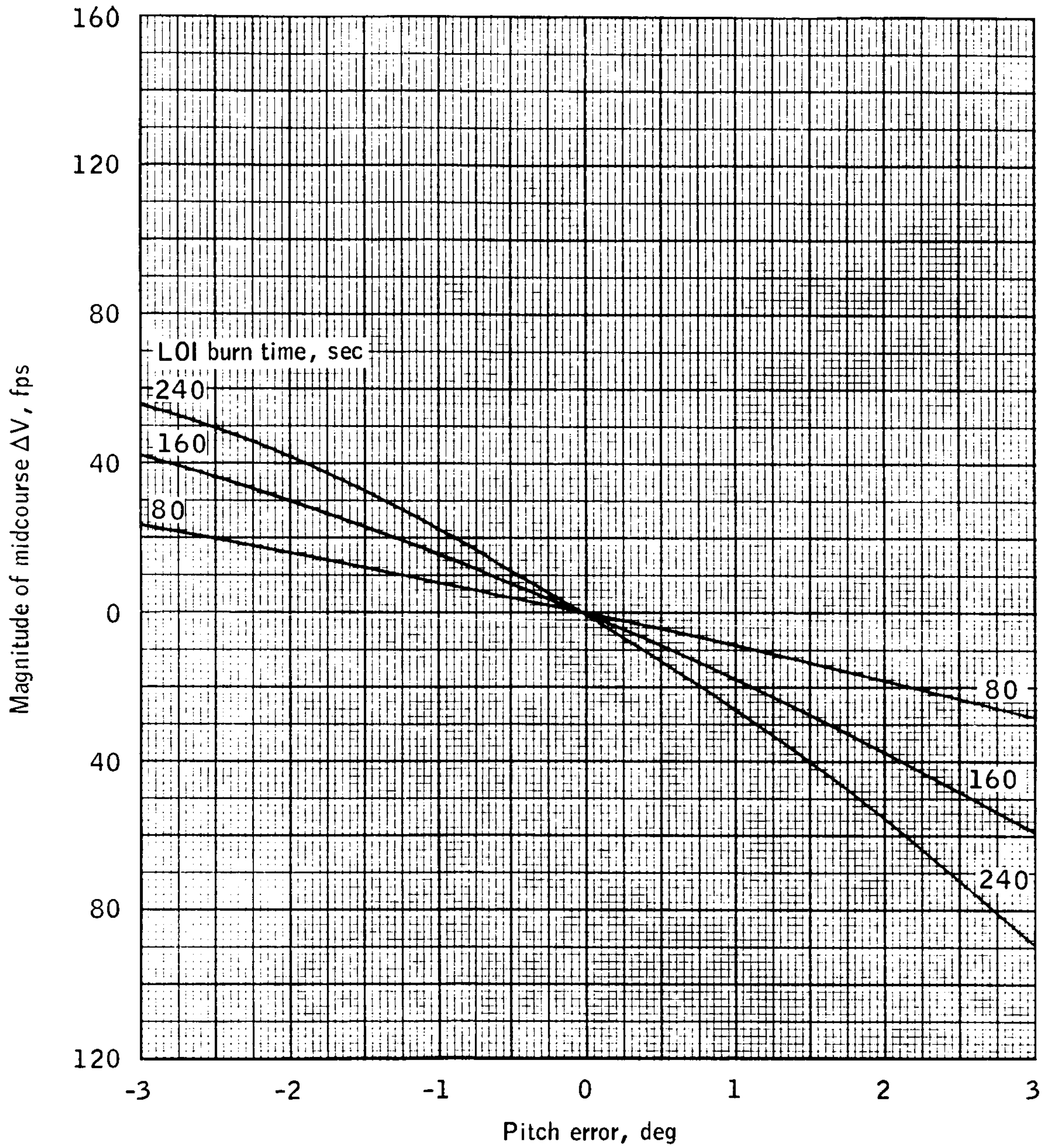


Figure 8-15.- Nominal Mission F crew chart for abort at LOI-1_{IG} + 15 minutes.



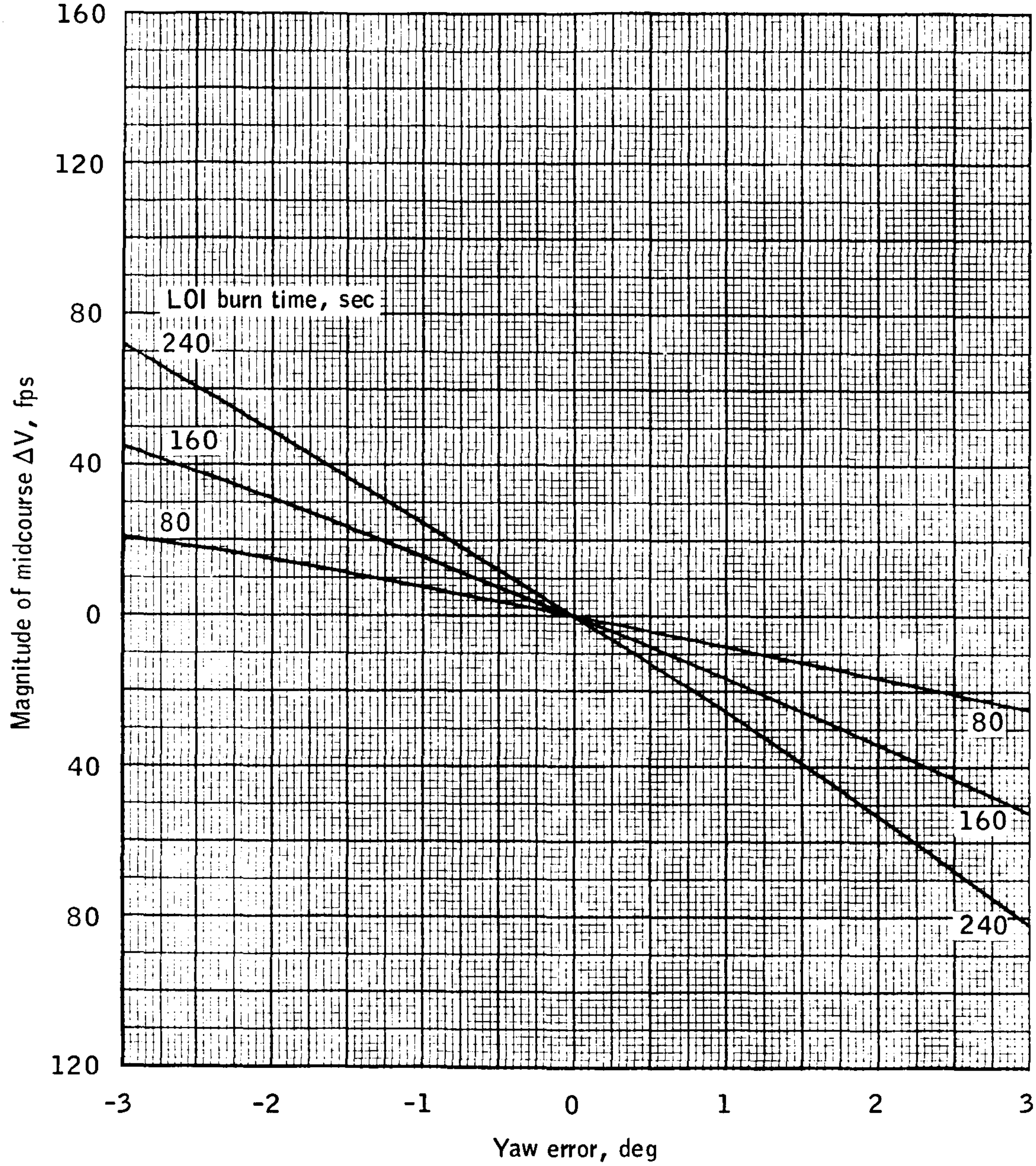
(a) MCC ΔV at MSI for ignition time errors.

Figure 8-16.- Model I (15 minute) crew-chart midcourse ΔV requirements.



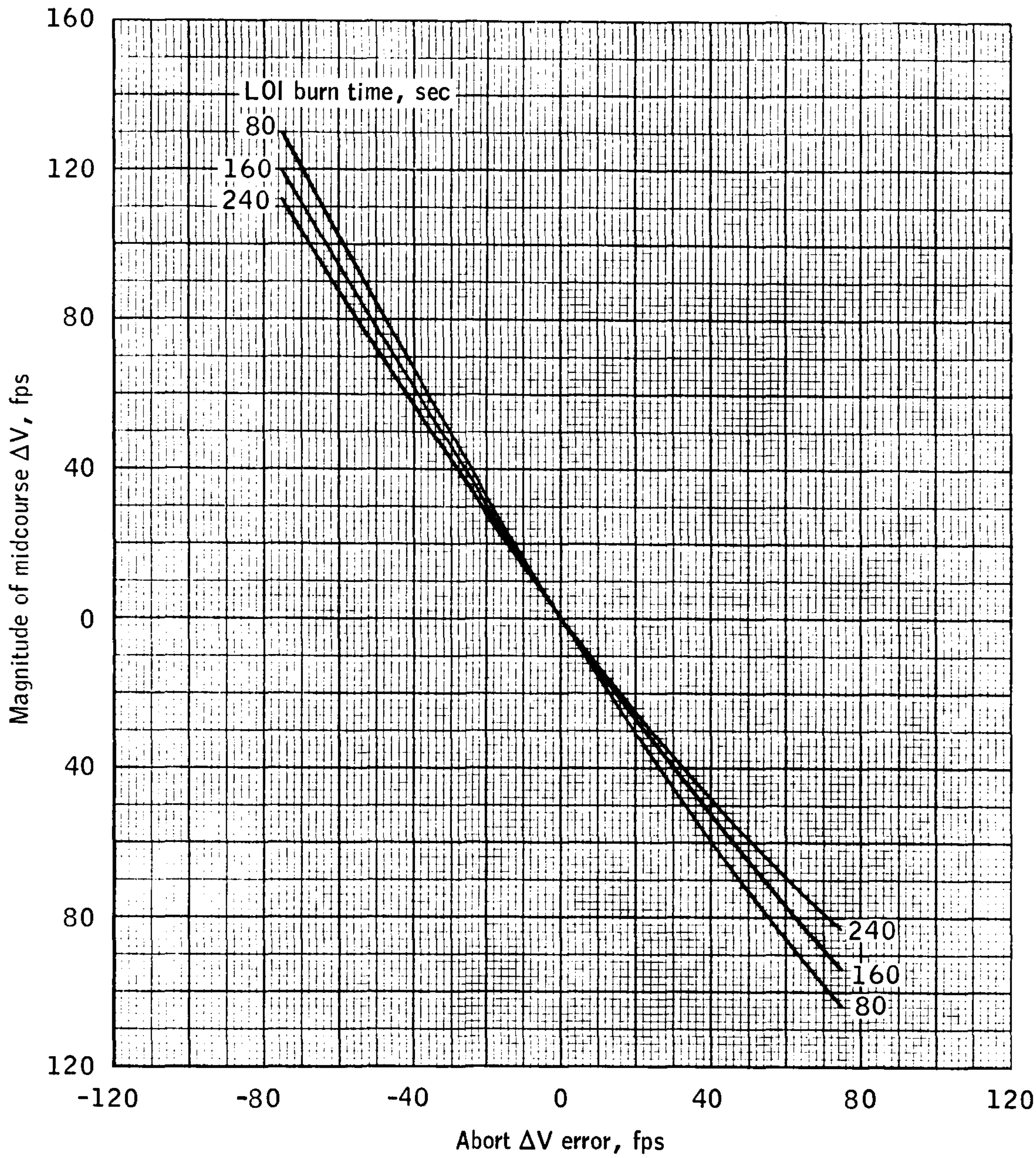
(b) MCC ΔV at MSI for pitch errors.

Figure 8-16.- Continued.



(c) MCC ΔV at MSI for yaw errors.

Figure 8-16.- Continued.



(d) MCC ΔV at MSI for abort ΔV errors.

Figure 8-16.- Concluded.

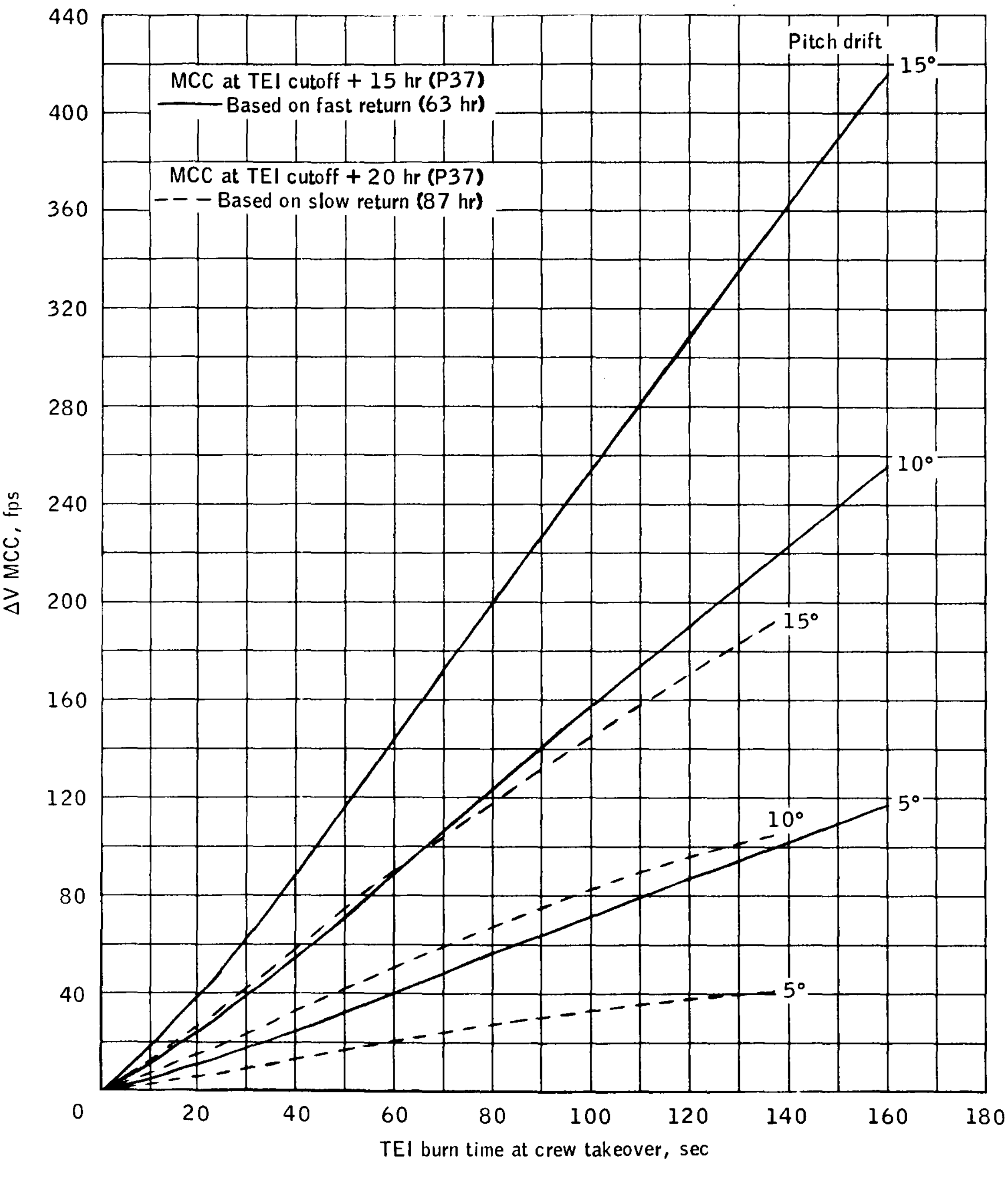


Figure 9-1.- ΔV for midcourse corrections as a function of TEI burn time at crew takeover for various pitch drifts.

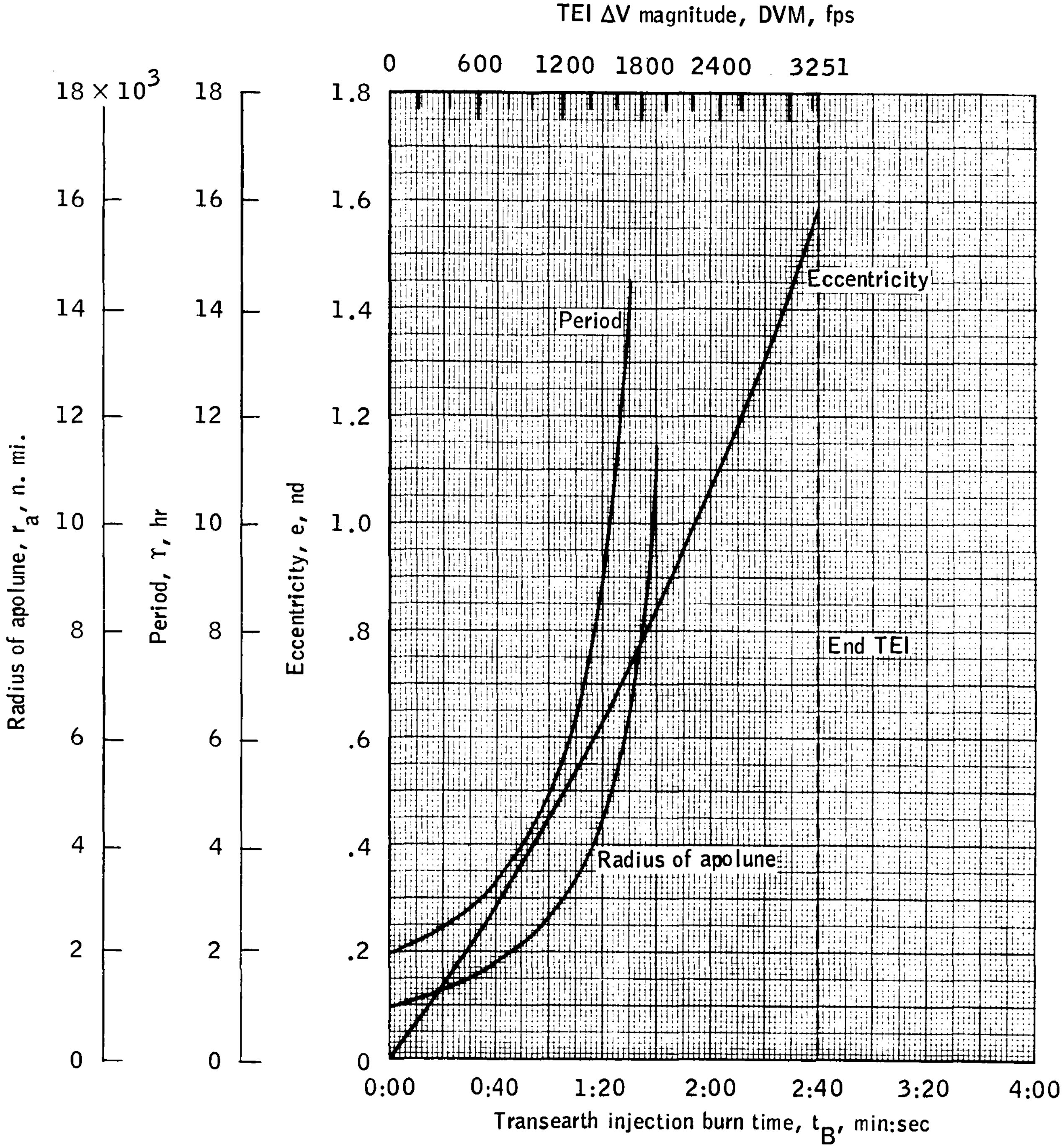


Figure 9-2.- Conic parameters as a function of SPS burn time during the TEI burn.

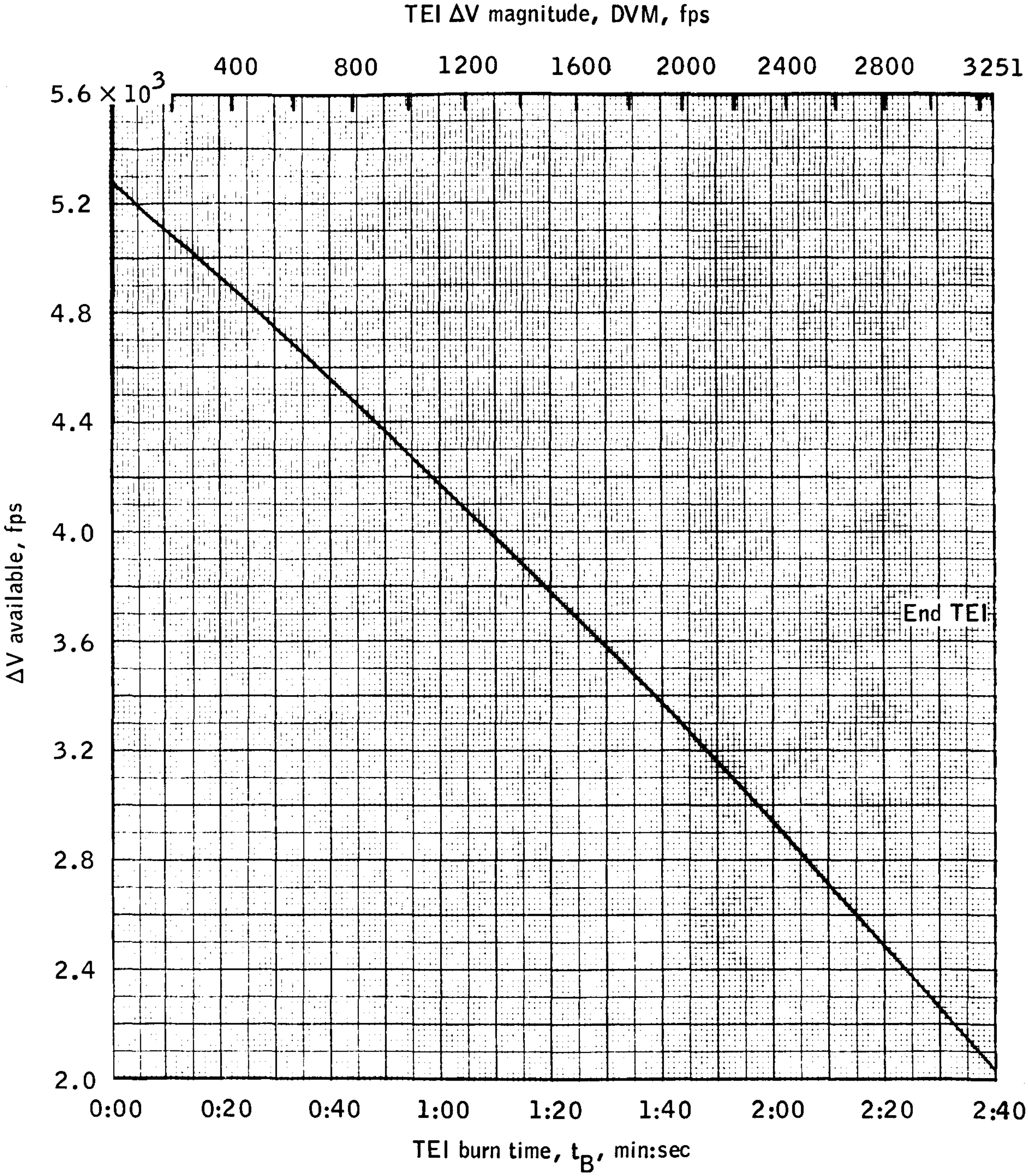
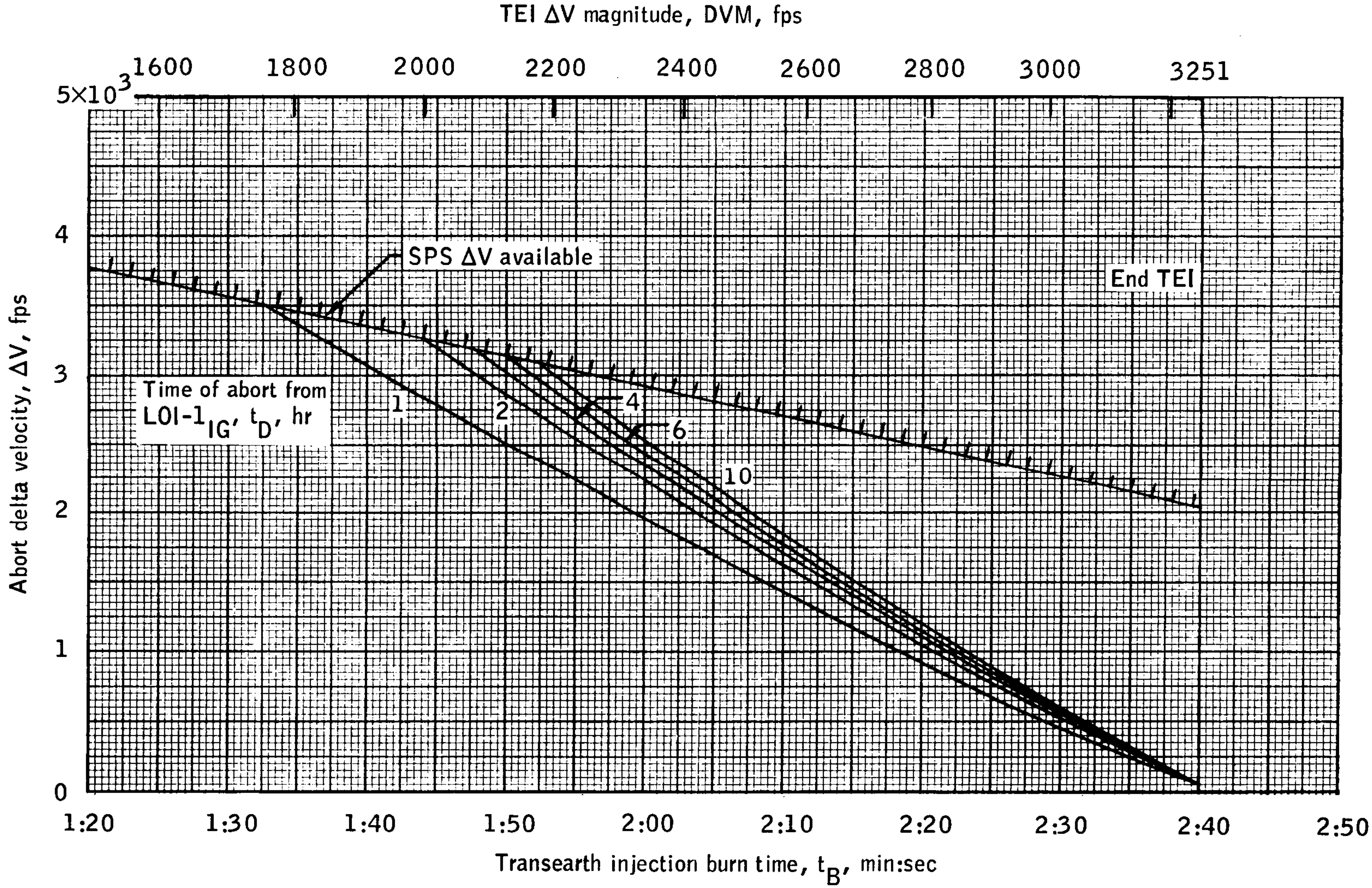
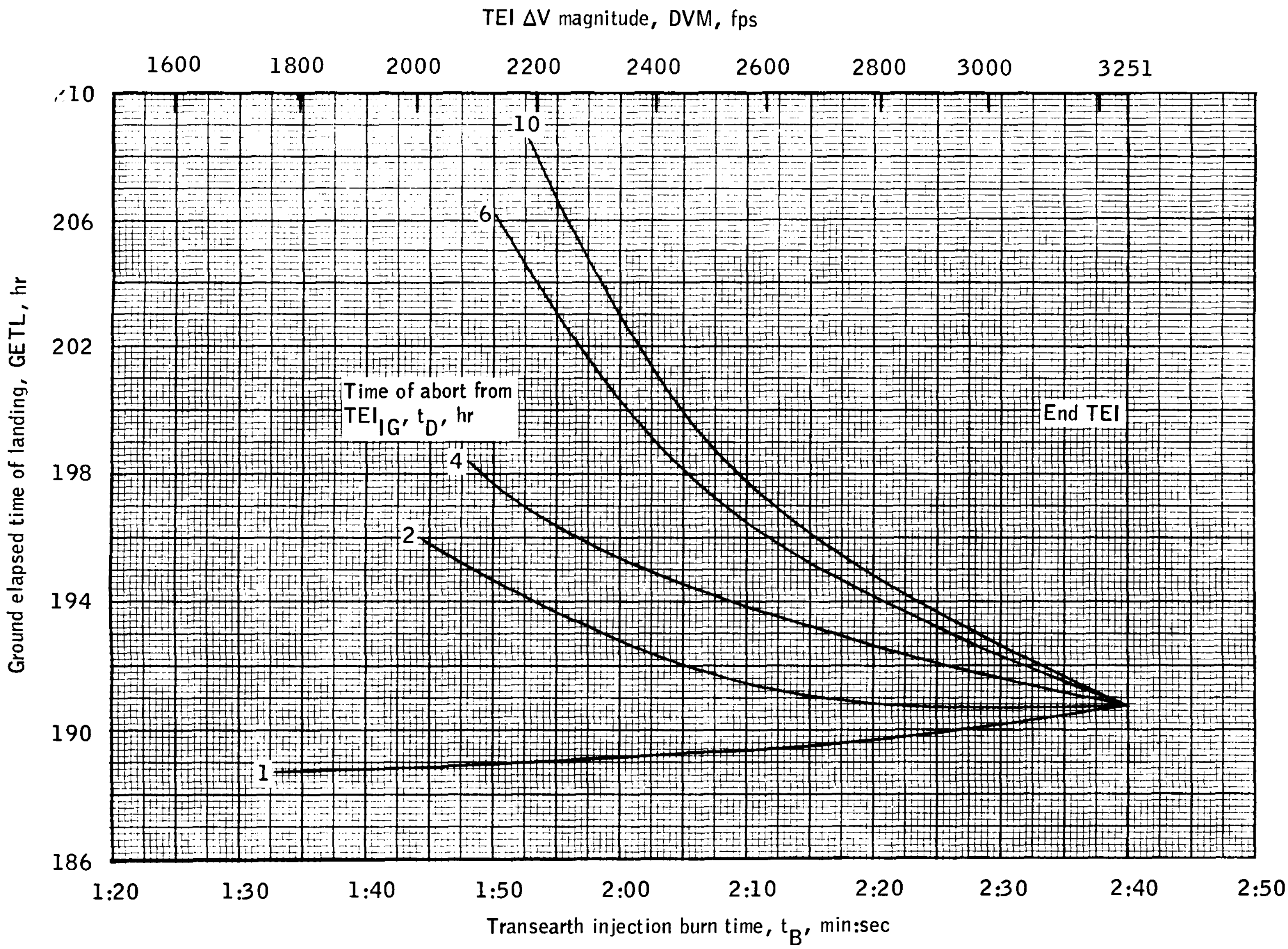


Figure 9-3.- Abort ΔV available following a premature SPS shutdown during TEI burn.



(a) Abort ΔV required for various times of ignition.

Figure 9-4.- Mode I unspecified area abort analysis as a function of TEI burn time.



(b) Ground elapsed time of landing for various times of ignition.

Figure 9-4.- Concluded.

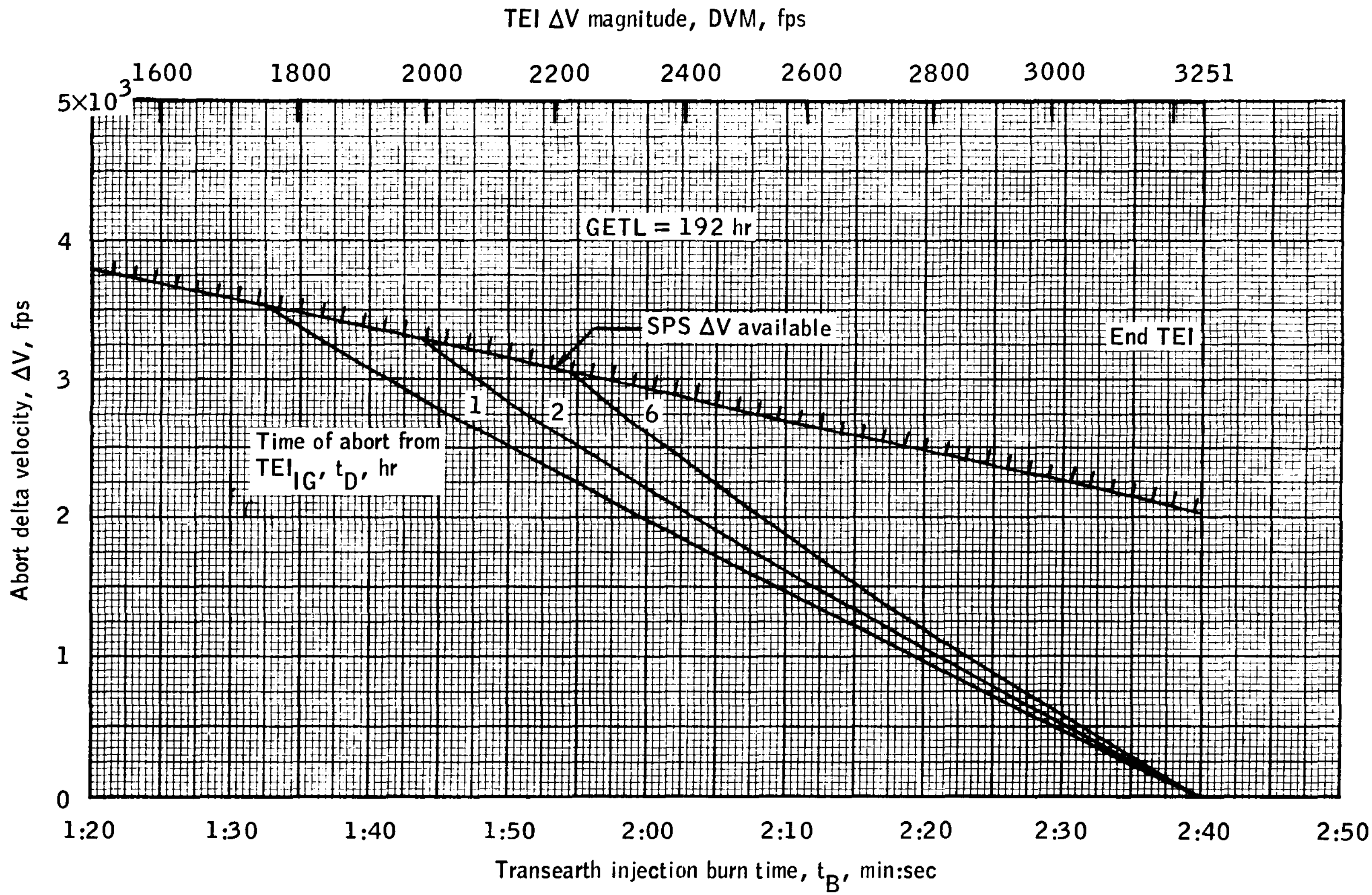
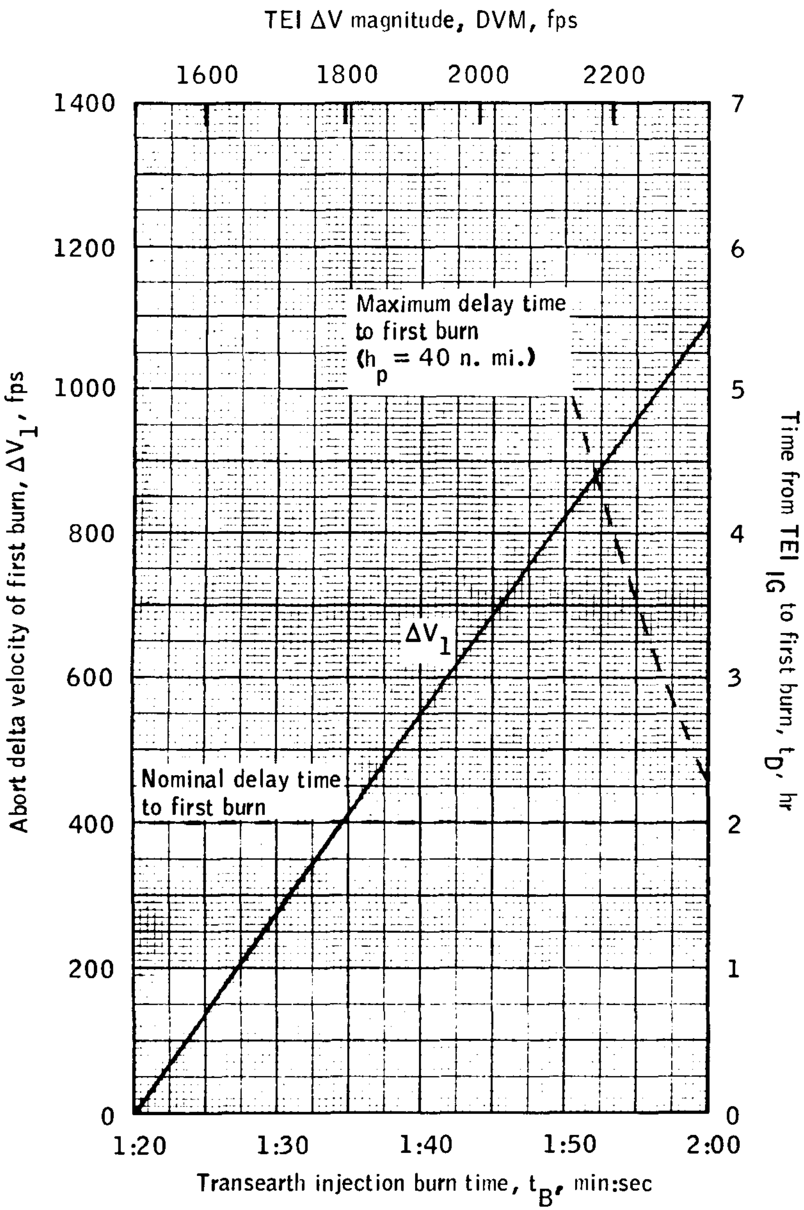
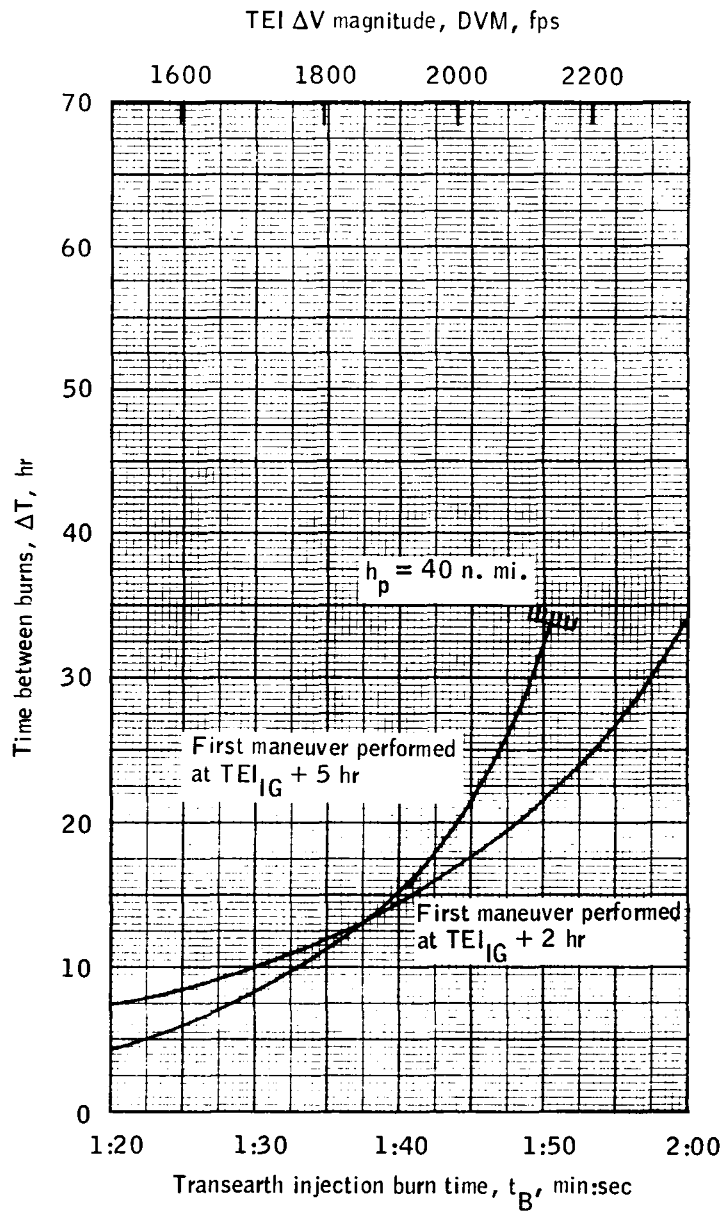


Figure 9-5.- Mode I MPL returns as a function of TEI burn time.

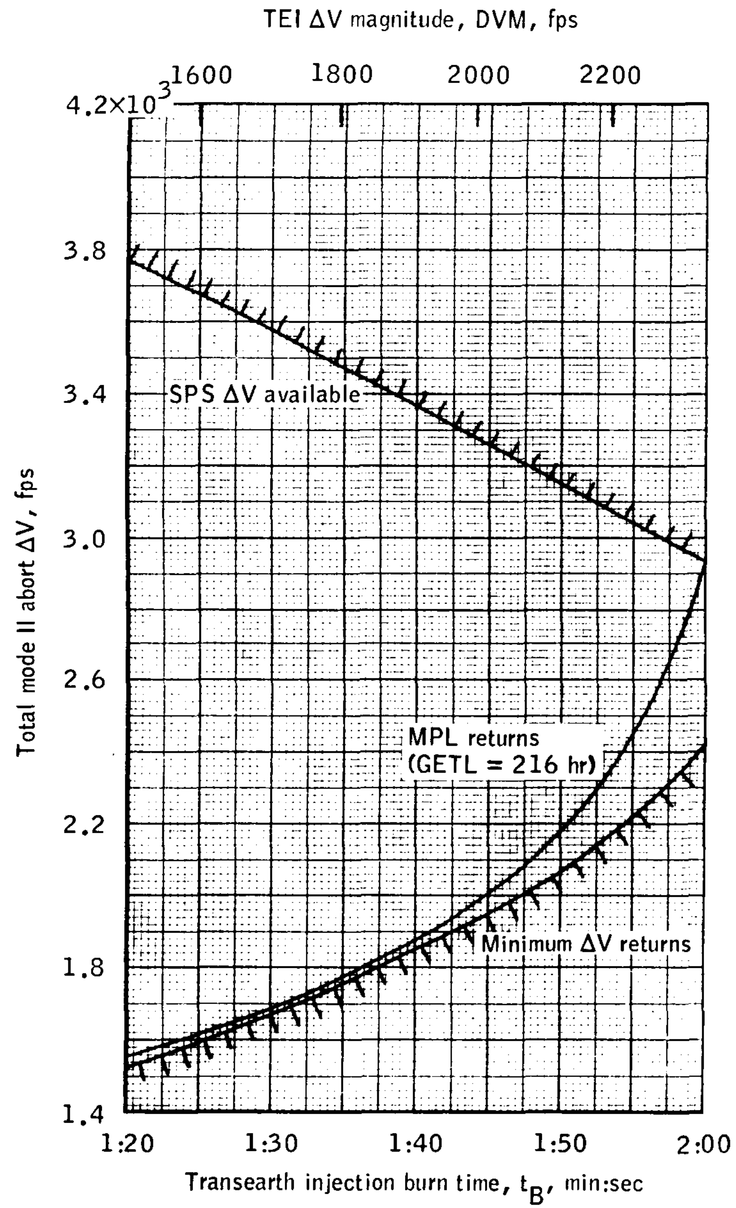


(a) First burn ΔV magnitude and allowable delay time to ignition.

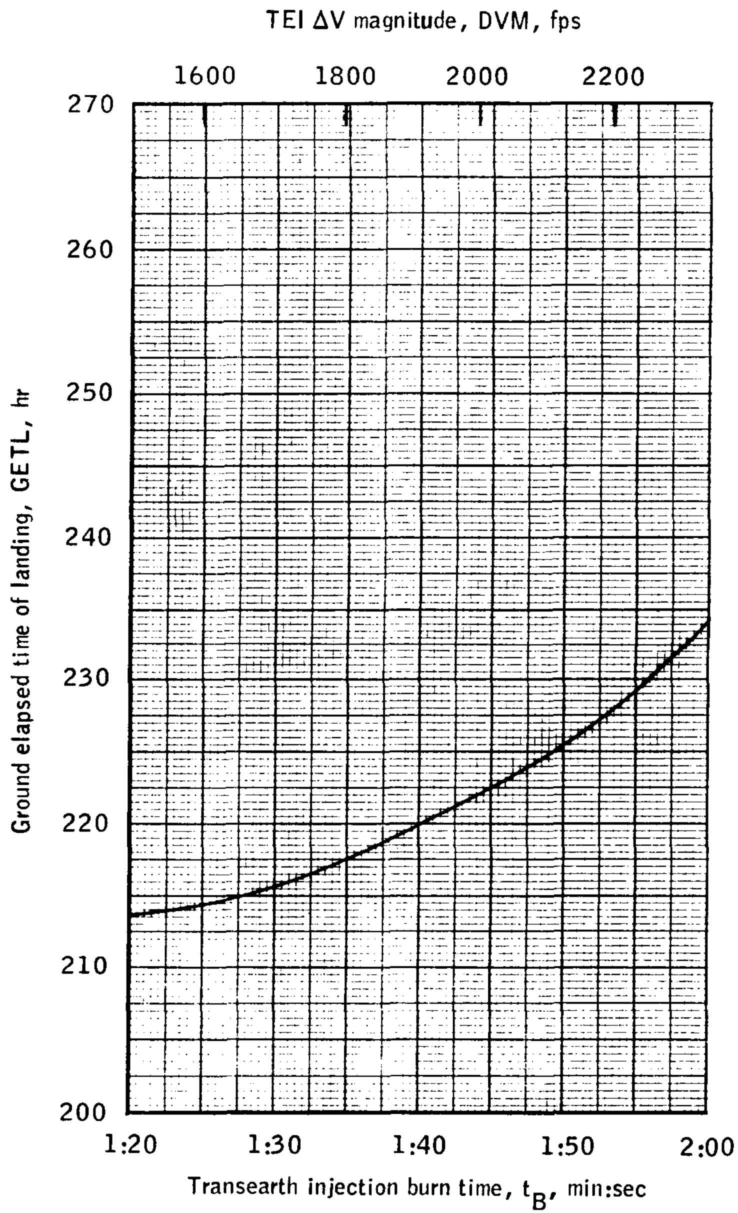


(b) Time between burns following nominal first burn.

Figure 9-6.- Mode II abort analysis as a function of TEI burn time.

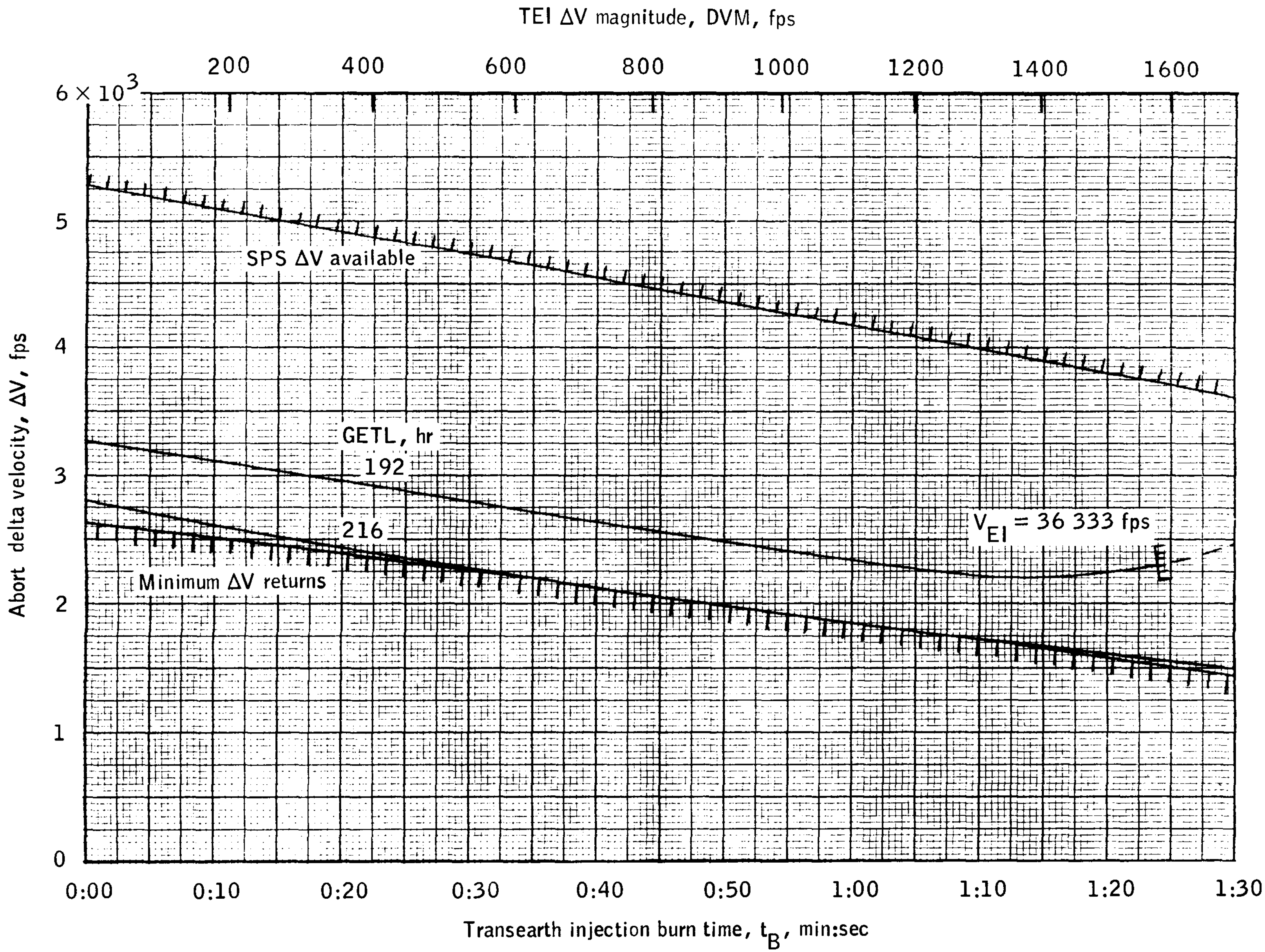


(c) Total abort ΔV required for FCUA and MPL returns ($\Delta V_1 + \Delta V_2$).



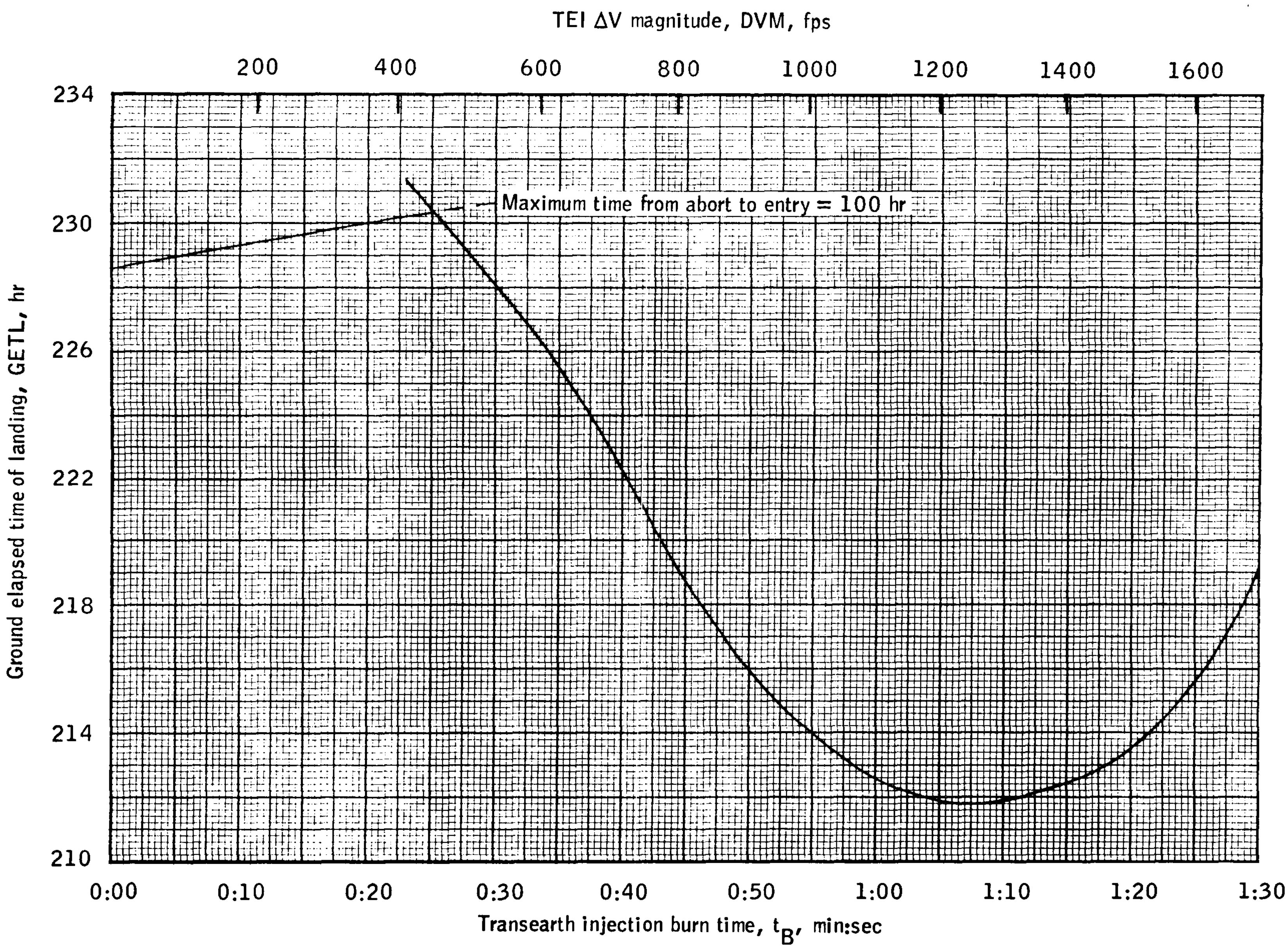
(d) Ground elapsed time of landing for FCUA returns.

Figure 9-6.- Concluded.



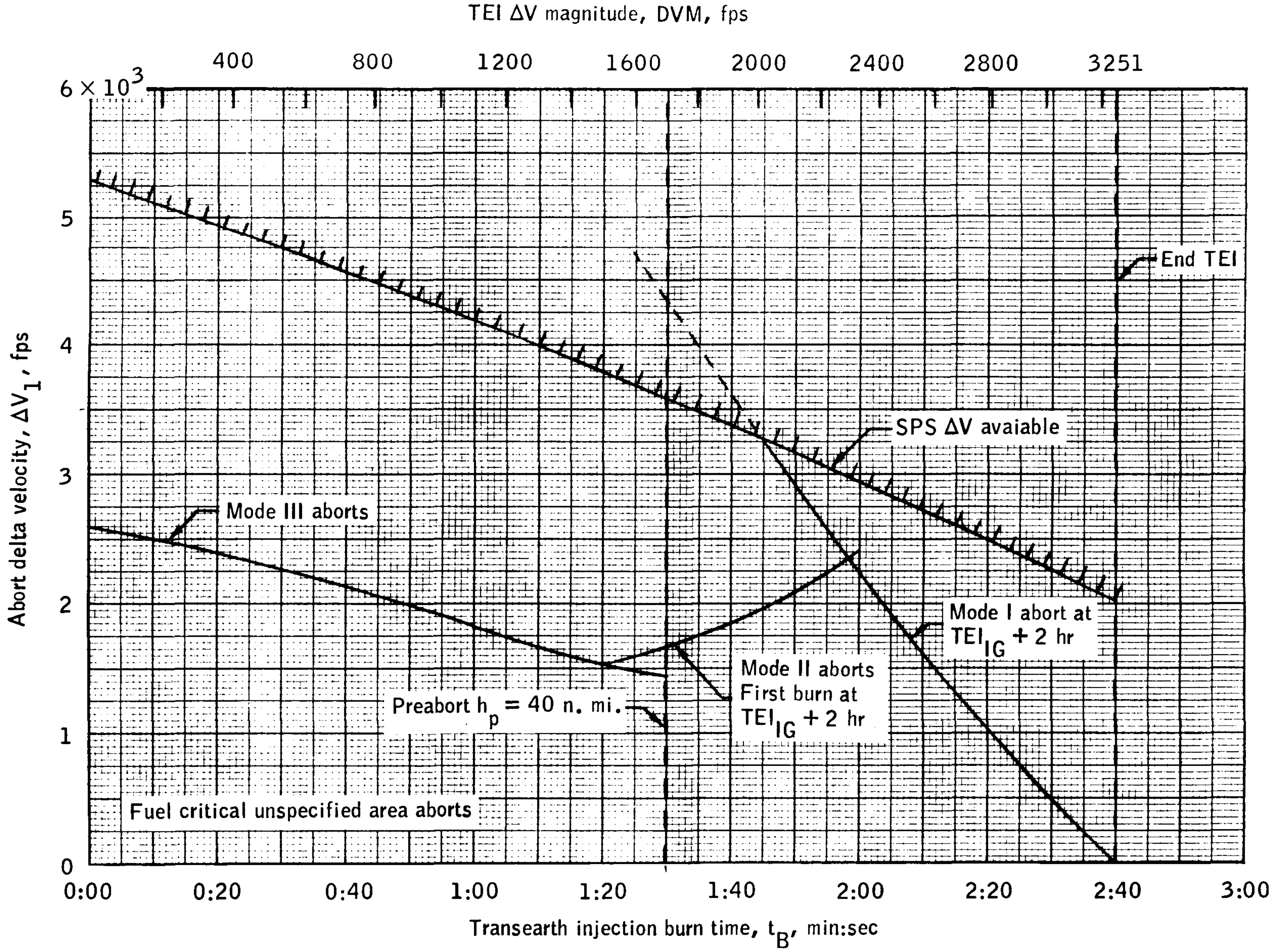
(a) Abort ΔV for FCUA and MPL returns.

Figure 9-7.- Mode III abort analysis as a function of TEI burn time.



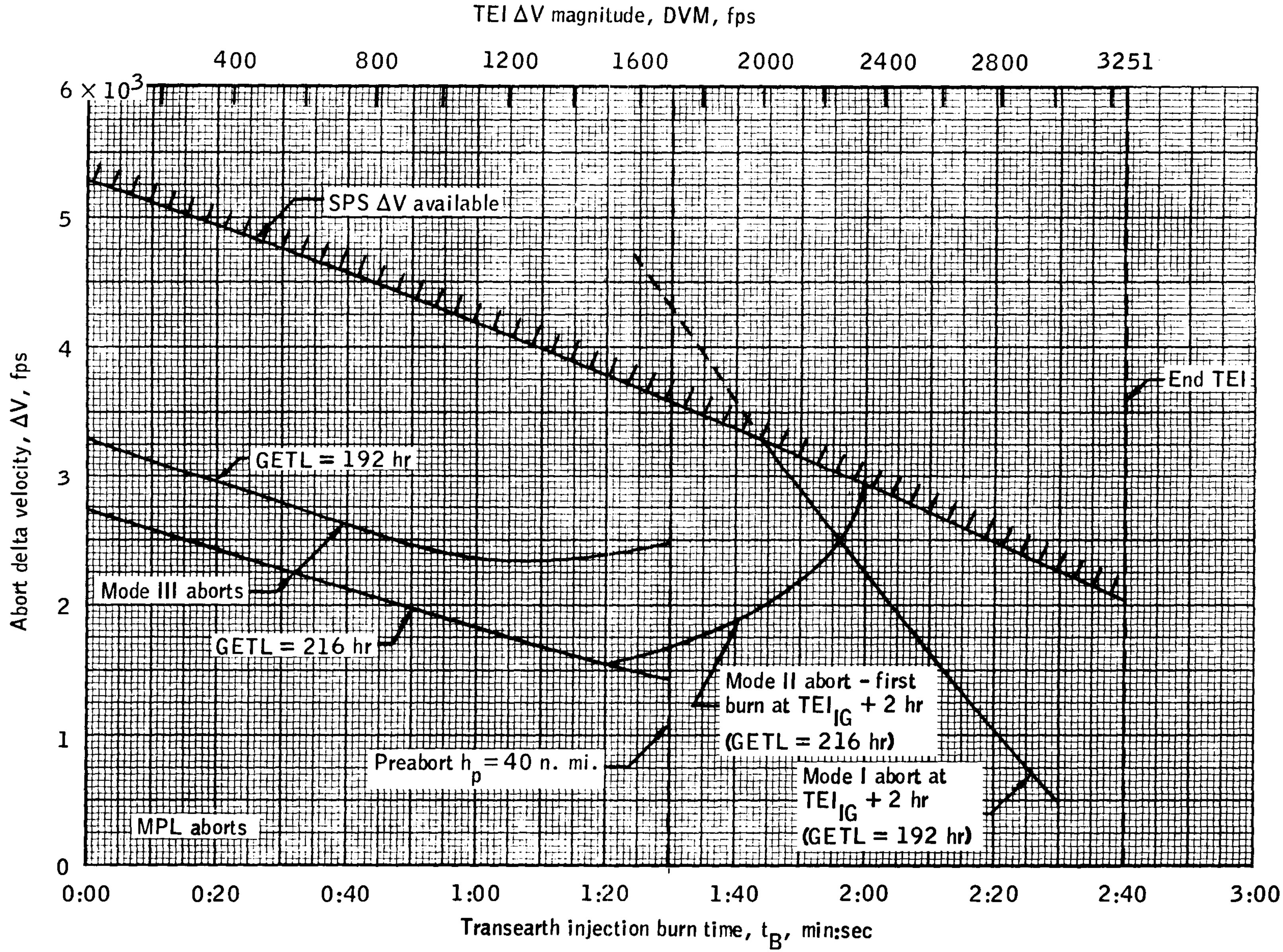
(b) Ground elapsed time of landing for FCUA returns.

Figure 9-7.- Concluded.



(a) Abort ΔV for FCUA returns.

Figure 9-8.- Summary of SPS abort capability as a function of TEI burn time.



(b) Abort ΔV for MPL returns.

Figure 9-8.- Concluded.

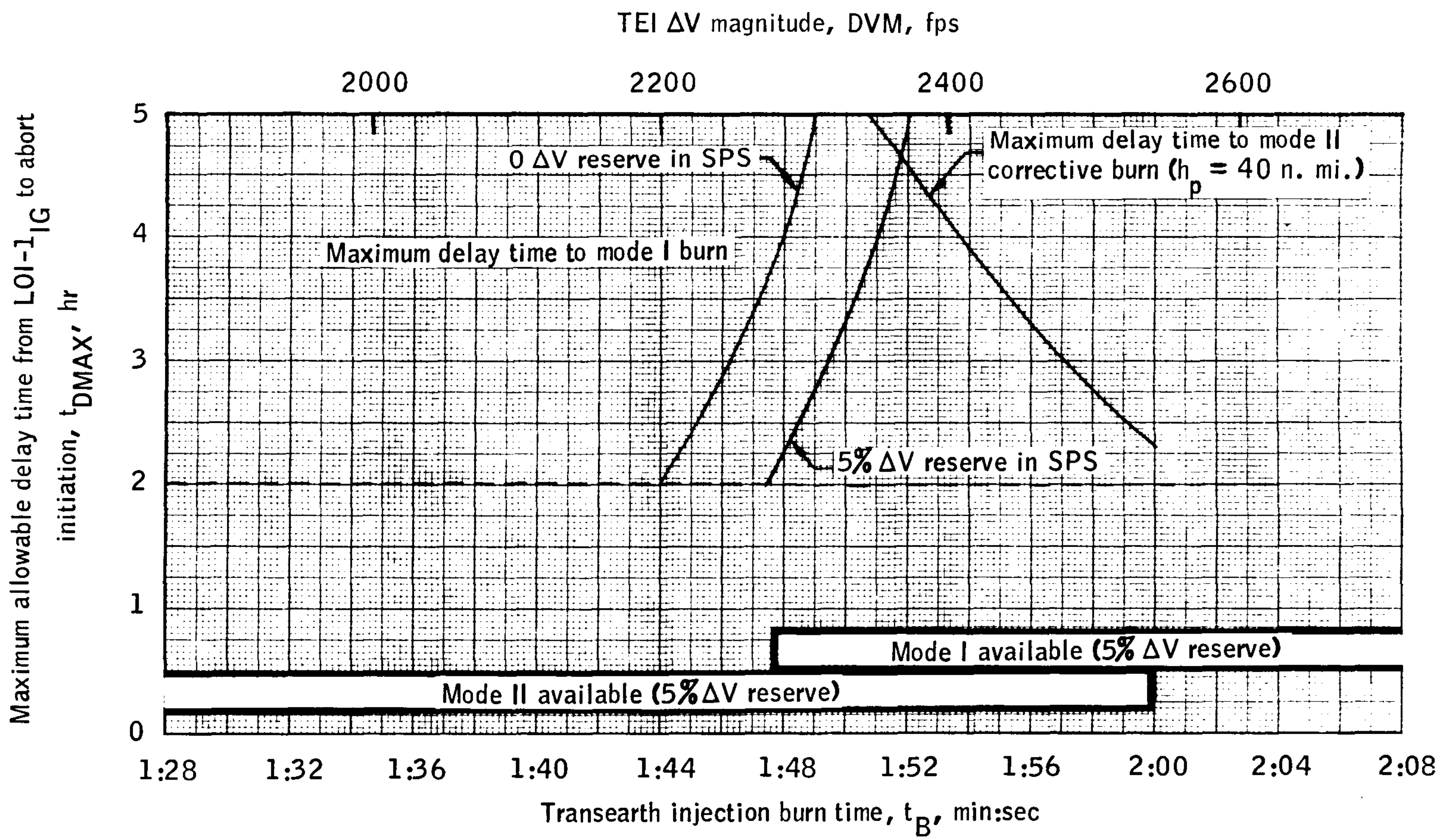


Figure 9-9.- Maximum allowable delay time from TEI_{IG} to abort for premature TEI shutdowns (FCUA returns).

APPENDIX A

CONTINGENCY ANALYSIS SECTION MISSION F DATA PACKAGE

APPENDIX A

CONTINGENCY ANALYSIS SECTION MISSION F DATA PACKAGE[a]

The data that were compiled by the Contingency Analysis Section, Flight Analysis Branch, to generate abort studies for Apollo 10 (Mission F) are presented in tables A-I through A-IX and figures A-I through A-4.

The Reentry Studies Section, Landing Analysis Branch, advised that the entry corridor and target lines as defined in figure A-3, which are the same as those used for the C' mission, are sufficiently accurate for premission studies for the F mission. The primary entry mode to be used in the abort studies will be a simulated constant relative entry range of 1350 n. mi. If necessary, entry range functions and end of mission L/D will reflect C' values. The landing areas (fig. A-4) to which aborts will normally be targeted are the same as those used for the C' mission.

[a]The information in appendix A was taken from MSC memo 69-FM36-59, February 13, 1969.

TABLE A-I.- ABORT CONSTRAINTS

Parameter	Value	Comment
ΔV_{max} (SPS - CSM only), fps	10 000	Assumes no propellant yet burned
ΔV_{max} (SPS - CSM/LM), fps	5 000	
ΔV_{max} (DPS - CSM/LM), fps	2 000	
Return to earth_{max} inclination, deg	40	
VEI_{max}, fps	36 333	Nontime-critical aborts
VEI_{max}, fps	37 500	Time-critical aborts

TABLE A-II.- CSM-106/LM-4 MASS DATA

CSM weights	
Command module, lb	12 276.8
Service module, lb	10 641.8
SLA ring, lb	98.0
CSM less prop., lb	23 016.6
SPS prop., lb	40 633.7
CSM with prop. at launch, lb	63 650.3
CSM at T&D, lb	63 644.9
CSM post-T&D	63 560.9
CSM at LOI-1, lb	[a]62 647.5
CSM at TEI, lb	[b]37 858.0
LM weights	
Ascent stage less prop., lb	4 781.0
LM RCS prop., lb	611.8
LM APS prop., lb	2 619.0
Ascent stage with prop., lb	8 011.8
Descent stage less prop., lb	4 703.0
LM DPS prop., lb	18 134.0
Descent stage with prop., lb	22 837.0
LM with prop. at launch, lb	30 848.8
LM at T&D, lb	30 846.1
CSM/LM docked weights	
CSM/LM post-T&D, lb	94 407.0
CSM/LM at LOI-1, lb	[b]93 133.0

[a]From alternate mission LOI-1 burn trajectory.

[b]These values were obtained from reference 4. All other mass data are from reference 3, amendment number 35, dated January 20, 1969.

TABLE A-III.- REFSMMATS

REFSMMATS[a] are based on a May 17, 1969, launch from pad 39A on an azimuth of 72°, with TLI occurring on the first opportunity. Time of launch is 16:33:49.371 G.m.t.

1. Launch pad REFSMMAT[b]

$$\begin{bmatrix} -.75732833-00 & .59401643-00 & .27128990-00 \\ .12185155+00 & .53667480-00 & -.83494452-00 \\ -.64156523-00 & -.59927005-00 & -.47882088-00 \end{bmatrix}$$

2. Lunar landing site REFSMMAT[c]

$$\begin{bmatrix} .93365762-00 & -.34652012-00 & -.90594009-01 \\ -.70754921-01 & -.42639751-00 & .90176433-00 \\ -.35110853-00 & -.83552916-00 & -.42262729-00 \end{bmatrix}$$

3. Passive thermal control (PTC) REFSMMAT[c]

$$\begin{bmatrix} -.45500032-00 & -.81698549-00 & -.35427310-00 \\ -.89049127-00 & .41744222-00 & .18101735-00 \\ .00000000-00 & .39784005-00 & -.91745479-00 \end{bmatrix}$$

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[a]The REFSMMAT's are listed in the following format.

XX	XY	XZ
YX	YY	YZ
ZX	ZY	ZZ

[b]The launch pad REFSMMAT was generated by the Flight Analysis Branch based on launch data from reference 4.

[c]The lunar landing site and PTC REFSMMAT's were obtained from Lunar Mission Analysis Branch.

TABLE A-IV.- TRIM AERODYNAMIC COEFFICIENTS[a]

FOR BEGINNING OF MISSION

[XCG = 1040.92, YCG = -0.10, ZCG = 5.64, WEIGHT = 12 071.30, DELTA X = -100.33, BANK ANGLE BIAS = -1.02 DEG, PITCH TRIM = 3.629 DEG]

MACH NO.	ALPHA	CL	CD	CL/CD
0.20	171.12	0.22658	0.82699	0.27398
0.40	167.99	0.22656	0.85598	0.26468
0.70	165.51	0.25065	0.99152	0.25279
0.90	162.86	0.30414	1.07398	0.28319
1.10	156.47	0.46833	1.18826	0.39413
1.20	156.55	0.45643	1.17343	0.38897
1.35	155.37	0.53731	1.29459	0.41504
1.65	154.55	0.53220	1.28217	0.41508
2.00	154.58	0.51872	1.29861	0.39945
2.40	155.19	0.49174	1.26969	0.38729
3.00	155.59	0.46405	1.24684	0.37218
4.00	157.39	0.42681	1.23790	0.34479
10.00	157.98	0.41460	1.24858	0.33206
29.50	161.30	0.37067	1.31264	0.28239

[a]The trim aerodynamic coefficients were generated by Flight Analysis Branch based on mass data from reference 3, amendment number 35, dated January 20, 1969.

TABLE A-V.- WEIGHT VERSUS CG FOR DOCKED SPS BURN

ASSUMING APS AND DPS HAVE NOT BEEN BURNED[a]

WEIGHT, LBS	XCG, INCHES	YCG, INCHES	ZCG, INCHES	PITCH, CG	YAW, CG
67 242	247.00	0.99	3.04	-0.705	0.230
69 442	242.02	1.31	2.99	-0.708	0.310
71 642	237.76	1.62	2.95	-0.710	0.389
73 842	234.12	1.90	2.91	-0.712	0.464
76 042	231.05	2.17	2.86	-0.710	0.537
78 242	226.41	2.32	2.95	-0.746	0.587
80 442	220.86	2.35	3.17	-0.822	0.610
82 642	216.03	2.38	3.38	-0.896	0.631
84 842	211.88	2.40	3.58	-0.968	0.650
87 042	208.35	2.43	3.77	-1.036	0.667
89 242	205.38	2.45	3.94	-1.100	0.684
91 442	202.96	2.48	4.12	-1.163	0.699
93 642	201.01	2.50	4.28	-1.219	0.712
95 842	199.57	2.52	4.43	-1.273	0.723

[a]Table A-V was generated by Flight Analysis Branch based on mass data from reference 3, amendment 34, dated January 10, 1969. Additional tables derived from table A-VI will be generated as needed.

TABLE A-VI.- WEIGHT VERSUS CG FOR CSM WITH SPS

PROPELLANT ON BOTTOM OF TANKS

WEIGHT, LBS	XCG, INCHES	YCG, INCHES	ZCG, INCHES	PITCH CG	YAW CG
23 200	983.59	-2.82	8.01	-3.05	-1.07
25 400	971.98	-1.61	7.45	-3.07	-0.66
27 600	962.87	-0.59	6.98	-3.08	-0.26
29 800	956.14	0.28	6.58	-3.06	0.13
32 000	951.27	1.02	6.23	-3.02	0.50
34 200	947.83	1.67	5.93	-2.96	0.84
36 400	945.61	2.25	5.66	-2.88	1.15
38 600	944.33	2.75	5.42	-2.79	1.42
40 800	943.91	3.21	5.21	-2.70	1.66
43 000	944.15	3.61	5.03	-2.59	1.87
45 200	944.98	3.98	4.85	-2.49	2.04
47 400	942.86	4.15	4.90	-2.56	2.17
49 600	939.03	4.12	5.17	-2.80	2.23
51 800	936.22	4.09	5.42	-3.01	2.27
54 000	934.30	4.06	5.65	-3.20	2.30
56 200	933.17	4.03	5.86	-3.35	2.31
58 400	932.72	4.01	6.05	-3.48	2.31
60 600	932.90	3.99	6.24	-3.58	2.29
62 800	933.62	3.97	6.40	-3.65	2.26
65 000	934.89	3.95	6.56	-3.69	2.23

TABLE A-VII.- WEIGHT VERSUS CG FOR LM WITH APS

AND DPS PROPELLANT ON TOP OF TANKS

WEIGHT, LBS	XCG, INCHES	YCG, INCHES	ZCG, INCHES	PITCH CG	YAW CG
14 000	213.14	-0.64	-0.90	0.87	-0.62
14 500	212.13	-0.62	-0.87	0.85	-0.61
15 000	211.12	-0.60	-0.84	0.84	-0.60
15 500	210.13	-0.58	-0.81	0.83	-0.59
16 000	209.13	-0.56	-0.78	0.82	-0.58
16 500	208.15	-0.54	-0.76	0.80	-0.58
17 000	207.19	-0.53	-0.74	0.80	-0.57
17 500	206.23	-0.51	-0.72	0.79	-0.56
18 000	205.29	-0.50	-0.70	0.78	-0.56
18 500	204.36	-0.49	-0.68	0.77	-0.55
19 000	203.43	-0.47	-0.66	0.77	-0.55
19 500	202.53	-0.46	-0.64	0.76	-0.54
20 000	201.63	-0.45	-0.63	0.75	-0.54
20 500	200.73	-0.44	-0.61	0.75	-0.54
21 000	199.84	-0.43	-0.60	0.75	-0.53
21 500	198.99	-0.42	-0.58	0.74	-0.53
22 000	198.12	-0.41	-0.57	0.74	-0.53
22 500	197.25	-0.40	-0.56	0.74	-0.53
23 000	196.38	-0.39	-0.55	0.74	-0.53
23 500	195.56	-0.38	-0.53	0.74	-0.53
24 000	194.69	-0.37	-0.52	0.74	-0.53
24 500	193.85	-0.37	-0.51	0.74	-0.53
25 000	193.02	-0.36	-0.50	0.74	-0.53
25 500	192.20	-0.35	-0.49	0.74	-0.53
26 000	191.37	-0.35	-0.48	0.74	-0.53
26 500	190.53	-0.34	-0.47	0.74	-0.53
27 000	189.72	-0.33	-0.46	0.75	-0.53
27 500	188.91	-0.33	-0.46	0.75	-0.54
28 000	188.09	-0.32	-0.45	0.75	-0.54
28 500	187.26	-0.32	-0.44	0.76	-0.54
29 000	186.47	-0.31	-0.43	0.76	-0.55
29 500	185.64	-0.30	-0.43	0.77	-0.55
30 000	184.81	-0.30	-0.42	0.78	-0.56
30 500	183.99	-0.29	-0.41	0.79	-0.56
31 000	183.58	-0.29	-0.40	0.78	-0.56
31 500	183.23	-0.29	-0.40	0.78	-0.56
32 000	182.88	-0.28	-0.39	0.78	-0.56
32 500	182.55	-0.28	-0.39	0.77	-0.55

TABLE A-VIII.- WEIGHT VERSUS CG FOR CSM WITH SPS

PROPELLANT ON TOP OF TANKS

WEIGHT, LBS	XCG, INCHES	YCG, INCHES	ZCG, INCHES	PITCH, CG	YAW, CG
23 200	982.28	-2.58	8.25	-3.17	-0.99
25 400	981.18	-1.40	7.67	-2.97	-0.54
27 600	979.27	-0.40	7.18	-2.81	-0.16
29 800	976.62	0.46	6.76	-2.70	0.18
32 000	973.41	1.19	6.40	-2.61	0.49
34 200	969.77	1.83	6.09	-2.55	0.77
36 400	965.84	2.39	5.81	-2.51	1.03
38 600	961.63	2.89	5.57	-2.48	1.29
40 800	957.15	3.34	5.35	-2.47	1.54
43 000	952.52	3.74	5.15	-2.47	1.79
45 200	947.72	4.10	4.98	-2.49	2.05
47 400	947.64	4.18	5.11	-2.56	2.09
49 600	948.14	4.15	5.38	-2.68	2.07
51 800	947.91	4.12	5.62	-2.80	2.06
54 000	947.04	4.09	5.84	-2.94	2.06
56 200	945.59	4.06	6.04	-3.08	2.07
58 400	943.66	4.04	6.23	-3.23	2.09
60 600	941.28	4.02	6.40	-3.39	2.13
62 800	938.50	4.00	6.56	-3.57	2.17
65 000	935.70	3.98	6.72	-3.75	2.22

TABLE A-IX.- WEIGHT VERSUS CG FOR LM WITH APS AND DPS PROPELLANT ON BOTTOM OF TANKS

WEIGHT, LBS	XCG, INCHES	YCG, INCHES	ZCG, INCHES	PITCH CG	YAW CG
14 000	208.60	-0.67	0.58	-0.60	-0.70
14 500	206.04	-0.65	0.56	-0.61	-0.71
15 000	203.73	-0.62	0.54	-0.62	-0.72
15 500	201.63	-0.60	0.52	-0.63	-0.73
16 000	199.72	-0.59	0.50	-0.63	-0.73
16 500	197.97	-0.57	0.49	-0.64	-0.74
17 000	196.38	-0.55	0.47	-0.64	-0.74
17 500	194.92	-0.54	0.46	-0.64	-0.75
18 000	193.59	-0.52	0.45	-0.65	-0.75
18 500	192.35	-0.51	0.44	-0.65	-0.76
19 000	191.23	-0.49	0.42	-0.65	-0.76
19 500	190.21	-0.48	0.41	-0.65	-0.76
20 000	189.27	-0.47	0.40	-0.65	-0.76
20 500	188.41	-0.46	0.39	-0.65	-0.76
21 000	187.62	-0.45	0.38	-0.65	-0.76
21 500	186.92	-0.44	0.37	-0.65	-0.76
22 000	186.24	-0.43	0.37	-0.65	-0.76
22 500	185.66	-0.42	0.36	-0.65	-0.75
23 000	185.14	-0.41	0.35	-0.64	-0.75
23 500	184.67	-0.40	0.34	-0.64	-0.74
24 000	184.21	-0.39	0.34	-0.64	-0.74
24 500	183.85	-0.38	0.33	-0.63	-0.73
25 000	183.52	-0.37	0.32	-0.63	-0.73
25 500	183.22	-0.37	0.32	-0.62	-0.72
26 000	182.95	-0.36	0.31	-0.61	-0.71
26 500	182.73	-0.35	0.30	-0.61	-0.70
27 000	182.56	-0.35	0.30	-0.60	-0.70
27 500	182.41	-0.34	0.29	-0.59	-0.69
28 000	182.28	-0.33	0.29	-0.58	-0.68
28 500	182.19	-0.33	0.28	-0.57	-0.67
29 000	182.15	-0.32	0.28	-0.57	-0.66
29 500	182.11	-0.32	0.27	-0.56	-0.65
30 000	182.11	-0.31	0.27	-0.55	-0.64
30 500	182.16	-0.31	0.26	-0.54	-0.62
31 000	182.20	-0.30	0.26	-0.53	-0.61
31 500	181.88	-0.30	0.26	-0.53	-0.61
32 000	181.52	-0.29	0.25	-0.52	-0.61
32 500	181.18	-0.29	0.25	-0.52	-0.61

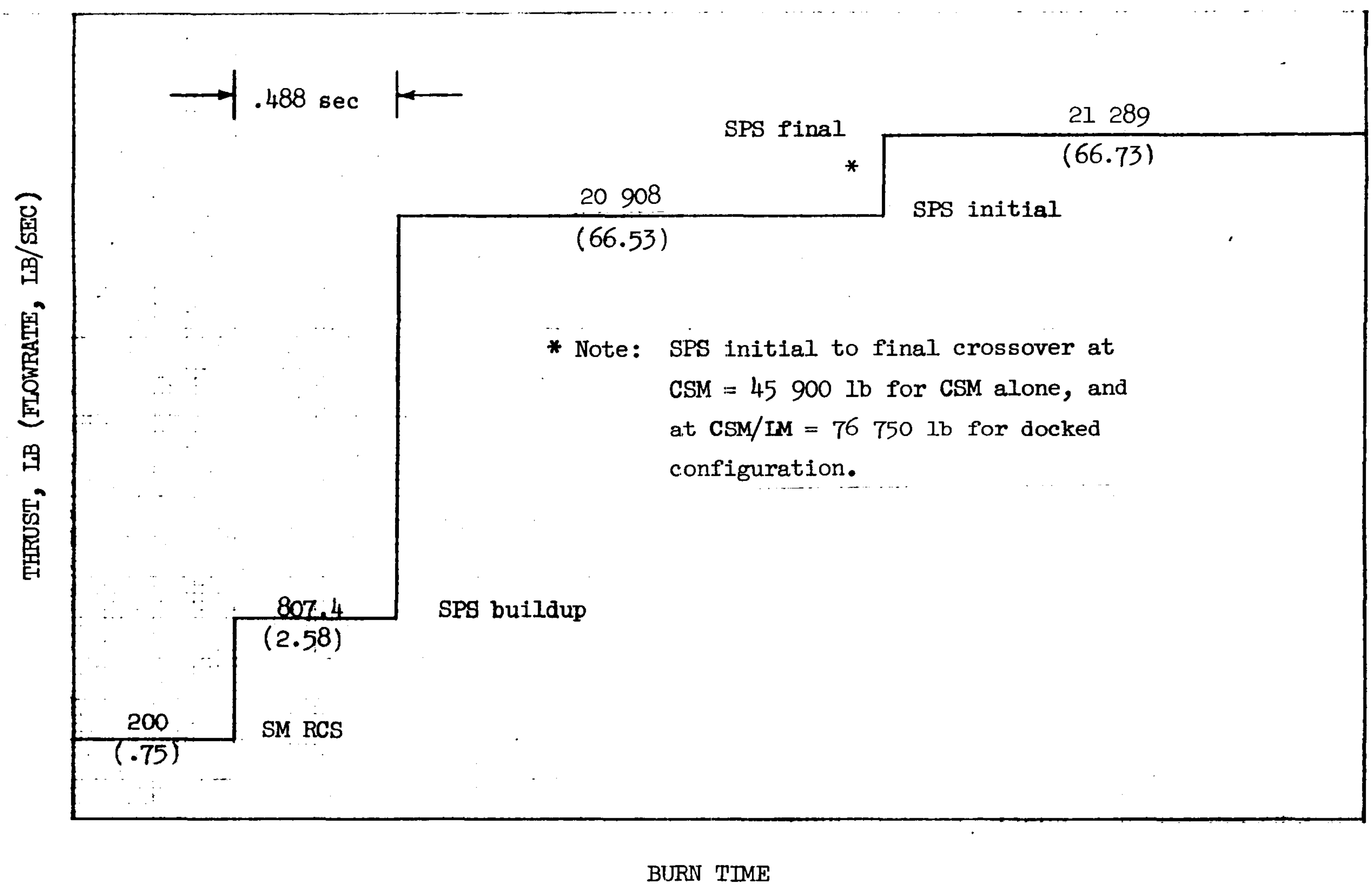


Figure A-1.- SM RCS and SPS engine parameters.

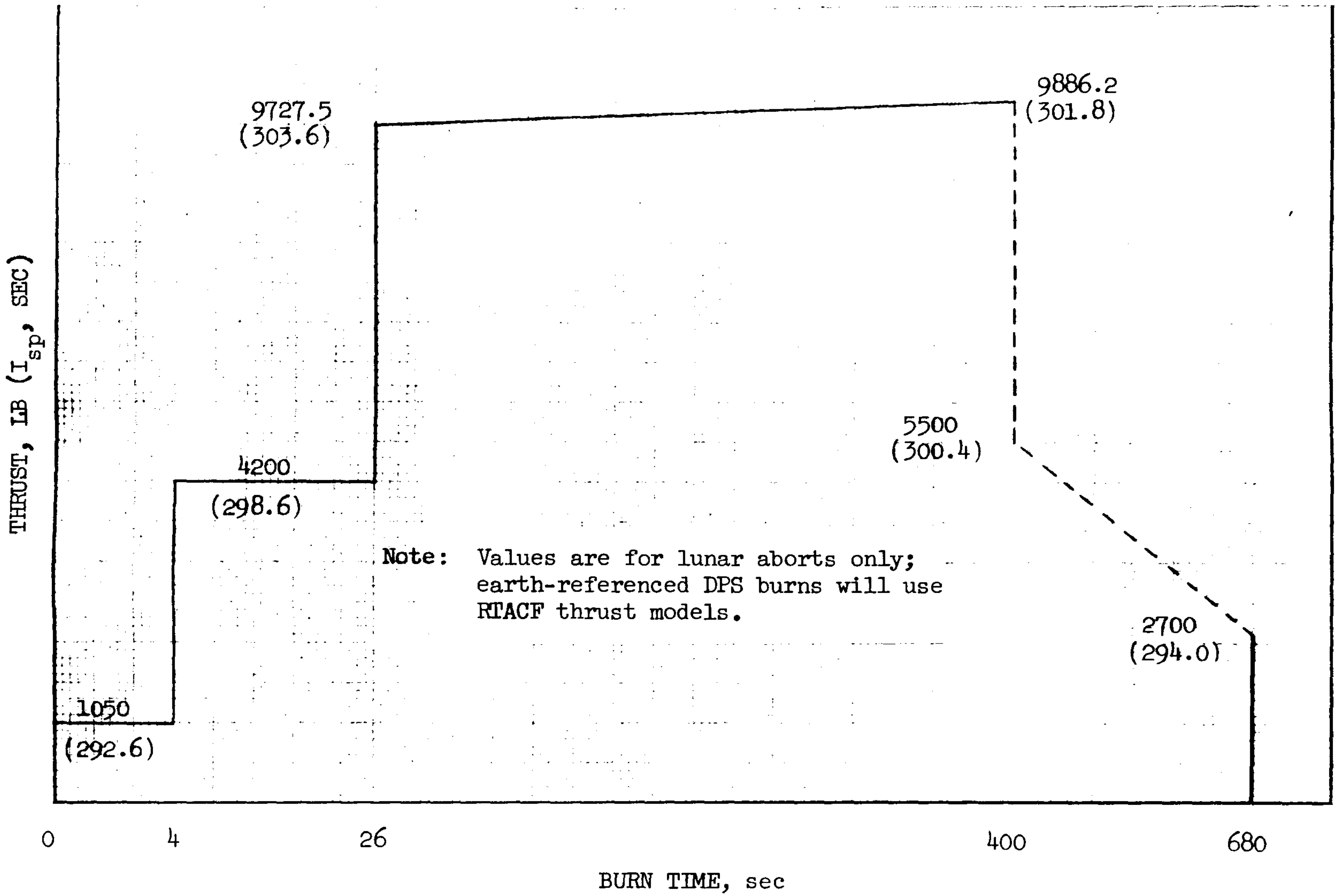


Figure A-2.- DPS engine parameters.

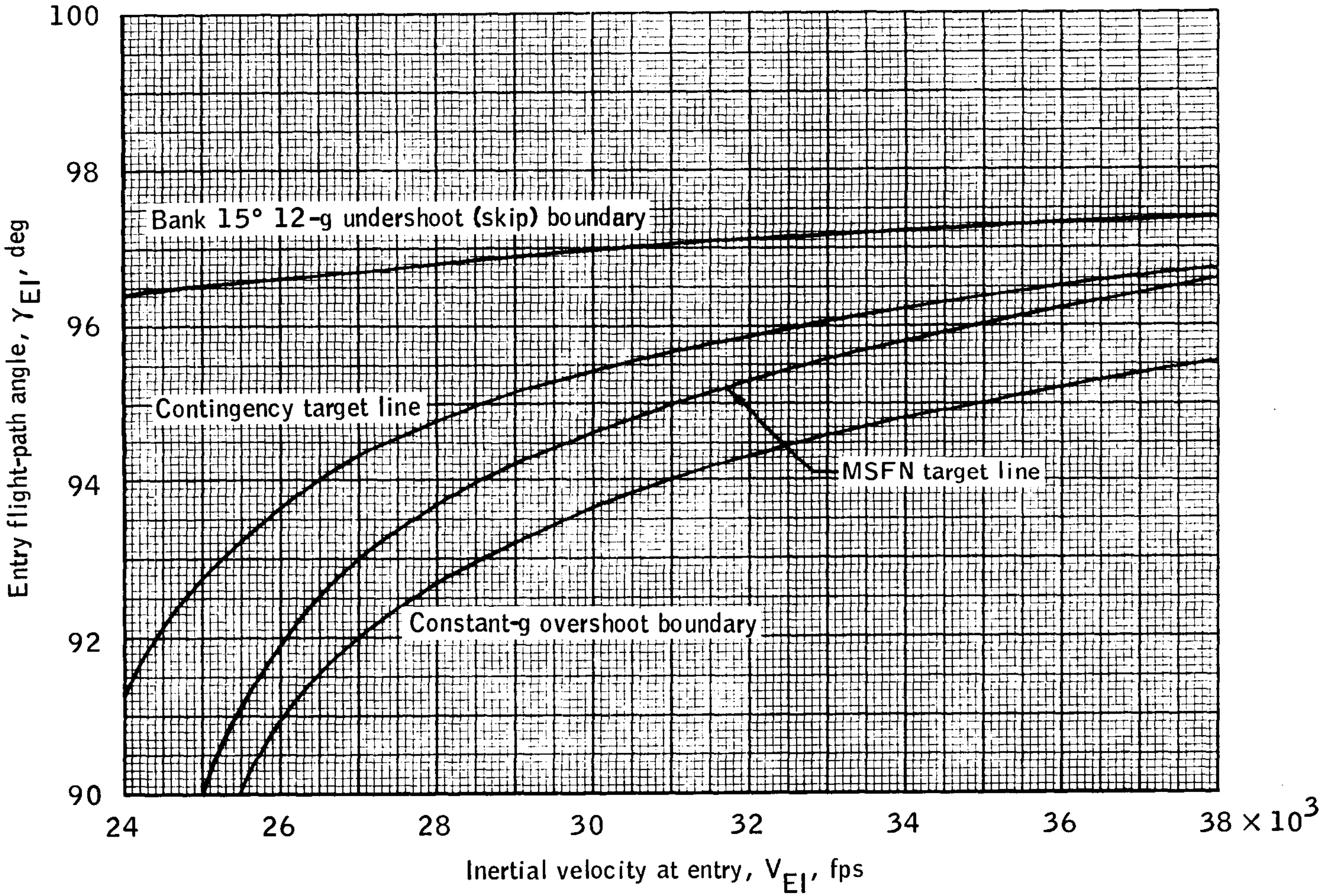


Figure A-3.- Entry corridor definition.

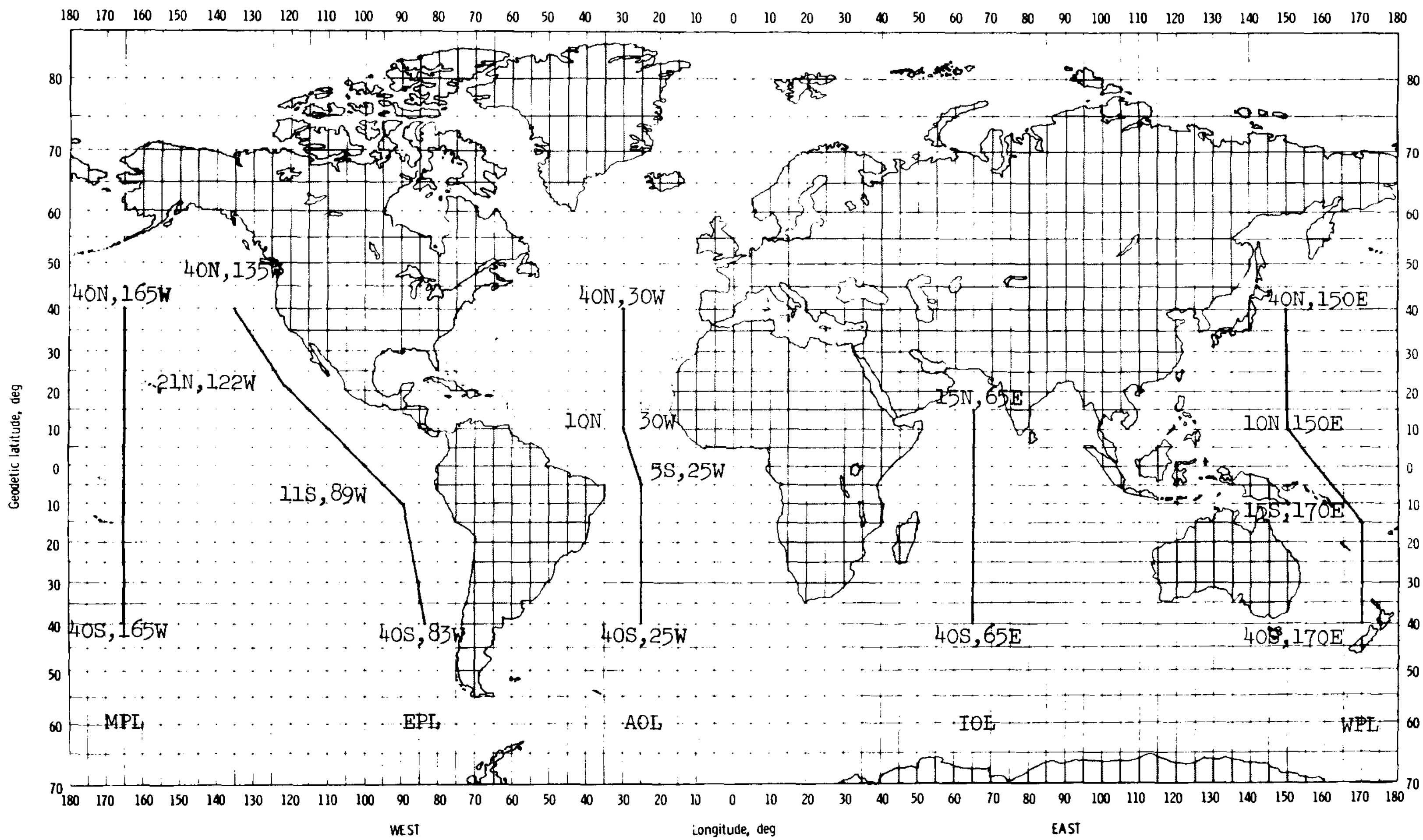
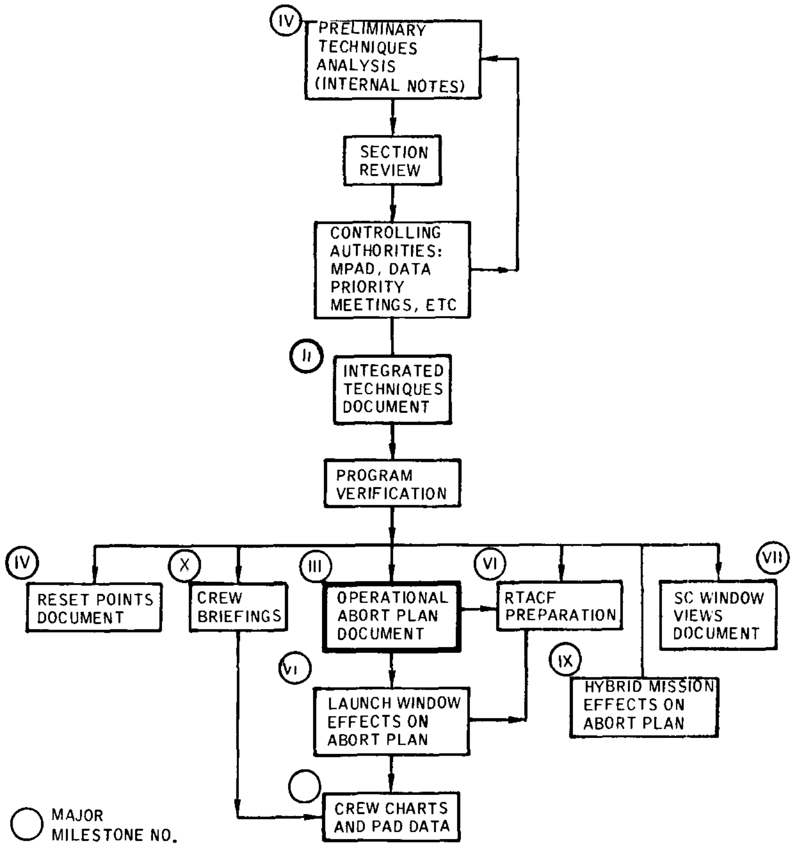


Figure A-4.- Contingency landing area definition.

APPENDIX B

MAJOR MISSION F MIBESTONES

FOR THE CONTINGENCY ANALYSIS SECTION



MAJOR MILESTONES

I. PRELIMINARY CONTINGENCY TECHNIQUES
(PRELIMINARY PROCEDURES TO BE FOLLOWED BY CREW AND GROUND)

II. INTEGRATED CONTINGENCY (ABORT) TECHNIQUES DOCUMENT
(DEFINES DETAILED PROCEDURES TO BE FOLLOWED BY CREW AND GROUND

III. OPERATIONAL ABORT PLAN DOCUMENT
(PROVIDES DETAILED ABORT SOLUTIONS AND ASSOCIATED DATA FOR THE PRIMARY LAUNCH OPPORTUNITY)

IV. RESET POINTS DOCUMENT
(INSURES CREW/RTCC SIMULATION OF ABORTS ARE COMPATIBLE AND PROVIDES DRIVER FOR SIMULATION)

V. CREW CHARTS AND PAD DATA
(PROVIDES FINAL PLOTS AND CHARTS CREW NEEDS TO CARRY OUT OPERATIONAL ABORT PLAN; INCLUDES LAUNCH, TLI MONITOR, TLI + 10 MIN ABORT AND LOI 15 MIN ABORT CHARTS)

VI. RTACF PREPARATION
(INSURES REAL-TIME OPERATIONAL COMPUTER SUPPORT REQUIREMENTS WILL BE MET)

VII. SC WINDOW VIEWS DOCUMENT
(PROVIDES STARFIELD, EARTH-MOON TERMINATOR, AND HORIZON SCHEMATICS FOR CREW FAMILIARIZATION AND INDEPENDENT MANEUVER ATTITUDE CHECKS)

VIII. LAUNCH WINDOW EFFECTS ON THE OPERATIONAL ABORT PLAN (DOCUMENT)
(INSURES ADEQUACY OF ABORT PLAN FOR LAUNCH WINDOW VARIATIONS)

IX. HYBRID MISSION EFFECTS ON THE OPERATIONAL ABORT PLAN
(SUPPLEMENTS ABORT PLAN FOR TRAJECTORY AND PROCEDURE CHANGES DUE TO A HYBRID PROFILE)

X. CREW BRIEFINGS
(EXPLANATION OF PROCEDURES AND ONBOARD ABORT CHARTS)

Figure 2-2. - Major Apollo F and G milestones for the Contingency Analysis Section.

APPENDIX C

TIME CRITICAL ABORTS FOR THE F MISSION

APPENDIX C

TIME CRITICAL ABORTS FOR THE F MISSION[a]

Abort parameters that result in minimum return times for the F mission are presented in figures C-1 and C-2. The minimum time from abort to landing as a function of ground elapsed time of abort is shown in figure C-1 for the translunar coast, free-return trajectory, and transearth coast. Although the data were generated in the unspecified area mode, approximate points have been identified during the three phases for which the landing areas for the time critical aborts would be one of the five contingency landing areas. In figure C-2, the impulsive abort ΔV and inertial velocity at entry are shown for those phases where either the maximum available ΔV could not be used or where the time critical abort entry velocity constraint of 37 500 fps was not reached.

Because the maximum abort ΔV during translunar coast could be achieved by first a burn by the DPS engine in the docked configuration (ΔV = 2000 fps) and then a jettison of the LM and a burn by the SPS engine for the CSM only (ΔV = 10 000 fps), the aborts during translunar coast were generated with an impulsive ΔV of 12 000 fps assumed to get an approximate, if optimistic, estimate of the minimum return times possible. Note that after approximately 73 hours g.e.t., the fastest return times require a coast past pericynthion before the abort is performed. All aborts used the entire 12 000 fps; none of the trajectories exceeded the entry velocity constraint (fig. C-2). For purposes of comparison, CSM only aborts (ΔV = 10 000 fps) are also shown for the translunar coast.

Aborts from the free-return trajectory are constrained by the maximum allowable entry velocity (V_{EI} = 37 500 fps); therefore, the ΔV capability of the spacecraft (ΔV = 12 000 fps) is never reached (fig. C-2).

Transearth coast aborts were generated from a ΔV of 2500 fps based on the CSM weight and SPS propellant that remained after transearth injection. The maximum allowable entry velocity is never reached for these aborts with a ΔV of 2500 fps (fig. C-2).

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[a]Information in this appendix was taken from MSC memo 69-FM36-58, February 18, 1969.

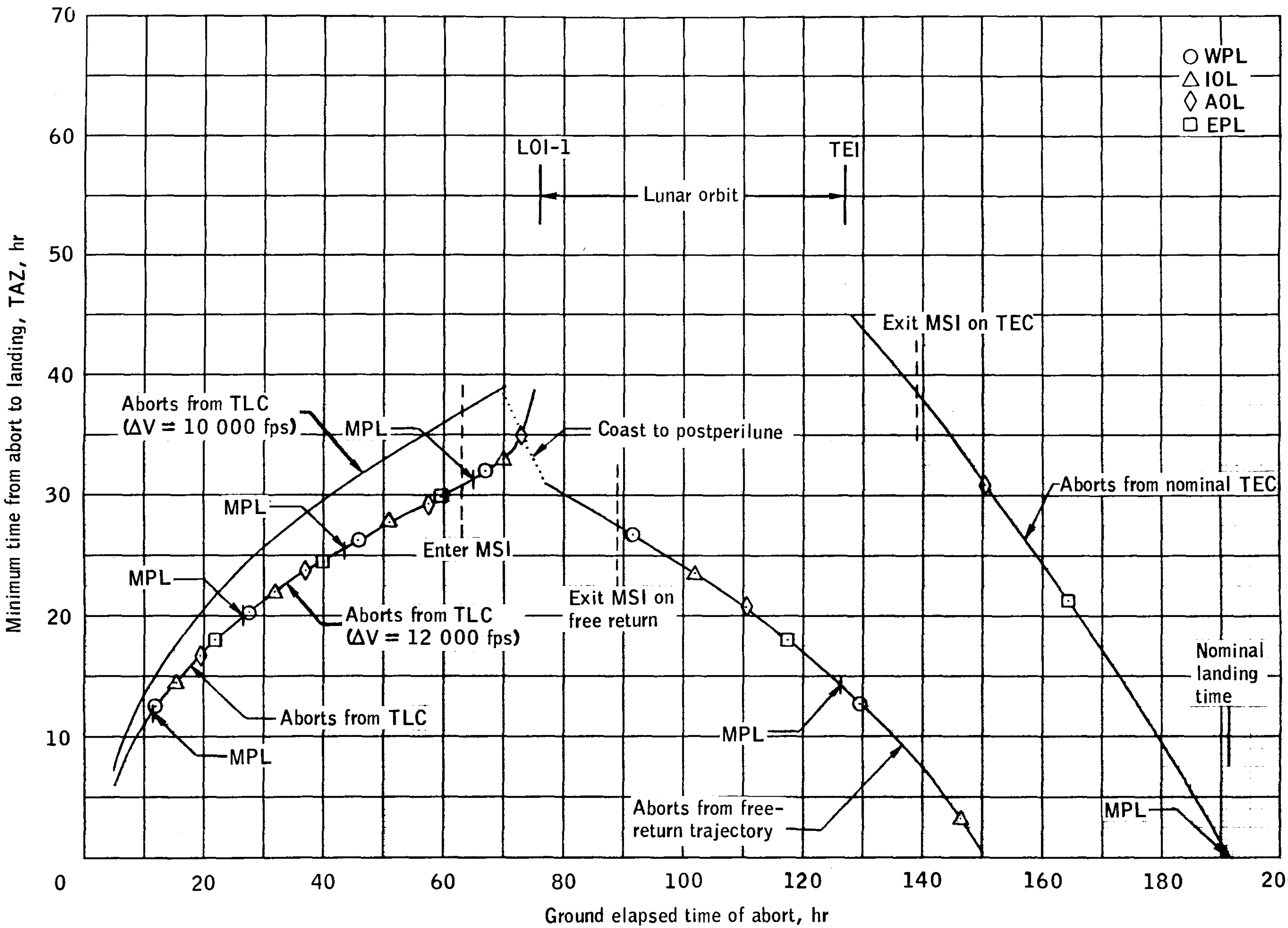

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Figure C-1.- Typical minimum return times for the F mission.

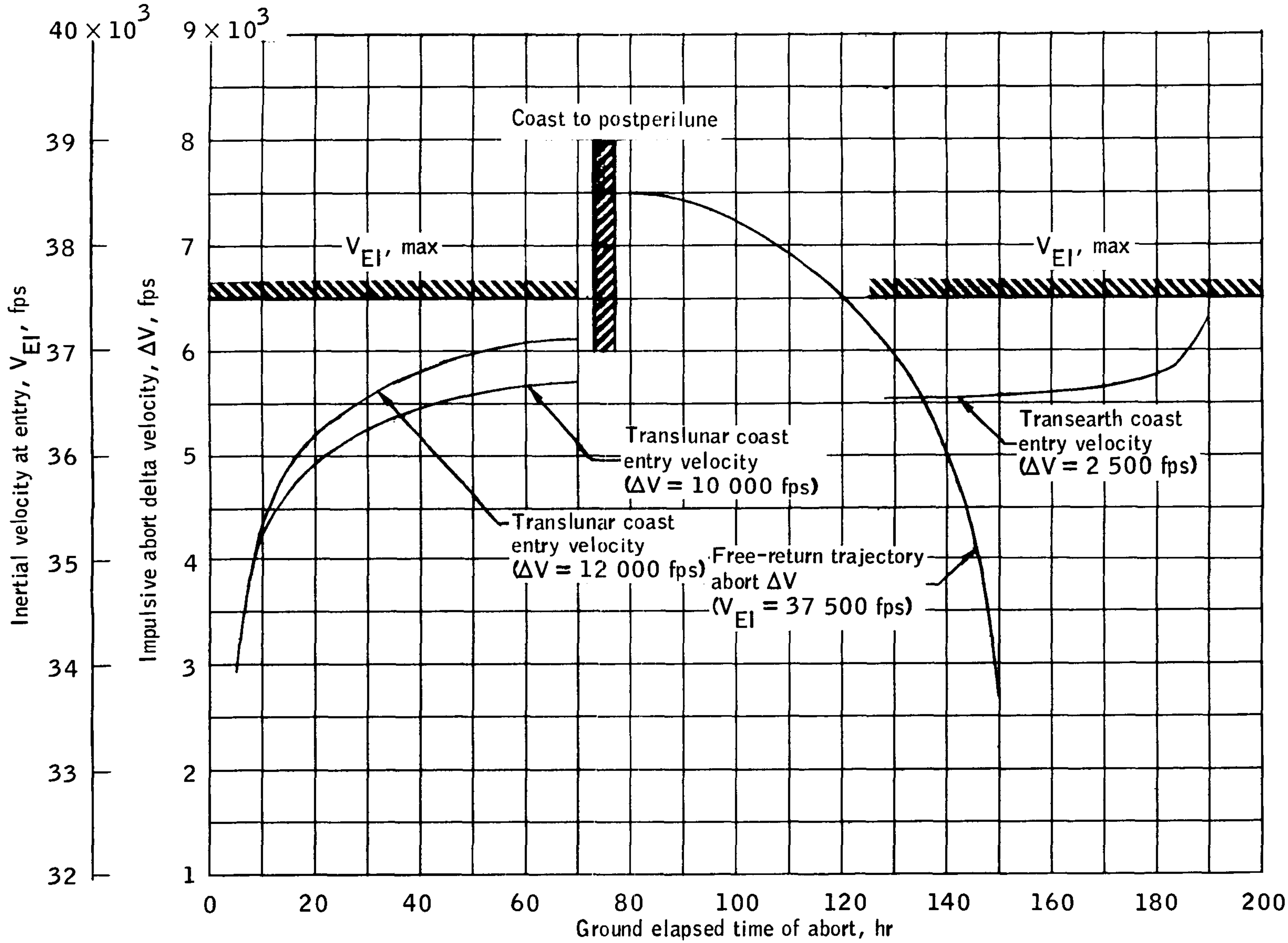

Figure C-2.- Abort ΔV and entry velocity for F mission minimum return times.

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