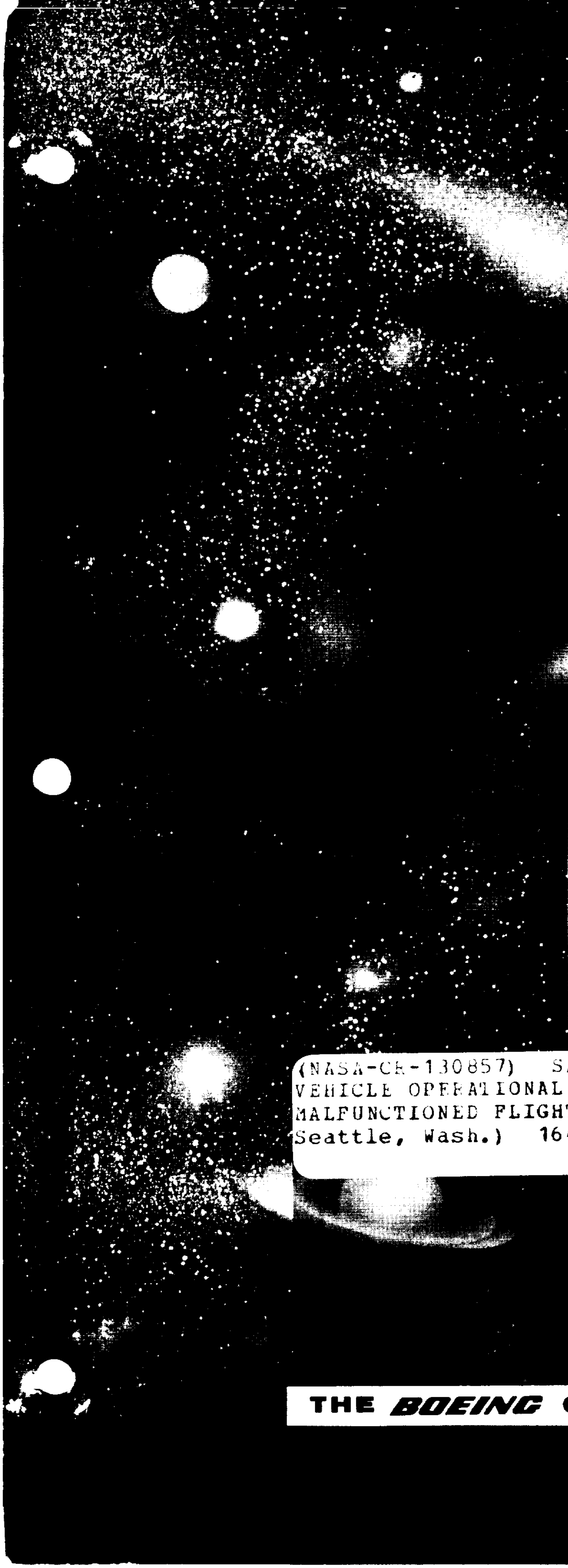

D5-15795-7

PRA

SATURN V AS-507 LAUNCH VEHICLE OPERATIONAL ABORT AND MALFUNCTIONED FLIGHT ANALYSIS

(NASA-CR-130857) SATURN 5 AS-507 LAUNCH VEHICLE OPERATIONAL ABORT AND MALFUNCTIONED FLIGHT ANALYSIS (Boeing Co., Seattle, Wash.) 164 p

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JULY 30, 1969

THE BOEING COMPANY · AEROSPACE GROUP

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EMERGENCY DETECTION SYSTEM,
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ABSTRACT AND LIST OF KEY WORDS

This document contains the abort and malfunctioned flight analysis conducted for the Saturn V AS-507 launch vehicle. The effects of various failure modes on S-IC, S-II, and S-IVB flight are evaluated in terms of abort criteria, mission completion capability, and communications and tracking surveillance.

Saturn V/Apollo Vehicle
Emergency Detection System
Abort
Malfunctioned Flight
Flight Dynamics
Control Systems Failures
Propulsion Malfunctions
Accelerometer Failures
Structural Integrity
Controllability
Vehicle Loss
Crew Safety
Acquisition and Loss Data
Vehicle Surveillance
Geometrical Communications Analysis

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ILLUSTRATIONS (CONTINUED)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (CONTINUED)

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REFERENCES

1. Boeing Document D5-15795-6, "Saturn V AS-506 Launch Vehicle Operational Abort and Malfunctioned Flight Analysis," dated June 3, 1969.

2. Boeing Coordination Sheet EDS-H-256, "EDS Criticality Determination and Bar Chart Summaries," dated April 29, 1969.

3. MSFC Memorandum S&E-ASTN-DIR-69-150, "Saturn V Overall Failure Mode Criticality and Probability of Occurrence," dated March 27, 1969.

4. MSFC Memorandum S&E-ASTN-STA-69-2, "Saturn V Overall Failure Mode Criticality and Probability of Occurrence - AS-506," dated June 25, 1969.

5. Boeing Document D5-15795-5A, "Saturn V AS-505 Launch Vehicle Operational Abort and Malfunctioned Flight Analysis," dated April 25, 1969.

6. MSFC Document I-MO-3-69, "Final Flight Mission Rule Input Document," dated May 16, 1966. (AS-506)

7. Boeing Memorandum 5-9600-H-280, "SSR-249, Spacecraft Backup for Saturn V Inertial Platform, AS-505," dated May 14, 1969.

8. MSC Memorandum ET 253/6810-170-B, "Required Lead Times, Saturn V," dated October 3, 1968.

SECTION 1

SUMMARY

This document contains the results of malfunctioned flight and abort analysis for the AS-507 G mission. The G mission is a lunar landing mission with launch scheduled for the September-October launch window.

Analysis results reported in this document include the following:

a. Evaluation of the operational effectiveness of the Emergency Detection System (EDS) automatic and manual abort limits to provide crew safety and minimize false aborts

b. Vehicle dynamic and structural load responses, regions of controllability, and trajectory envelopes resulting from vehicle malfunction

c. Vehicle capability to achieve the performance plateaus defined by launch vehicle mission requirements - earth parking orbit, translunar injection, and a 75 nautical mile contingency orbit

d. Verification of LVDC flight program presettings for malfunctioned flight

e. An evaluation of the surveillance network's capability to provide adequate tracking and communications during malfunctioned flight.

The AS-507 G mission analysis is similar to that for the AS-506 G mission, reported in Reference 1, with the following major differences:

a. The EDS rate limit between 120 seconds and S-IC OBECO is changed from 4 degrees per second to 10 degrees per second.

b. The setting for the overrate light between 120 seconds and 134 seconds is changed from 4 degrees per second to 10 degrees per second.

c. The effects of the S-IC Automatic Abort Cutoff (AACO) logic following dual adjacent engines out after CECO are evaluated in detail.

d. The analysis is expanded to evaluate the effects of single-axis platform failures for various failure rates and accelerations.

SUMMARY (Continued)

e. The effects of accelerometer failures coupled with thrust misalignments and failures across various launch windows are evaluated.

Significant results derived from this analysis are summarized in the following paragraphs.

1.1 MALFUNCTION DYNAMICS

Malfunction dynamics for the AS-507 are essentially the same as for AS-506, in spite of increased winds. Times during which loss of control occurs are approximately the same for AS-507 as for AS-506. Control is maintained for adjacent simultaneous S-IC engine failures occuring after 135 seconds. This is due to all engines being shut down when any two adjacent control engines are out after 135 seconds. Loss of control in S-II, however, can result from sequential S-IC adjacent engine failures where the second engine fails before approximately 150 seconds.

Spacecraft loads after engine-out are slightly increased for AS-507 because of higher winds but remain below structural limits for all flight times.

A summary of vehicle controllability for all types of malfunctions in all stages is shown in Figures 1-1 through 1-8.

1.2 CREW SAFETY

Periods of possible crew loss for AS-507 are essentially the same as for AS-506.

These periods occur only for short time intervals during S-IC flight and are the result of a single engine out, an actuator hardover, or a saturated error signal. No other malfunctions considered cause crew loss.

The crew loss criticalities for these three malfunctions, as well as the flight times during which crew loss can occur, are shown in Figure 1-2. Probability of crew loss due to catastrophic S-IC engine failure remains unchanged at 61×10^{-6}.

1.3 PERFORMANCE

Predicted malfunctioned flight performance summarized in Figure 1-8 for AS-507 indicates generally improved POI/TLI capability over AS-506 for propulsion failures. TLI may

SUMMARY (Continued)

be achieved with an S-IC single engine out approximately 15 seconds prior to that predicted for AS-506. TLI may be achieved with an S-II single engine out 30 seconds prior to that predicted for AS-506. Improvement for other propulsion failures ranges from 5 to 15 seconds. Parking orbit capability exists for any S-IC adjacent control engine failure occuring after approximately 150 seconds. Capability to achieve an orbit greater than 75 NMI perigee for accelerometer failures is unchanged from that predicted for AS-506, except for X-axis accelerometer failures with 3-sigma high and low performing vehicles. The earliest 3-sigma high failure time to 75 NMI perigee orbit has been extended from 110 seconds from first motion for AS-506 to 135 seconds for AS-507. The 3-sigma low time has been extended from 0.0 second to 30 seconds.

A summary of AS-507 POI/TLI capability for engine out and early staging is given in Figure 1-8 and for accelerometer failures in Figure 1-9.

Improvement in propulsion malfunction performance is due to the steeper trajectory profile for AS-507 during S-IC and the first portion of S-II burn; this results in a higher altitude for the same failure time. The AS-507 into-orbit pitch profile differs from the AS-506 pitch profile and results in lower perigees for X-accelerometer failures with the present presettings.

1.4 TRACKING AND COMMUNICATIONS

The communications and tracking analysis indicates that the geometrical coverage for malfunctioned flight is comparable to that for nominal flight.

MALFUNCTION	FLIGHT TIMES - MALFUNCTION RESULTS IN VEHICLE LOSS (FLIGHT TIME - SECONDS)	PROBABILITY OF VEHICLE LOSS
SINGLE ENGINE OUT		
• PAD FALLBACK	0.2	$<1 \times 10^{-6}$
• LIFTOFF INTERFERENCE	0.9	$<3 \times 10^{-6}$
• TOWER COLLISION (NO. 1 OR NO. 2 ONLY)	5.7	$<8 \times 10^{-6}$
ACTUATOR HARDOVER		
• PAD FALLBACK	NONE	$<1 \times 10^{-6}$
• LIFTOFF INTERFERENCE	0.7	$<2 \times 10^{-6}$
• TOWER COLLISION (+ YAW ONLY)	3.2	$<3 \times 10^{-6}$
DUAL ENGINE OUT		
• PAD FALLBACK	30	$<1 \times 10^{-6}$
• LIFTOFF INTERFERENCE	1.5	$<1 \times 10^{-6}$
• TOWER COLLISION (TWO TOWER-SIDE ENGINES)	7.5	$<1 \times 10^{-6}$
SATURATED CONTROL SIGNAL		
• PAD FALLBACK	30	$<11 \times 10^{-6}$
• LIFTOFF INTERFERENCE	1.3	$<1 \times 10^{-6}$
• TOWER COLLISION (+ YAW)	3.5	$<1 \times 10^{-6}$
• TOWER COLLISION (- YAW)	6.5	$<1 \times 10^{-6}$

FIGURE 1-1 SUMMARY OF MALFUNCTIONS REQUIRING NEAR-PAD ABORT

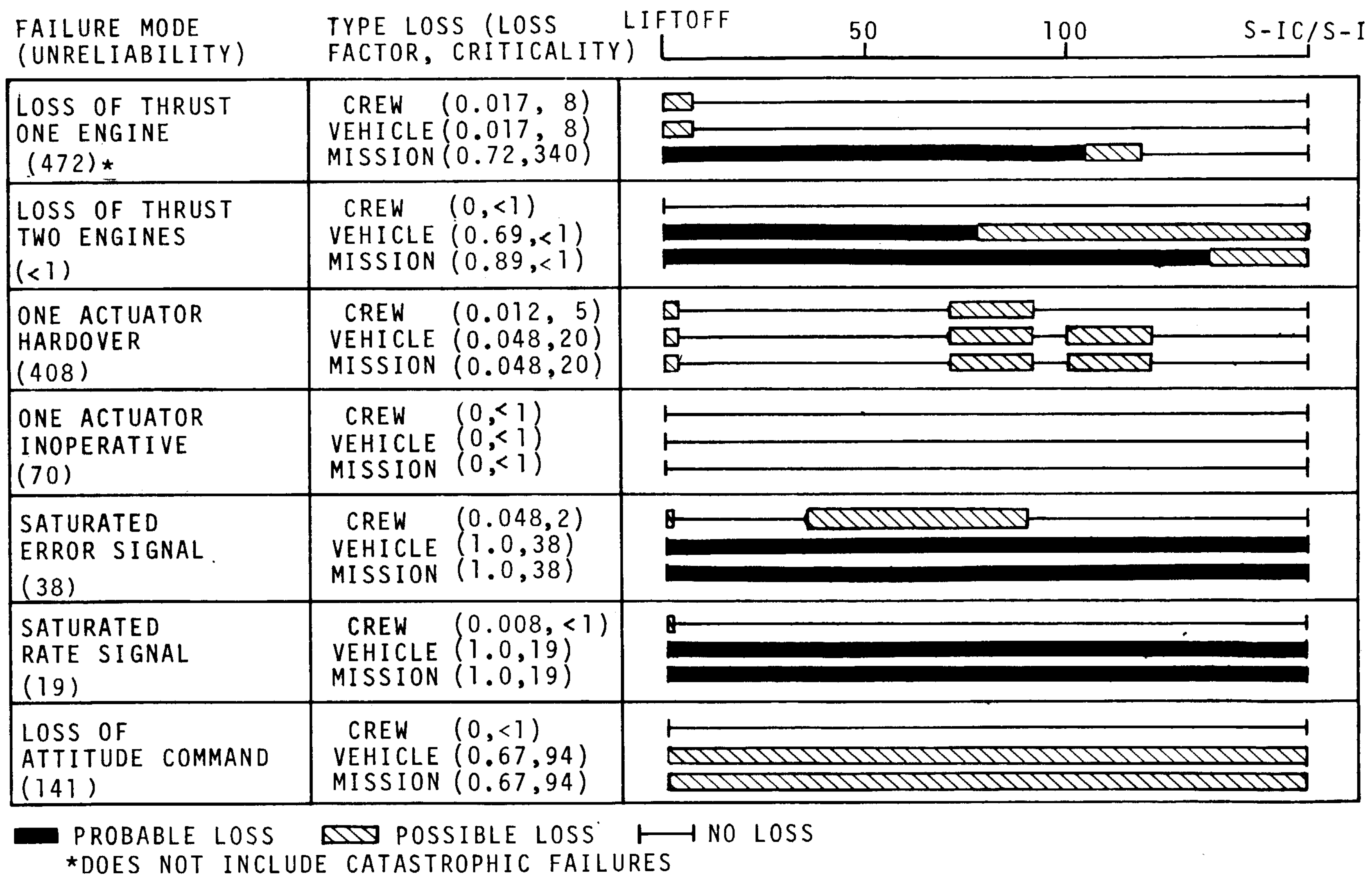
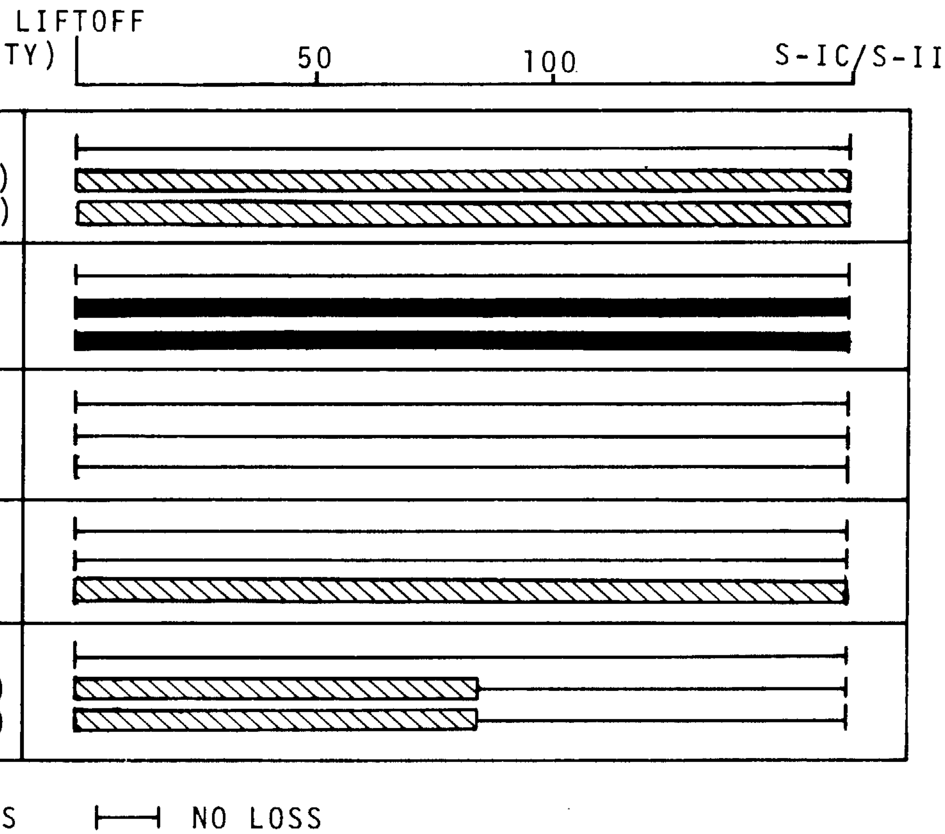


FIGURE 1-2 S-IC MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF ATTITUDE ERROR (254)	CREW (0,<1) VEHICLE (0.67,169) MISSION (0.67,169)
LOSS OF ATTITUDE RATE (12)	CREW (0,<1) VEHICLE (1.0,12) MISSION (1.0,12)
LOSS OF * INERTIAL ATTITUDE (2091)	CREW (0,<1) VEHICLE (0,<1) MISSION (0,<1)
ACCELEROMETER FAILURE (1960)	CREW (0,<1) VEHICLE (0,<1) MISSION (0.5,980)
SEQUENCING AND STAGING MALFUNCTIONS (1012)	CREW (0,<1) VEHICLE (0.27,273) MISSION (0.27,273)



*Assumes no delay between platform failure and recognition of the failure by the LVDC.

FIGURE 1-2 S-IC MALFUNCTION SUMMARY (SHEET 2 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF THRUST ONE ENGINE (18370)	CREW (0,<1) VEHICLE(0.03,551) MISSION(0.34,6246)
LOSS OF THRUST TWO ENGINES (103)	CREW (0,<1) VEHICLE (0.33,34) MISSION (0.70,72)
ONE ACTUATOR HARDOVER (2238)	CREW (0,<1) VEHICLE (0.50,1119) MISSION (0.50,1119)
ONE ACTUATOR INOPERATIVE (80)	CREW (0,<1) VEHICLE (0,<1) MISSION (0,<1)
SATURATED ERROR SIGNAL (36)	CREW (0,<1) VEHICLE (1.0,36) MISSION (1.0,36)
SATURATED RATE SIGNAL (18)	CREW (0,<1) VEHICLE (1.0,18) MISSION (1.0,18)
LOSS OF ATTITUDE COMMAND (176)	CREW (0,<1) VEHICLE (0.51,90) MISSION (0.67,118)

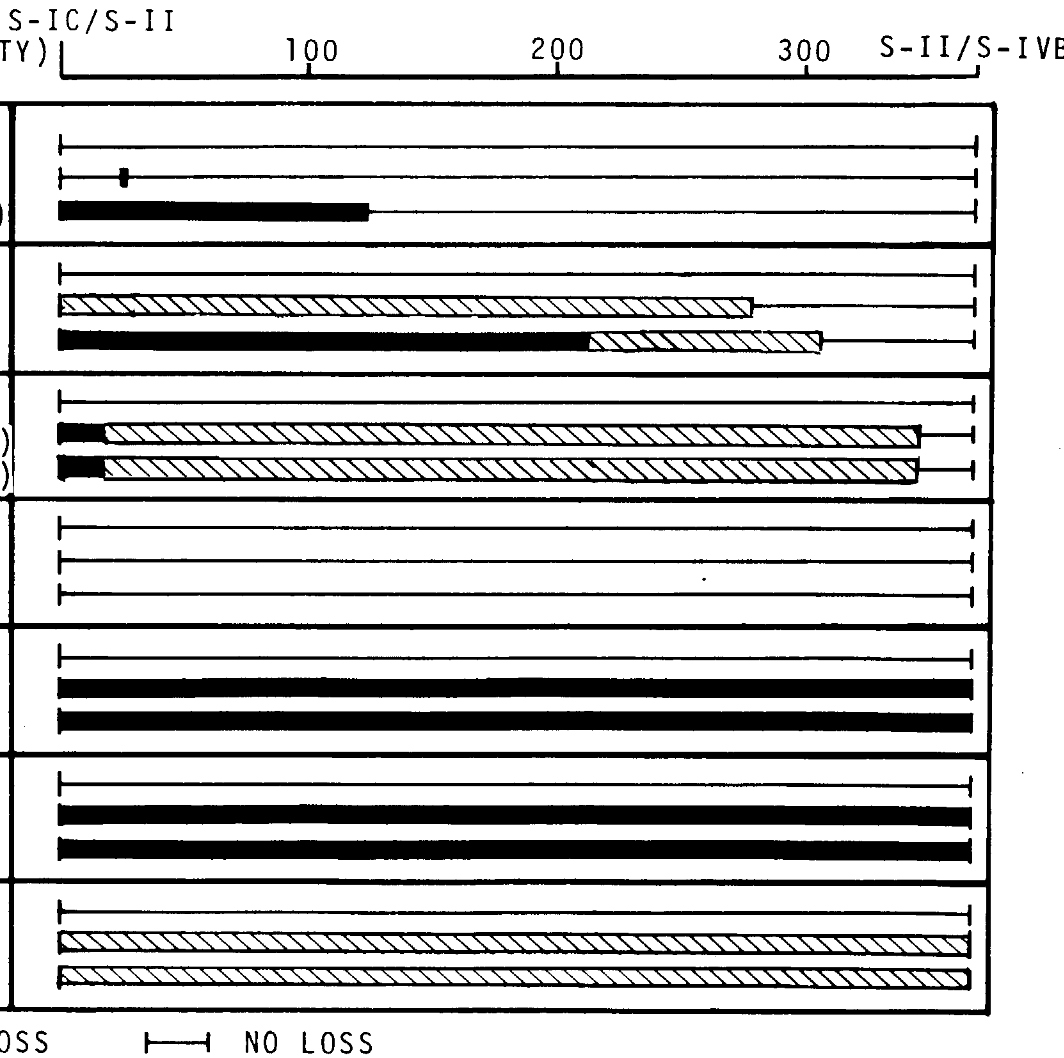
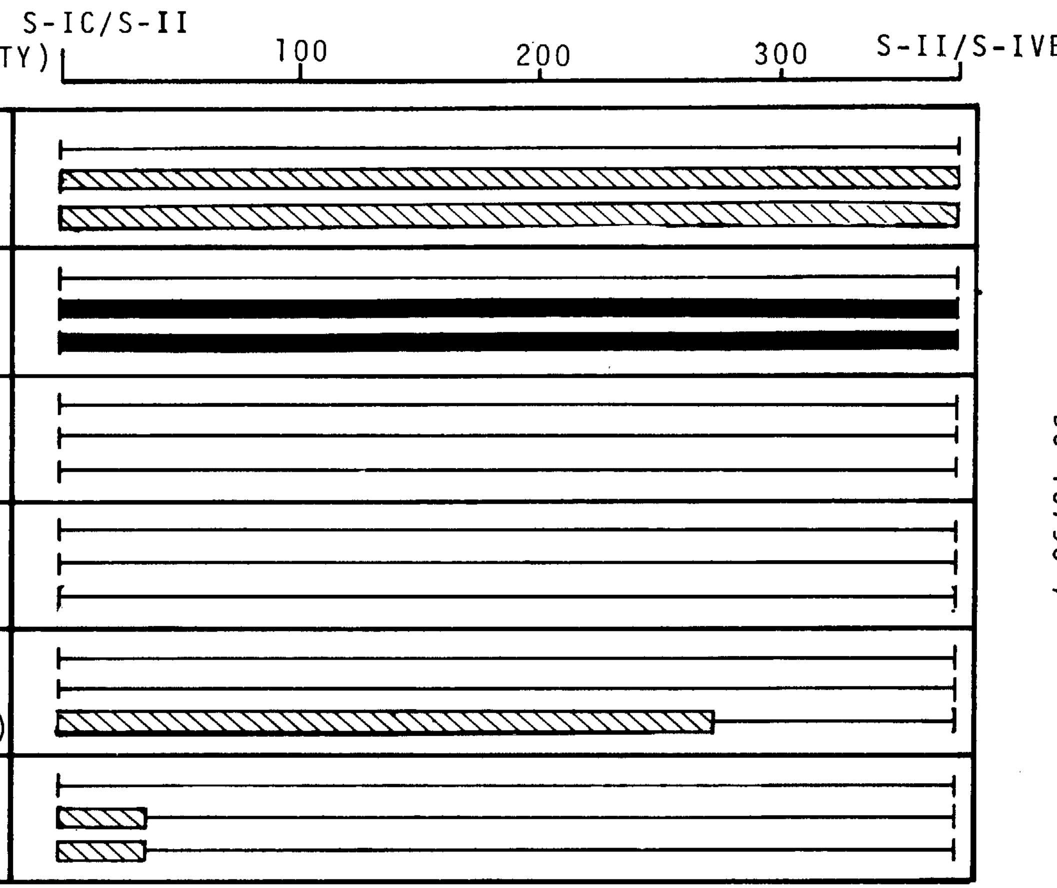


FIGURE 1-3 S-II MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF ATTITUDE ERROR (244)	CREW (0,<1) VEHICLE (0.67,163) MISSION (0.34,83)
LOSS OF ATTITUDE RATE (13)	CREW (0,<1) VEHICLE (1.0,13) MISSION (1,0,13)
ACCELEROMETER FAILURE (1880)	CREW (0,<1) VEHICLE (0,<1) MISSION (0,<1)
LOSS OF * INERTIAL ATTITUDE (1962)	CREW (0,<1) VEHICLE(0,<1) MISSION(0,<1)
PU SYSTEM MALFUNCTIONS (8872)	CREW (0,<1) VEHICLE(0,<1) MISSION(0.39,3460)
SEQUENCING AND STAGING MALFUNCTIONS (1380)	CREW (0,<1) VEHICLE (0.48,662) MISSION (0.48,662)

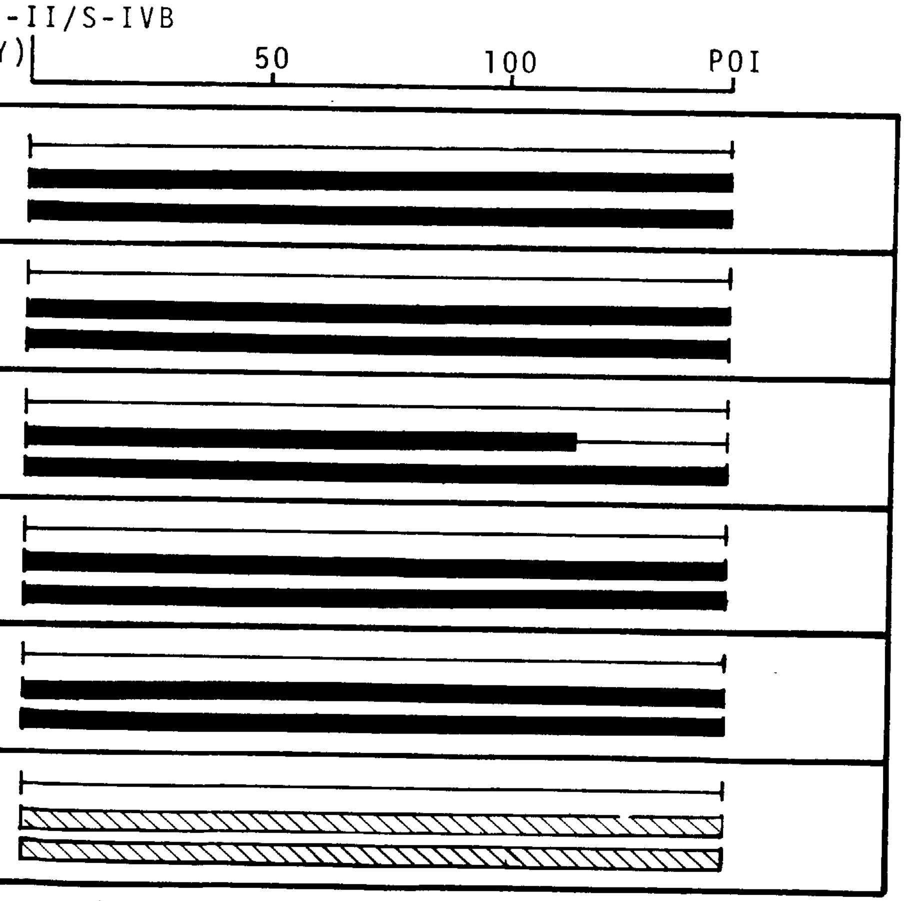


PROBABLE LOSS POSSIBLE LOSS NO LOSS

*Assumes no delay between platform failure and recognition of the failure by the LVDC.

FIGURE 1-3 S-II MALFUNCTION SUMMARY (SHEET 2 OF 2)

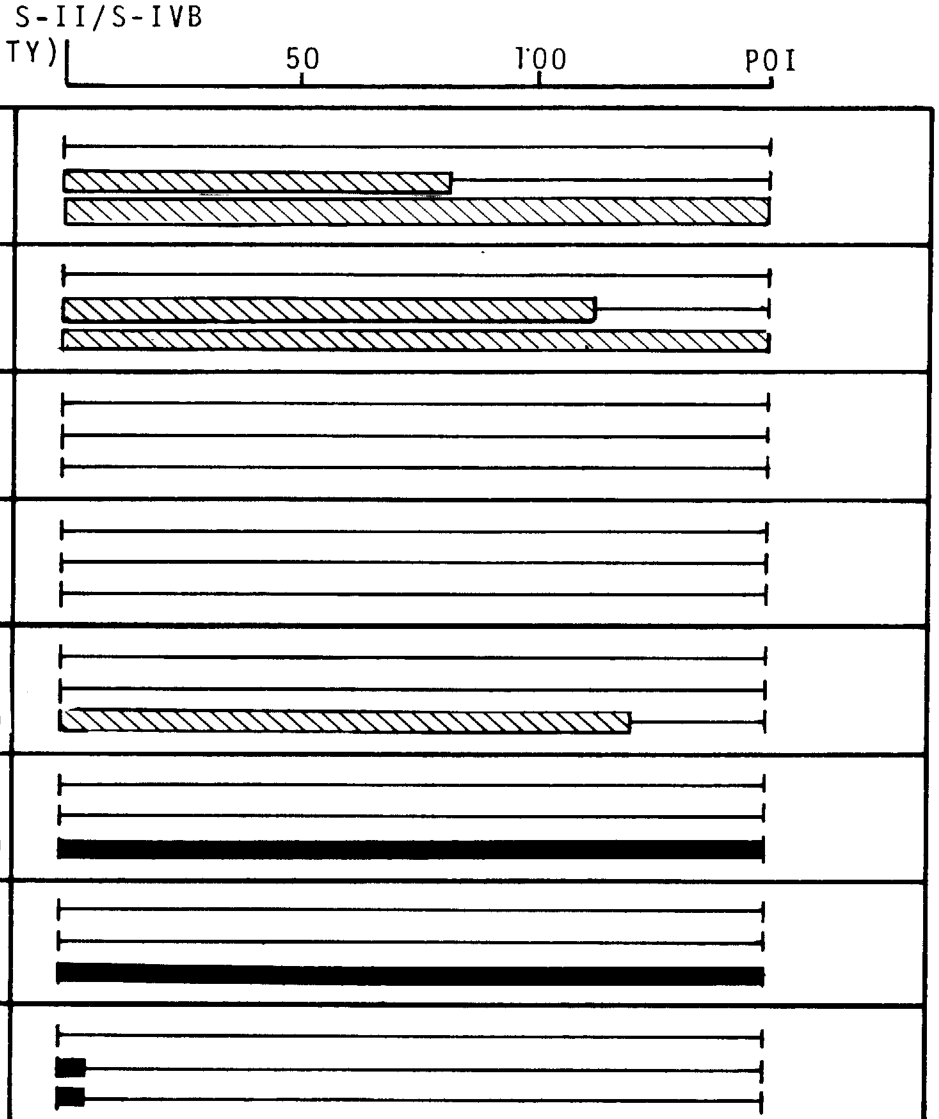
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF THRUST ONE ENGINE (13409)	CREW (0,<1) VEHICLE(1.0,13409) MISSION(1.0,13409)
ONE ACTUATOR HARDOVER (5)	CREW (0,<1) VEHICLE (1.0,5) MISSION (1.0,5)
ONE ACTUATOR INOPERATIVE (241)	CREW (0,<1) VEHICLE (0.83,200) MISSION (1.0,241)
SATURATED ERROR SIGNAL (1)	CREW (0,<1) VEHICLE (1.0, 1) MISSION (1.0, 1)
SATURATED RATE SIGNAL (<1)	CREW (0,<1) VEHICLE (1.0,<1) MISSION (1.0,<1)
LOSS OF ATTITUDE COMMAND (<1)	CREW (0,<1) VEHICLE (0.23,<1) MISSION (0.67,<1)



PROBABLE LOSS   POSSIBLE LOSS   NO LOSS

FIGURE 1-4 S-IVB 1ST BURN MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF ATTITUDE ERROR (95)	CREW (0,<1) VEHICLE (0.37,35) MISSION (0.67,63)
LOSS OF ATTITUDE RATE (<1)	CREW (0,<1) VEHICLE (0.35,<1) MISSION (0.67,<1)
LOSS OF * INERTIAL ATTITUDE (768)	CREW (0,<1) VEHICLE (0,<1) MISSION (0,<1)
ACCELEROMETER FAILURE (471)	CREW (0,<1) VEHICLE (0,<1) MISSION (0,<1)
PU SYSTEM MALFUNCTIONS (3445)	CREW (0,<1) VEHICLE (0,<1) MISSION (0.29,999)
LOSS OF ONE APS MODULE (2400)	CREW (0,<1) VEHICLE (0,<1) MISSION (1.0,2400)
LOSS OF BOTH APS MODULES (<1)	CREW (0,<1) VEHICLE (0,<1) MISSION (1.0,<1)
SEQUENCING AND STAGING MALFUNCTIONS (29)	CREW (0,<1) VEHICLE (0.67,19) MISSION (0.67,19)

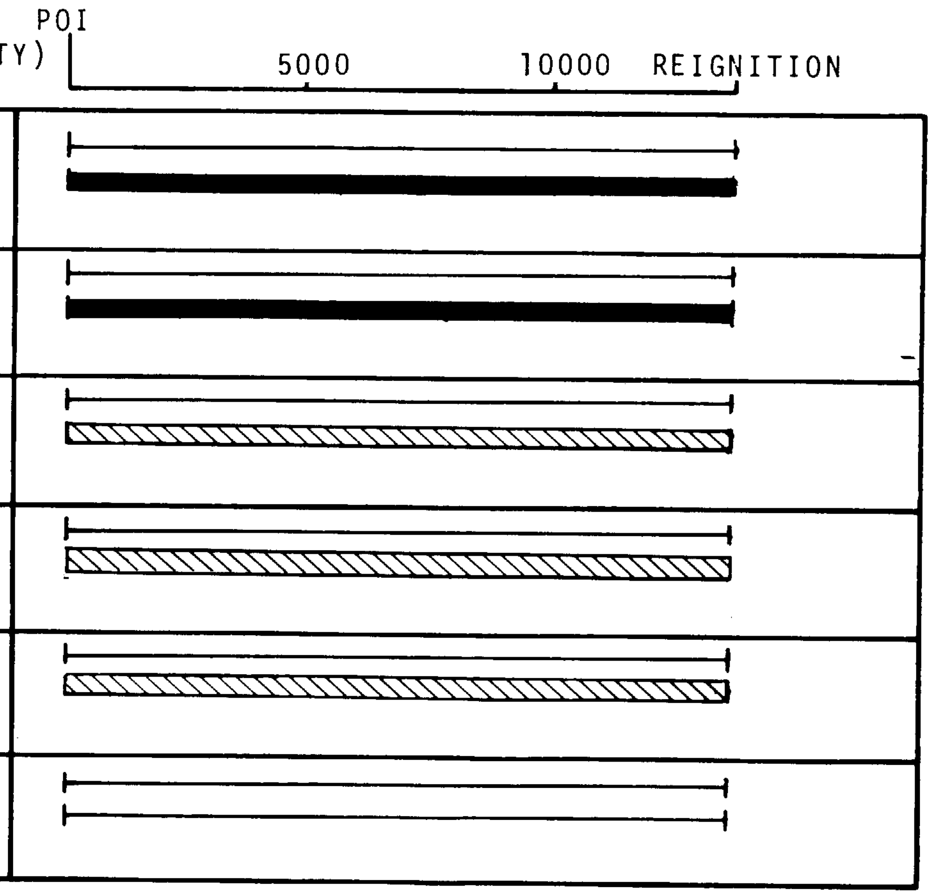


PROBABLE LOSS   POSSIBLE LOSS   NO LOSS

*Assumes no delay between platform failure and recognition of the failure by the LVDC.

FIGURE 1-4 S-IVB 1ST BURN MALFUNCTION SUMMARY (SHEET 2 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
SATURATED ERROR SIGNAL (3)	CREW (0,<1) MISSION (1.0,3)
SATURATED RATE SIGNAL (<1)	CREW (0,<1) MISSION (1.0,<1)
LOSS OF ATTITUDE COMMAND (<1)	CREW (0,<1) MISSION (0.67,<1)
LOSS OF ATTITUDE ERROR (54)	CREW (0,<1) MISSION (0.67,36)
LOSS OF ATTITUDE RATE (<1)	CREW (0,<1) MISSION (0.67,<1)
LOSS OF INERTIAL ATTITUDE (488)	CREW (0,<1) MISSION (0,<1)



PROBABLE LOSS   POSSIBLE LOSS   NO LOSS

FIGURE 1-5 S-IVB PARKING ORBIT COAST MALFUNCTION SUMMARY (SHEET 1 OF 2)

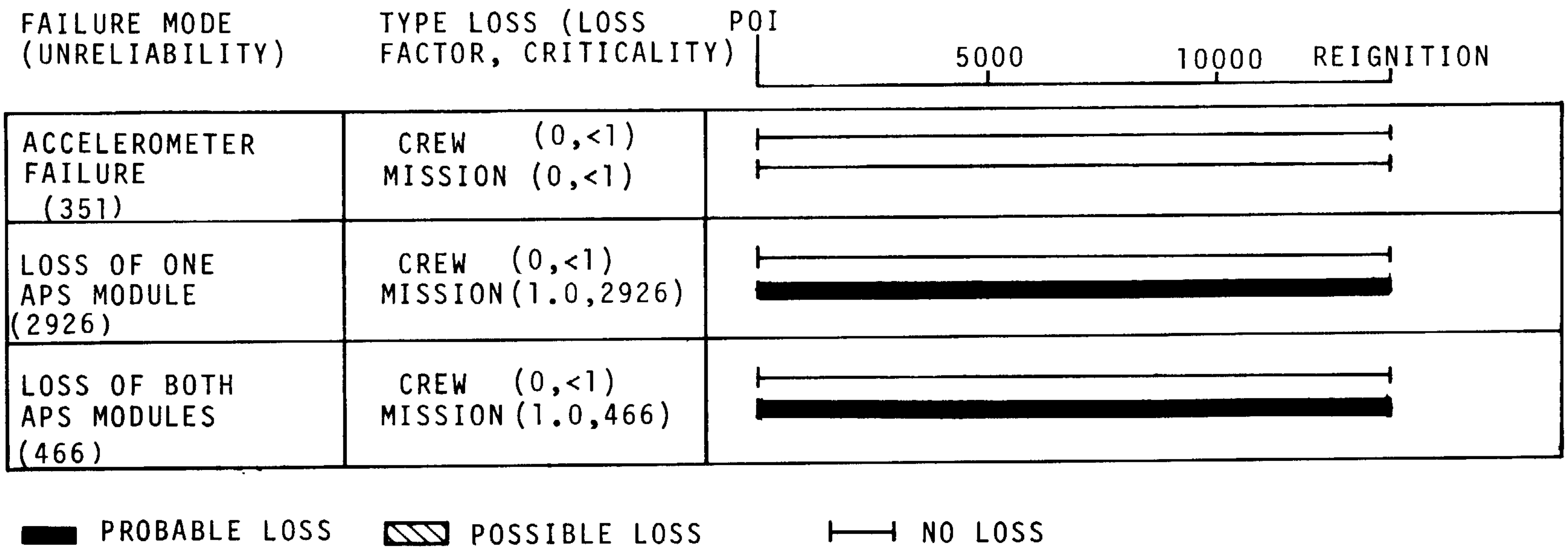
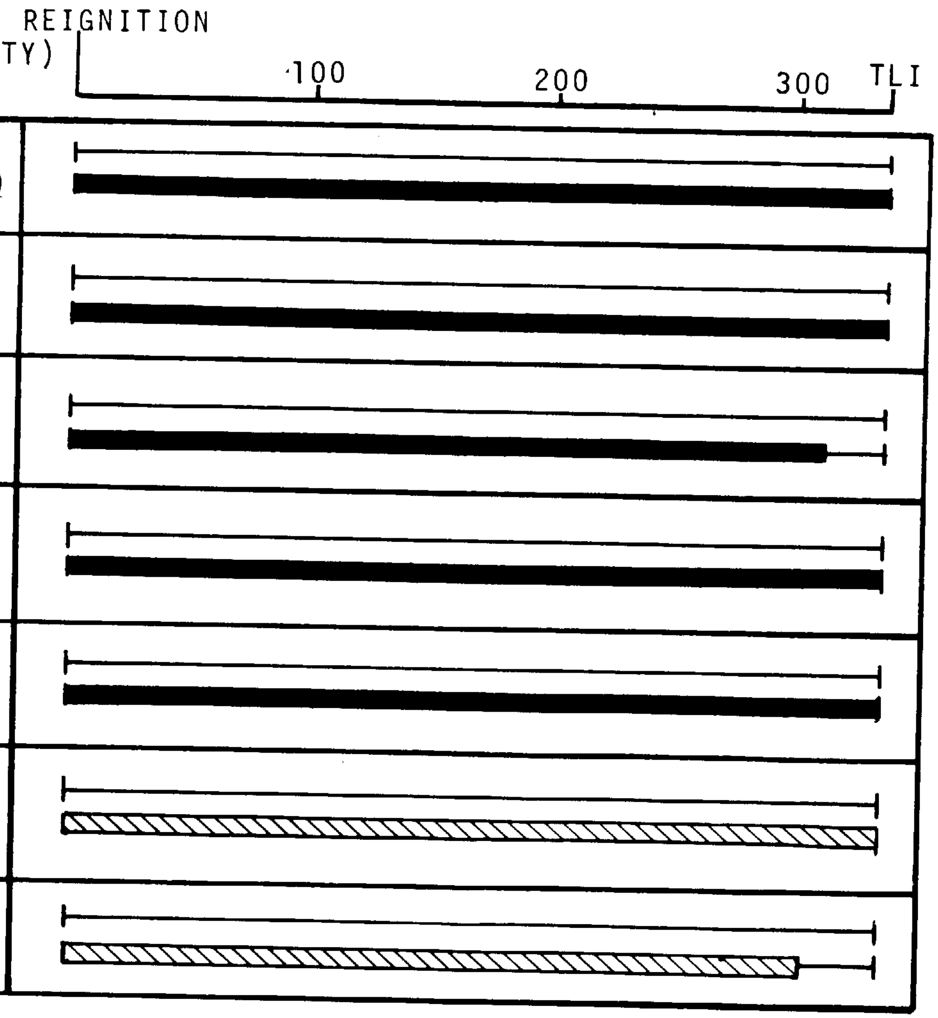


FIGURE 1-5 S-IVB PARKING ORBIT COAST MALFUNCTION SUMMARY (SHEET 2 OF 2)

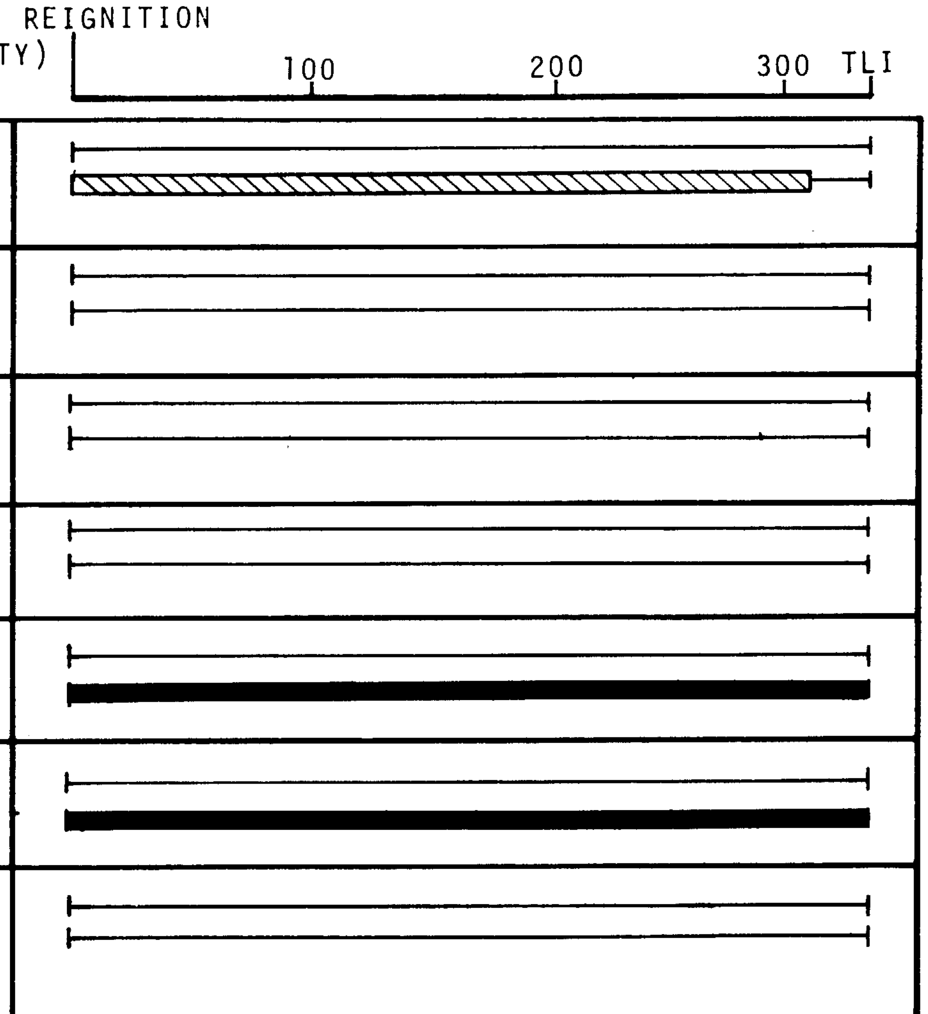
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF THRUST ONE ENGINE (9225)	CREW (0,<1) MISSION (1.0,9225)
ONE ACTUATOR HARDOVER (3)	CREW (0,<1) MISSION (1.0, 3)
ONE ACTUATOR INOPERATIVE (163)	CREW (0,<1) MISSION (0.91,148)
SATURATED ERROR SIGNAL (3)	CREW (0,<1) MISSION (1.0,3)
SATURATED RATE SIGNAL (<1)	CREW (0,<1) MISSION (1.0,<1)
LOSS OF ATTITUDE COMMAND (<1)	CREW (0,<1) MISSION (0.30,<1)
LOSS OF ATTITUDE ERROR (115)	CREW (0,<1) MISSION (0.61,70)



PROBABLE LOSS   POSSIBLE LOSS   NO LOSS

FIGURE 1-6 S-IVB 2ND BURN MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
LOSS OF ATTITUDE RATE (<1)	CREW (0,<1) MISSION (0.53,<1)
LOSS OF * INERTIAL ATTITUDE (1712)	CREW (0,<1) MISSION (0,<1)
ACCELEROMETER FAILURE (1076)	CREW (0,<1) MISSION (0,<1)
PU SYSTEM MALFUNCTIONS (5511)	CREW (0,<1) MISSION (0,<1)
LOSS OF ONE APS MODULE (1200)	CREW (0,<1) MISSION (1.0,1200)
LOSS OF BOTH APS MODULES (<1)	CREW (0,<1) MISSION (1.0,<1)
SEQUENCING AND STAGING MALFUNCTIONS (40)	CREW (0,<1) MISSION (0,<1)



PROBABLE LOSS ▨ POSSIBLE LOSS ⊢⊣ NO LOSS

*Assumes no delay between platform failure and recognition of the failure by the LVDC.

FIGURE 1-6 S-IVB 2ND BURN MALFUNCTION SUMMARY (SHEET 2 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)
SATURATED ERROR SIGNAL (1)	CREW (0,<1) MISSION (1.0, 1)
SATURATED RATE SIGNAL (<1)	CREW (0,<1) MISSION (1.0,<1)
LOSS OF ATTITUDE COMMAND (<1)	CREW (0,<1) MISSION (0,<1)
LOSS OF ATTITUDE ERROR (29)	CREW (0,<1) MISSION (0,<1)
LOSS OF ATTITUDE RATE (<1)	CREW (0,<1) MISSION (0,<1)
LOSS OF INERTIAL ATTITUDE (265)	CREW (0,<1) MISSION (0,<1)

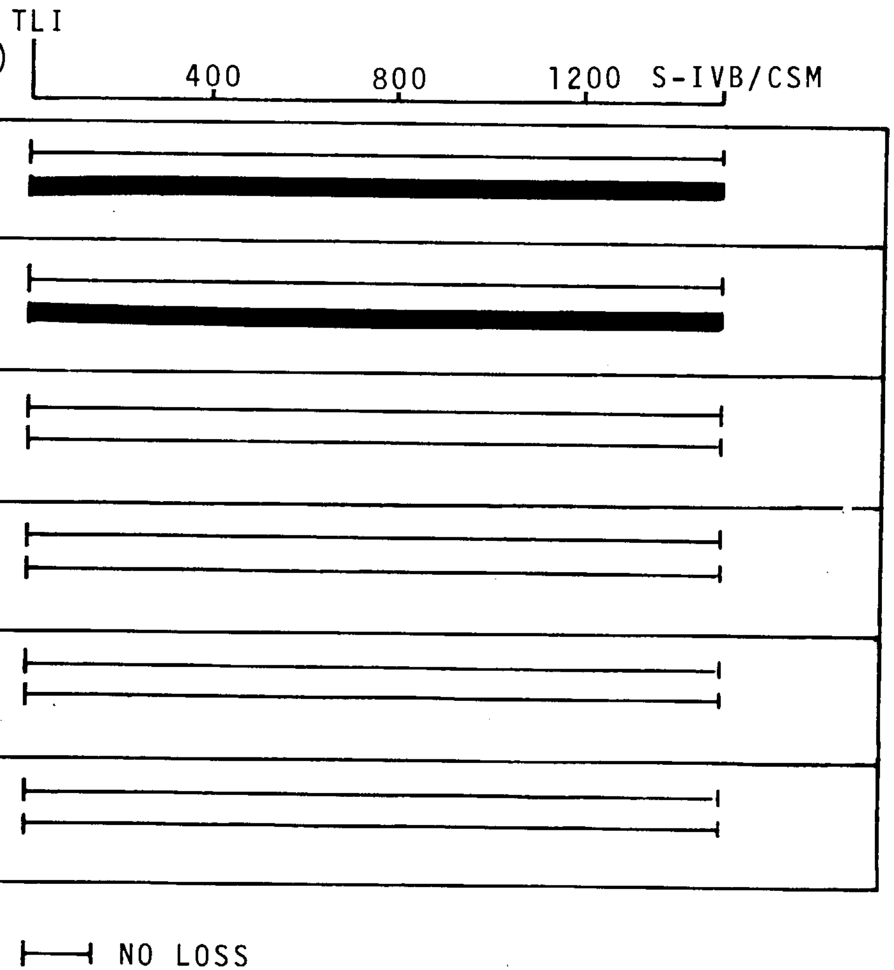


FIGURE 1-7 S-IVB TRANSLUNAR COAST MALFUNCTION SUMMARY (SHEET 1 OF 2)

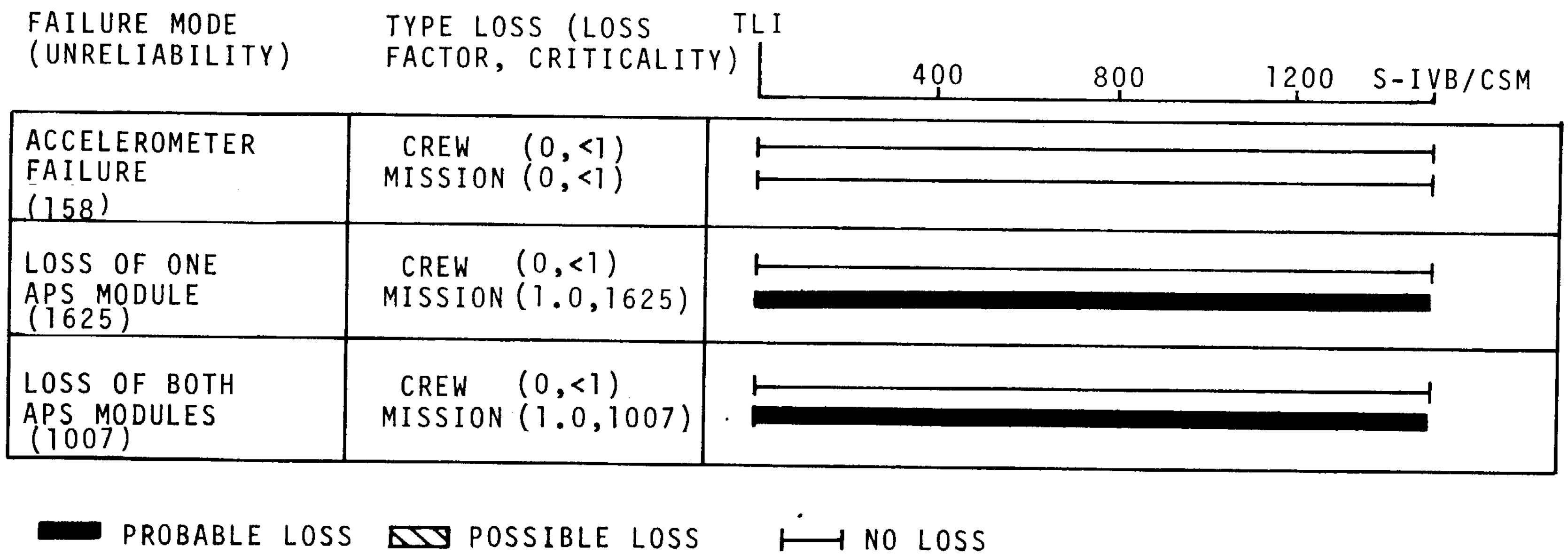


FIGURE 1-7 S-IVB TRANSLUNAR COAST MALFUNCTION SUMMARY (SHEET 2 OF 2)

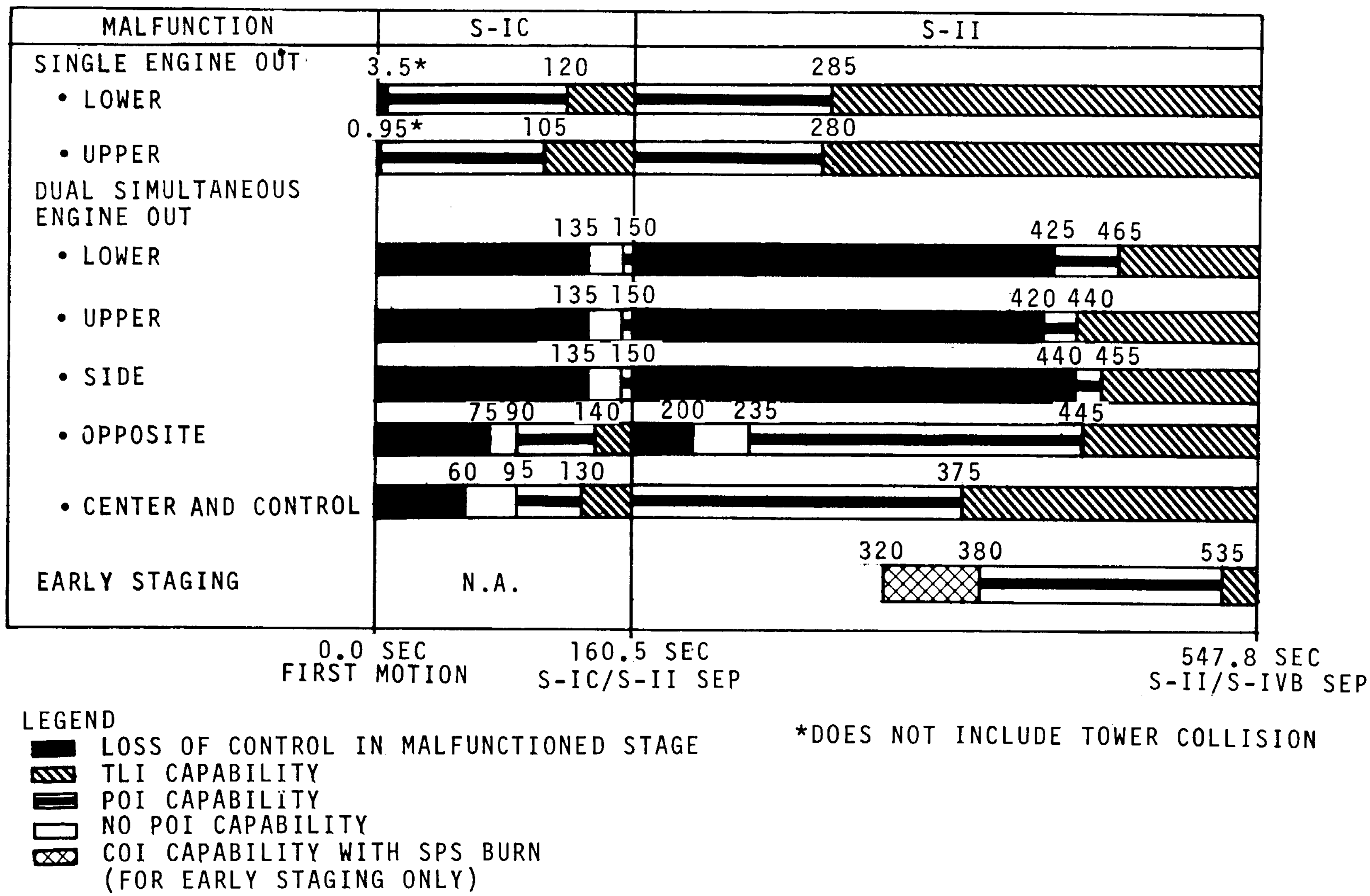


FIGURE 1-8 SUMMARY OF ENGINE OUT AND EARLY STAGING CAPABILITY

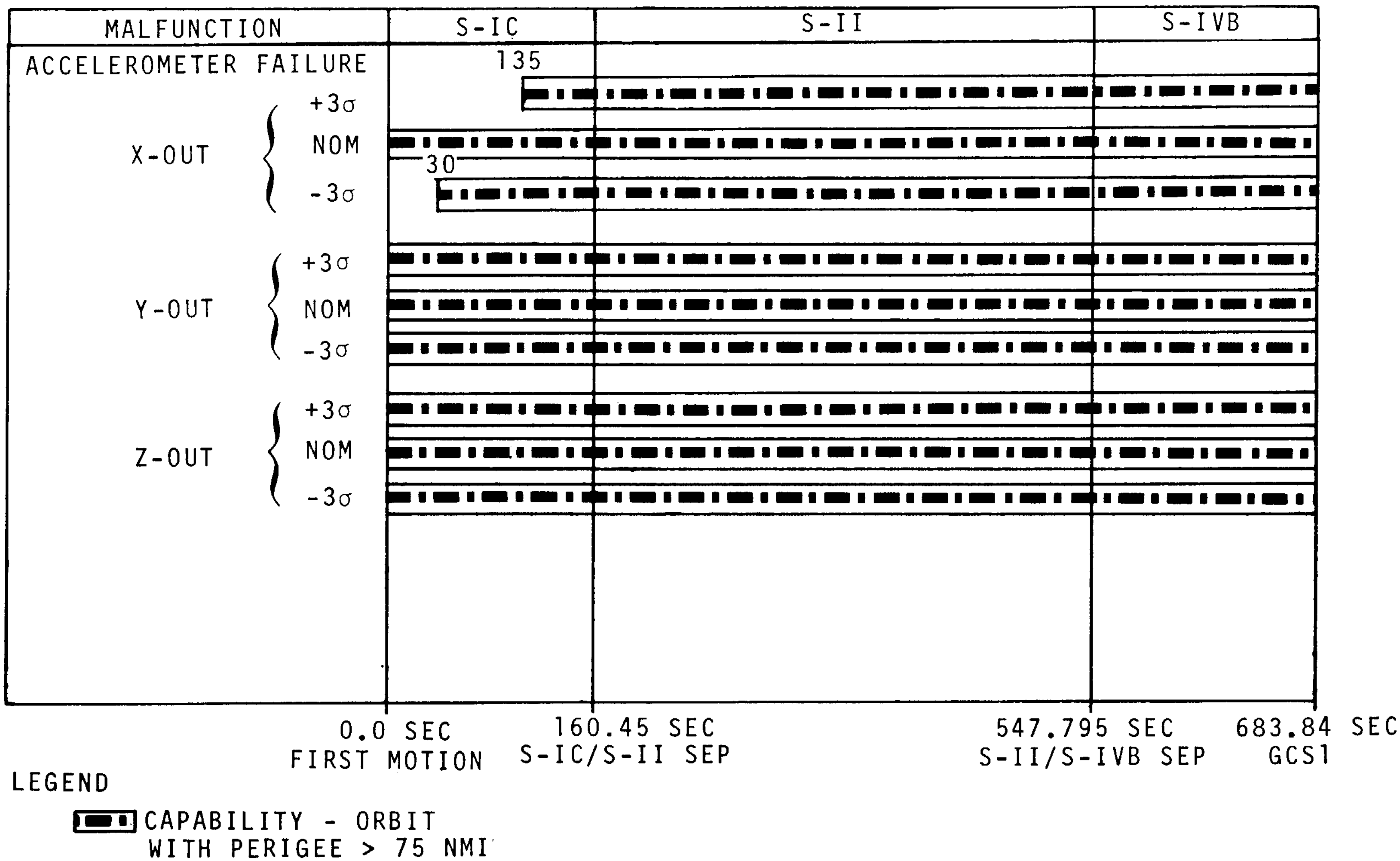


FIGURE 1-9 SUMMARY OF ACCELEROMETER FAILURE CAPABILITY

SECTION 2

EMERGENCY DETECTION SYSTEM DESCRIPTION

The Saturn V Emergency Detection System (EDS) is designed to provide crew safety in the event of malfunctioned flight. The system monitors critical flight parameters and furnishes advance warning of impending emergency conditions. If abort is required, it is initiated either automatically or manually depending on the nature and time of the malfunction.

In the following paragraphs, the EDS system is described in terms of EDS displays, EDS controls, abort controls, and abort modes.

2.1 EMERGENCY DETECTION SYSTEM DISPLAYS

The EDS displays are selected to present parameters which indicate failures leading to vehicle abort. Automatic abort parameters are implemented triple redundant, voted two-out-of-three, to preclude single point hardware or sensing failures causing an inadvertant abort. Manual abort parameters are implemented with redundant sensing and displays to provide highly reliable indications to the crew.

Displays are designed to provide onboard detection capability for rapid rate malfunctions which may require abort. Pilot abort action must be based on two separate but related abort cues. These cues may be derived from the EDS displays, ground information, physiological cues, or any combination of two valid cues. The EDS displays and controls referred to throughout this document are shown in Figure 2-1. As each is discussed, it is identified by use of grid designators listed on the border of the figure.

2.1.1 Flight Director Attitude Indicator (FDAI)

There are two Flight Director Attitude Indicators, each of which provides a display of Euler attitude, attitude errors and angular rates. These displays are active at liftoff and remain active throughout the mission, except that attitude errors are not displayed during S-II and S-IVB flight. The FDAI's are used to monitor normal launch vehicle guidance and control events. The pilot's FDAI is shown in Figure 2-1, I-9.

The FDAI ball displays the Euler attitudes, the needle type pointers across the face of the ball indicate attitude errors, and the triangular pointers around the periphery of the ball display angular rates.

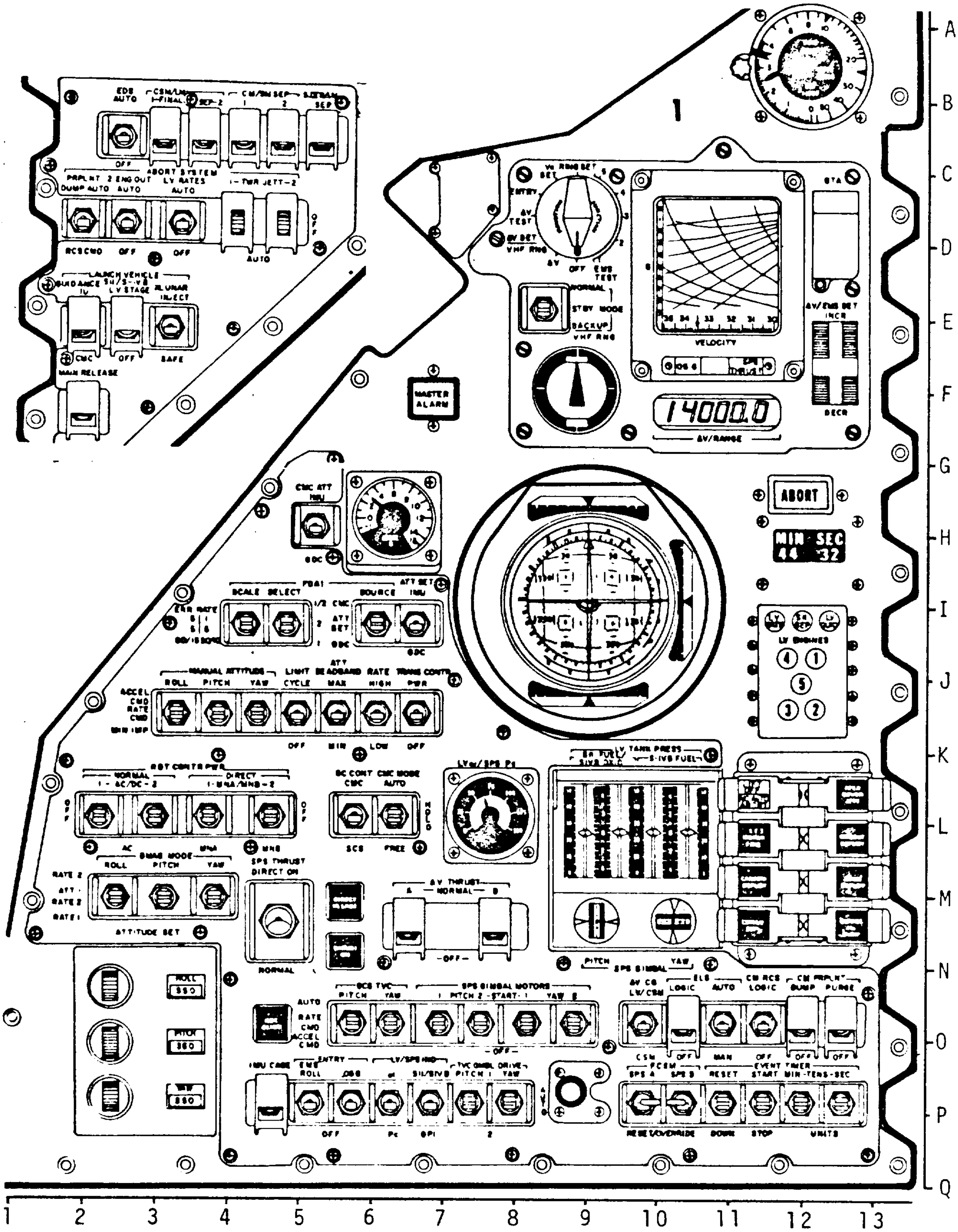


FIGURE 2-1 EDS DISPLAY AND CONTROL PANEL

2.1.1 (Continued)

Signal inputs to the FDAI's are switch selectable and can come from a number of different sources in the spacecraft. This flexibility and redundancy provide the required attitude and error backup display capability.

2.1.2 LV ENGINES Lights

Each of the five LV ENGINES lights shown in Figure 2-1, J-12, represents the respective numbered engine on the operating stage. A light ON indicates its corresponding engine is operating below a nominal thrust level (90 percent on F-1 engines and 65 percent of J-2 engines). During staging all lights are turned OFF momentarily to indicate physical separation has occurred.

2.1.3 LV RATE Light

The LV RATE light (Figure 2-1, I-12) when ON, is the primary cue from the launch vehicle that the following preset over-rate settings have been exceeded:

Pitch/Yaw	4.0 (±0.5) deg/sec	Liftoff to automatic abort deactivation (120 seconds)
	9.2 (±0.8) deg/sec	Automatic abort deactivation to S-IVB cutoff
Roll	20.0 (±0.5) deg/sec	Liftoff to S-IVB cutoff.

In the event the LV GUID light is illuminated during the automatic abort phase, the LV RATE light will be illuminated as a redundant indicator.

2.1.4 LV GUID Light

Launch vehicle attitudes are measured and provided to the launch vehicle digital computer every 40 milliseconds. The computer checks the attitude for reasonableness. If the reasonableness tests fail, the attitude error signals to the flight control computer are frozen and the LV GUID light, shown in Figure 2-1, I-12, is illuminated.

2.1.5 ABORT Light

The ABORT light, shown in Figure 2-1, G-12, may be illuminated by ground command from the Flight Director, the Mission Control Center (MCC) Booster Systems Engineer, the Flight Dynamics Officer (FDO), the Complex 39 Launch Operations Manager (until tower clearance +10 seconds), or in conjunction with range safety booster engine cutoff.

2.1.6 Angle-of-Attack Meter

The angle-of-attack ($q\alpha$) meter, shown in Figure 2-1, L-8, is time shared with the Service Propulsion System (SPS) chamber pressure. The $q\alpha$ display is a pitch and yaw vector summed angle-of-attack/dynamic pressure product. It is expressed in percentage of total pressure for predicted launch vehicle breakup (abort limit equals 100 percent). It is effective as an abort parameter only during the high q flight region. During other portions of boost through the atmosphere, the $q\alpha$ meter provides trend information on launch vehicle flight performance and provides a secondary cue for slow-rate guidance and control malfunctions.

2.1.7 Accelerometer

The accelerometer, shown in Figure 2-1, H-6, indicates longitudinal acceleration/deceleration in G's. It provides a secondary cue for certain engine failures and is a gross indicator of launch vehicle performance.

2.1.8 Event Timer

The event timer, shown in Figure 2-1, H-12, is a digital clock which displays time from liftoff. It is a critical display because it is the primary cue for the transition of abort modes, manual sequenced events, monitoring roll and pitch program, staging, and S-IVB insertion cutoff.

The event timer is reset to zero automatically with abort initiation.

2.2 EMERGENCY DETECTION SYSTEM CONTROLS

The main EDS control switches are the EDS, the 2 ENG OUT, and the LV RATES. They are two-position (AUTO and OFF) toggle switches which are placed in the AUTO position prior to liftoff. When all three switches are in the AUTO position, automatic abort is initiated if:

a. A LV structural failure occurs between the Instrument Unit and Command Service Module.
b. Two or more S-IC engines drop below 90 percent thrust.

2.2 (Continued)

c. LV rates exceed the EDS limits.

While these switches can be manually disabled at any time by placing them in the OFF position, the normal procedure requires disabling at 120 seconds. They are automatically disabled by the LV sequencer just prior to center engine cutoff.

2.3 ABORT CONTROLS

The translational controller (T-Handle) mounted on the left arm of the commander's couch is used to initiate abort. A manual Launch Escape System (LES) abort sequence is initiated by rotating the T-Handle fully counterclockwise. This sends redundant engine cutoff commands to the LV, initiates Command Module/Service Module separation, fires the LES motors, resets the spacecraft sequencer and initiates the post abort sequence. (Engine cutoff from the spacecraft is inhibited during the first 30 seconds of flight).

For a manually initiated SPS abort after the Launch Escape Tower has been jettisoned, counterclockwise rotation of the T-Handle commands LV cutoff, resets the spacecraft sequencer and initiates the Command Service Module/Launch Vehicle separation sequence. However, returning the T-Handle to neutral before 3 seconds expires results in only a LV cutoff rather than the full abort sequence.

2.4 ABORT MODES

Aborts during boost are performed using either the LES or the SPS.

2.4.1 LES Aborts (Modes 1A, 1B, and 1C)

The LES consists of a solid propellant launch escape motor used to propel the CM a safe distance from the launch vehicle, a tower jettison motor, and a canard subsystem. LES abort modes are as follows:

a. Mode 1A: Low Altitude Mode (Pad to 42 Seconds)

In Mode 1A a pitch control motor, mounted normal to the launch escape motor, propels the vehicle downrange to ensure water landing and escape from the "fireball." The automatic sequence of major events following abort initiation is as follows:

2.4.1 (Continued)

Time	Event
00:00	Abort, SM Reaction Control System (RCS) Oxidizer Rapid Dump, Launch Escape & Pitch Control Motors Fire
00:05	RCS Fuel Rapid Dump
00:11	Canards Deploy
00:14.4	Apex Cover Jettison
00:16	Drogue Deploy
00:28	Main Chute Deploy

The automatic sequence can be prevented, interrupted, or replaced by crew action.

b. Mode 1B: Medium Altitude Mode (42 Seconds to 100,000 Feet)

Mode 1B is essentially the same as Mode 1A with the exception of deleting rapid RCS propellant dump and PC motor fire. The canard subsystem is designed specifically for this altitude region to initiate a tumble in the pitch plane. Upon closure of barometric switches at 24,000 feet, the tower is jettisoned. The main parachutes are automatically deployed at 10,000 feet.

c. Mode 1C: High Altitude Mode (100,000 Feet to Tower Jettison)

During Mode 1C the launch vehicle is above the atmosphere and therefore the canard subsystem cannot be used to induce a pitch rate to the escape vehicle. If the launch vehicle is stable at abort, the LET is manually jettisoned and the CM oriented to the reentry attitude. This method requires a functioning attitude reference system.

With a failed attitude reference system the alternate method is to introduce a 5 degree per second pitch rate using the attitude control thrusters. The CM/tower combination will then stabilize blunt end forward as in Mode 1B. The LES then deploys the parachutes at the proper altitude.

2.4.2 SPS Aborts (Modes II, III, and IV)

The SPS aborts utilize the Service Module SPS engine to propel the CSM combination away from the launch vehicle, maneuver for reentry to a planned landing area, or boost into a contingency orbit. The SPS abort modes are as follows:

2.4.2 (Continued)

a. Mode II

The SM Reaction Control System engines are used to propel the CSM away from the launch vehicle unless the vehicle is in danger of exploding or excessive tumble rates are present at LV/CSM separation. In these two cases the SPS engine would be used due to greater ΔV and attitude control capability. When at a safe distance, the CM is separated from the SM and maneuvered to a reentry attitude.

b. Mode III

The SPS engine is used to slow the CSM combination so as to land at a predetermined point in the Atlantic Ocean. CM/SM separation then occurs and normal reentry procedures follow.

c. Mode IV

The SPS engine can be used to make up for a deficiency in insertion velocity up to approximately 3000 feet per second. This is accomplished by holding the CSM in an inertial attitude and applying the needed ΔV with the SPS to acquire the acceptable orbital velocity.

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SECTION 3

MALFUNCTION ANALYSIS CRITERIA

Malfunction analysis criteria consists of:

a. Abort criteria
b. Crew, vehicle, and mission loss criteria
c. Abort cues and EDS limits

3.1 ABORT CRITERIA

Abort criteria are those criteria which are used to establish the need for abort in the event of a vehicle malfunction. The criteria are:

a. Launch platform interference, pad fallback, and tower collision.
b. Controllability
c. Structural capability
d. Range safety limits
e. Staging limits
f. Spacecraft heating limit
g. Platform yaw gimbal limit
h. Safe abort limits

Conditions for a safe abort must satisfy the following requirements and limits:

a. Water impact
b. Abort lead time requirements (Figure 3-1)
c. LEV α-limit
d. 16g reentry limit
e. 100 second free fall limit
f. Spacecraft platform tumble limit (90 degrees yaw)

3.2 CREW, VEHICLE, AND MISSION LOSS CRITERIA

3.2.1 Crew Loss

Crew loss is assumed to occur if the abort lead time is less than the required lead time, or if 90 degrees yaw attitude occurs during abort. Abort lead time requirements during S-IC flight are taken from Reference 8 and shown in Figure 3-1.

Crew loss factors (β's) are calculated as follows:

$$\beta = \frac{(\Delta T_{LOSS})(P_{LOSS})}{\text{STAGE FLIGHT TIME}}$$

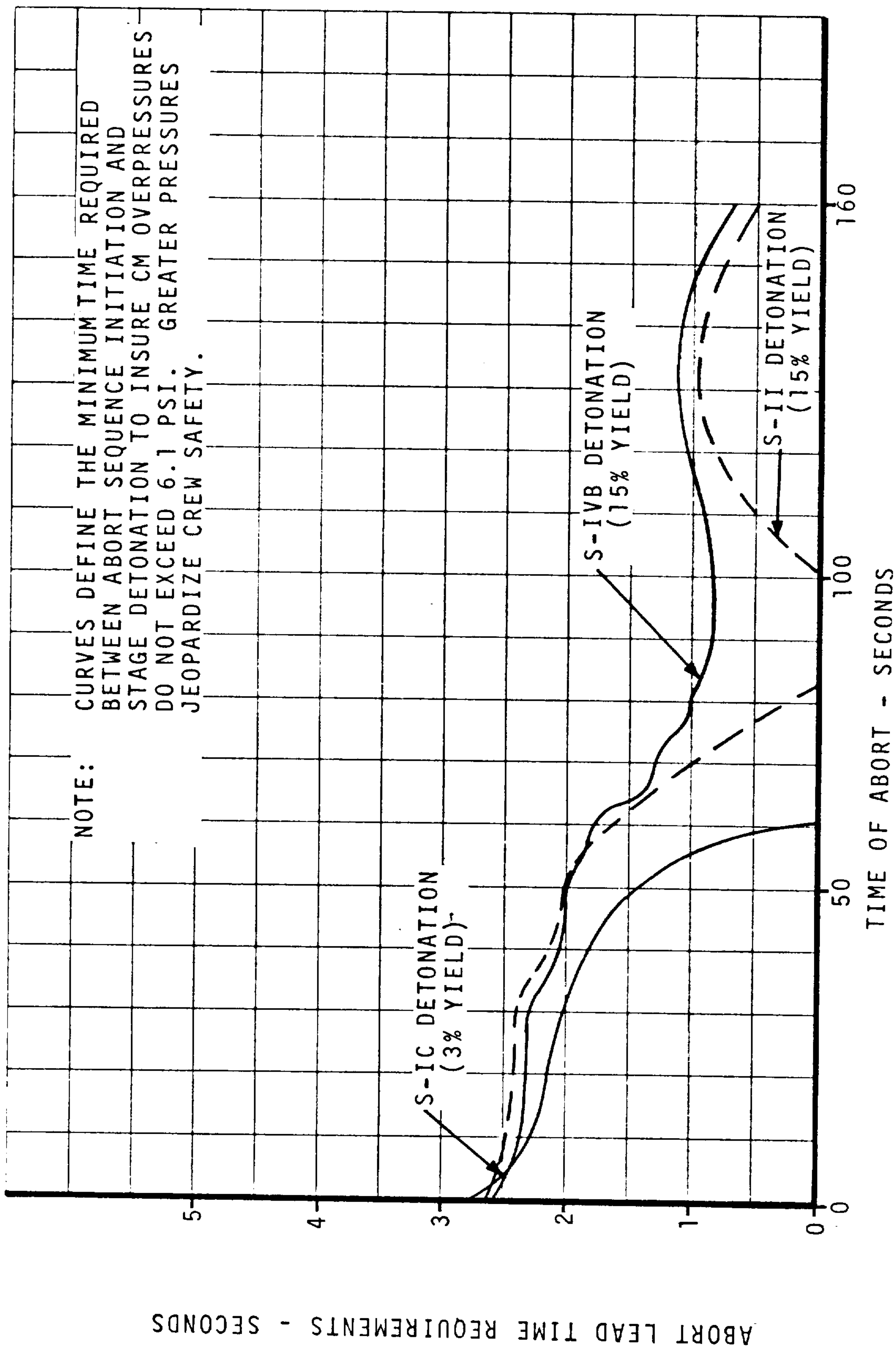

FIGURE 3-1 ABORT LEAD TIME REQUIREMENTS

3.2.1 (Continued)

where: ΔT_{LOSS} = time interval in which crew loss occurs due to a particular malfunction

P_{LOSS} = probability that the particular malfunction will cause crew loss; i.e., if, considering single engine failures in S-IC, any control engine failure causes crew loss, then $P_{LOSS} = 4/5$; or if only a particular engine causes loss of control, then $P_{LOSS} = 1/5$

In case of crew loss β's, it is assumed that the probability of an explosion within one second following breakup is 0.2.

Criticality numbers are calculated as follows:

$$CN = \beta \cdot UN$$

where UN (unreliability) is defined as the probable number of occurrences of a failure mode per one million flights.

A detailed explanation of loss factor determination is given in Reference 2. Unreliability values used in this analysis are given in References 3 and 4.

3.2.2 Vehicle Loss

Vehicle loss is assumed to occur if the vehicle or spacecraft structural capability is exceeded before the vehicle or payload achieves at least a 75 NMI perigee orbit. Situations causing structural failure are:

a. Vehicle collision with a solid object, e.g., liftoff interference with launch platform and holddown posts, tower collision, pad area fallback, and collision between separating stages following staging.

b. Excessive aerodynamic forces due to loss of control or trajectory deviations which lead to atmospheric reentry.

c. Structural dynamic response following a malfunction and abort cutoff.

Vehicle loss factors are calculated in the same manner as crew loss factors.

3.2.3 Mission Loss

Launch vehicle primary mission loss is assumed if, at TLI plus 3 hours, a ΔV correction greater than 800 feet per second is required.

Mission loss factors are calculated in the same manner as crew loss factors.

3.3 ABORT CUES AND EDS LIMITS

Automatic abort occurs during the first 120 seconds of flight when:

a. A vehicle overrate is measured by two of the three EDS rate gyros in each axis. An overrate is 4 degrees per second in pitch or yaw, or 20 degrees per second in roll.

b. Two engines are below 90 percent thrust as indicated by two of the three "Thrust-OK" pressure switches on each engine.

c. A structural failure occurs between the Instrument Unit and the Command Module as indicated by two out of three structural wires.

Manual abort will be initiated by the crew using the abort cues and EDS limits shown in Table 3-I. Two independent cues measured and indicated by separate systems are necessary to prevent false abort.

TABLE 3-I MANUAL ABORT CUES AND EDS LIMITS

STAGE	FLIGHT TIME (SEC)	ABORT CUE	EDS LIMIT
S-IC	$0<t<50$	Engine Status Lights Voice Request Abort Request Light	
	$50<t<120$	Engine Status Lights Attitude Error Q-Ball ΔP	 ±5 deg 3.2 PSID (100%)
	$120<t<OBECO$	Engine Status Light L/V Rate Light S/C Rate Indicator: Roll Pitch or Yaw Attitude Error	 ±20 deg/sec ±10 deg/sec ±5 deg*
S-II & S-IVB	All Times	S/C Rate Indicator: Roll Pitch or Yaw FDO Display Abort Request Light L/V Rate Light Engine Status Lights Attitude Deviation Yaw Attitude Voice Request	 ±20 deg/sec ±10 deg/sec Limit Exceeded ±20 deg ±45 deg

*Recommended EDS limit for dual engine out malfunction occurring after CECO is ±20 degrees.

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SECTION 4

MALFUNCTIONED FLIGHT AND ABORT ANALYSIS

The objectives of this analysis are to:

a. Determine that the Emergency Detection System protects the crew following vehicle malfunctions

b. Determine the mission completion capability following vehicle malfunctions.

The scope of the analysis is limited to the evaluation of malfunctioned flight for a G mission with a September 13, 1969, 78.051 degree launch azimuth, and first opportunity translunar injection.

The points of failure for the various malfunctions considered in this analysis are shown in Figure 4-1.

Except for the effects of accelerometer malfunctions, all effects of malfunctions are evaluated for a reference vehicle. Accelerometer malfunction effects are evaluated for ±3-sigma vehicles.

The AS-507 S-IC pitch polynomial is biased for the average 50 percentile September/October/November winds. For malfunctions where winds have a significant effect, the vehicle is flown in design winds with magnitudes based on the maximum 95 percentile September/October wind. The gust is phased with the malfunction to establish a worst case. For malfunctions where winds do not have a significant effect, the average 50 percentile September/October wind from 258 degrees is used.

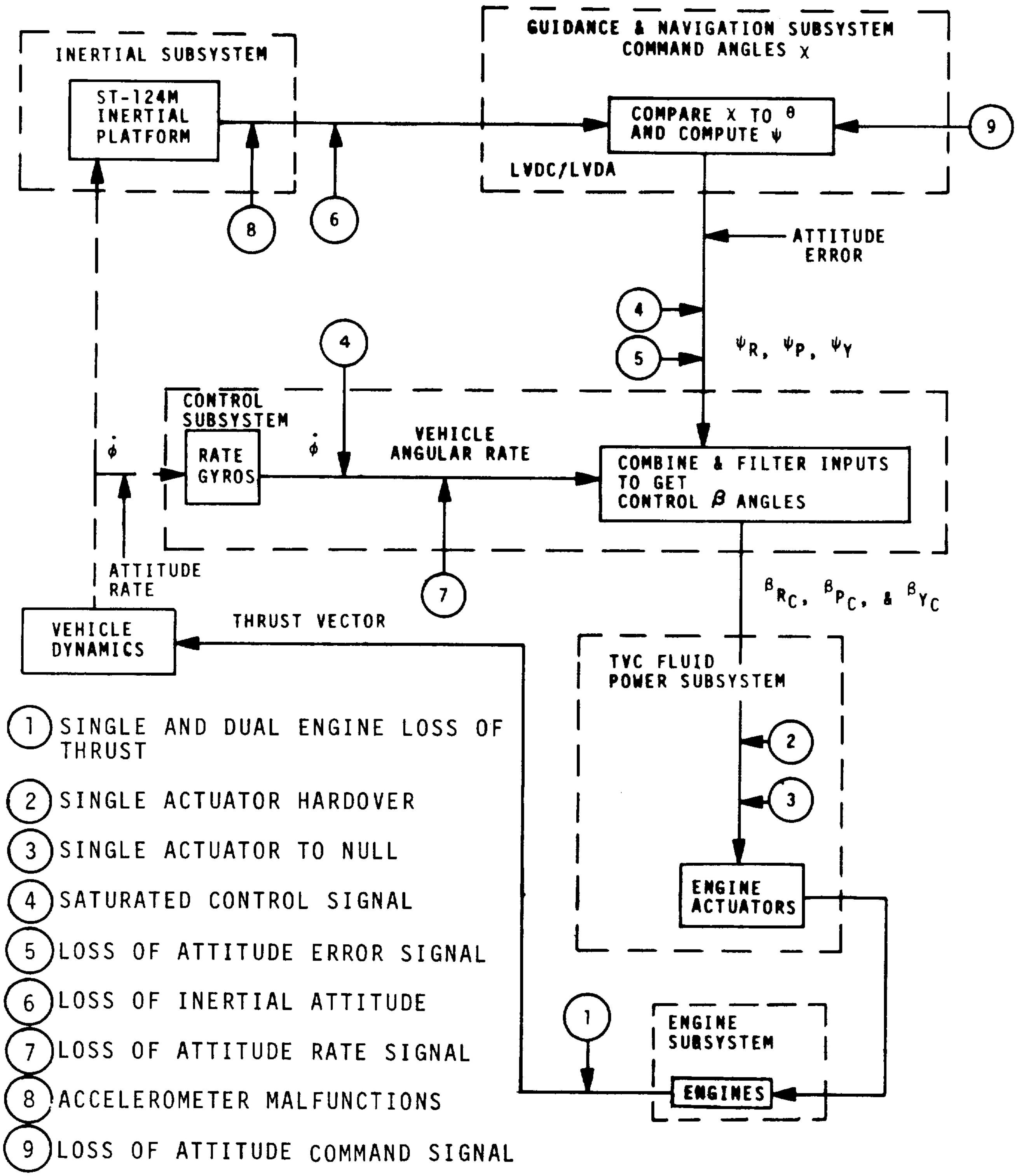


FIGURE 4-1 SATURN V GUIDANCE AND CONTROL FAILURE POINTS

4.1 SINGLE ENGINE LOSS OF THRUST

4.1.1 Malfunction Description

Single engine loss of thrust malfunctions are categorized as follows:

a. Thrust loss following a shutdown command initiated by the thrust OK pressure switches (TOPS). A TOPS shutdown occurs when some malfunction causes propellant inlet pressure, and ultimately thrust, to drop below 90 percent of rated thrust.

b. Thrust loss resulting from an inadvertant shutdown command initiated by an electrical malfunction.

c. Sudden thrust loss resulting from a catastrophic failure such as engine explosion.

Failure of a J-2 engine to start during S-II and S-IVB flight is also considered in this analysis.

Thrust decay characteristics resulting from TOPS dropout are shown in Reference 5 . TOPS dropout is inhibited until TB1 + 14 seconds, after which TOPS dropout occurs automatically when the thrust decreases below 90 percent of rated thrust.

The S-IC engine-out guidance χ-freeze schedule used in this study is shown in Figure 4-2.

4.1.2 Malfunction Dynamics

Any engine out prior to 0.2 second results in pad fallback. Any engine out between 0.2 and 0.9 second results in the vehicle colliding with the holddown posts. Tower collision occurs for a Number 1 or 2 engine out prior to 5.7 seconds. Number 4 engine out before 3.5 seconds results in loss of control in the late high-q region due to vehicle aerodynamic instability and eventual loss of control authority. The max-q region occurs between 110 and 130 seconds for these very early failures. Figure 4-3 shows a time history of dynamics for a typical early single engine failure.

Figure 4-4 shows that for single engine-out malfunctions in the high-q region, from approximately 60 to 95 seconds, controllability exists for all 95 percentile September/October winds. Staging criteria are met for all engine-out cases that maintain control to cutoff.

4.1.2 (Continued)

Structural capability is not exceeded at the Command Service Module joint following TOPS shutdown. Figure 4-5 shows a typical bending moment history following TOPS shutdown.

Figure 4-6 shows inertial flight path angle versus inertial velocity envelope for S-IC single engine failures. In deriving envelopes, engine failures are simulated with a 3-sigma performance dispersion for the failed stage. The 100 second free-flight-to-reentry limit is violated for some S-IC engine shutdown times. However, in most cases this violation occurs either with an altitude greater than nominal, or prior to launch escape tower jettison. Figure 4-7 shows the altitude versus surface range envelope for S-IC single engine failures.

S-II single engine-out failures do not require abort. S-II envelopes of flight path angle versus inertial velocity, and altitude versus surface range for this malfunction are shown in Figures 4-8 and 4-9 , respectively.

S-IVB engine failure requires staging to the Command Service Module for abort.

4.1.3 EDS Effectiveness and Crew Safety

Since there are no effective abort cues for early failures resulting in pad fallback, holddown post collision or tower collision, it is assumed that these failures result in crew loss.

The 10 degree per second manual cue provides positive abort lead times for late high-q aborts resulting from early failures. These abort lead times are shown in Figure 4-10.

False automatic aborts on the 4 degree per second rate limit can occur for engines 2 or 3 out in the 75 to 80 second region.

4.1.4 Mission Completion Capability

Figure 1-8 summarizes orbital capability for S-IC and S-II single engine-out failures. S-IVB first burn duration as a function of S-IC and S-II malfunction time is shown in Figures 4-11 and 4-12, respectively. For S-IVB failure, Command Service Module parking orbit insertion is possible provided the failure occurs after 605 seconds flight time, and assuming Service Propulsion System ΔV capability to be 500 meters per second.

SHUTDOWN TIME - t_f (SEC)	FREEZE DURATION (SEC)	TIME OF INITIATION (SEC)
$0 \leq t_f \leq 14$	60	38
$14 \leq t_f \leq 38$	$67.2 - 1.2\ t_f$	38
$38 \leq t_f \leq 66$	$50.9 - 0.77\ t_f$	t_f
$66 \leq t_f$	0	NONE

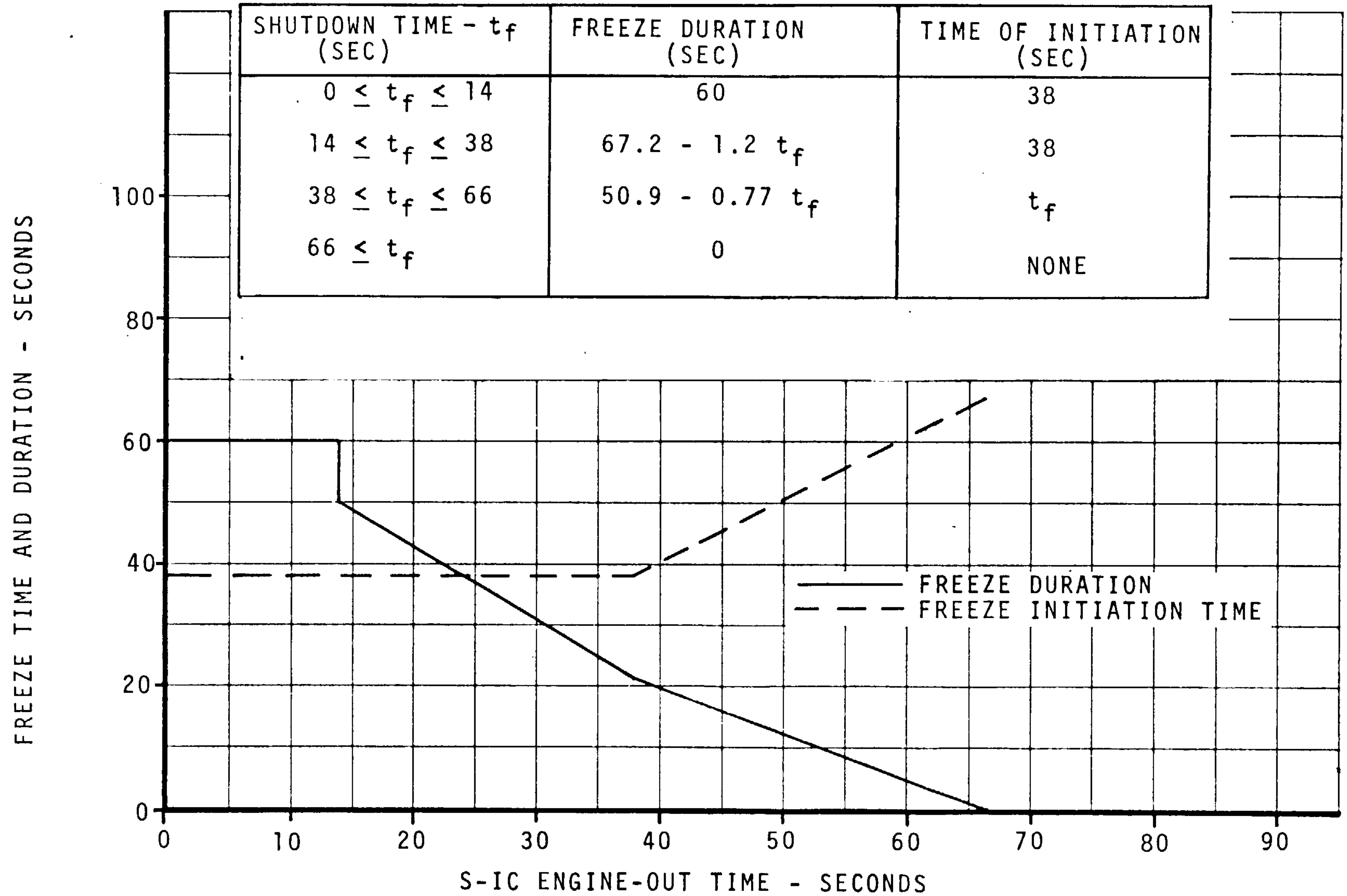


FIGURE 4-2 AS-507 S-IC ENGINE-OUT GUIDANCE x-FREEZE SCHEDULE

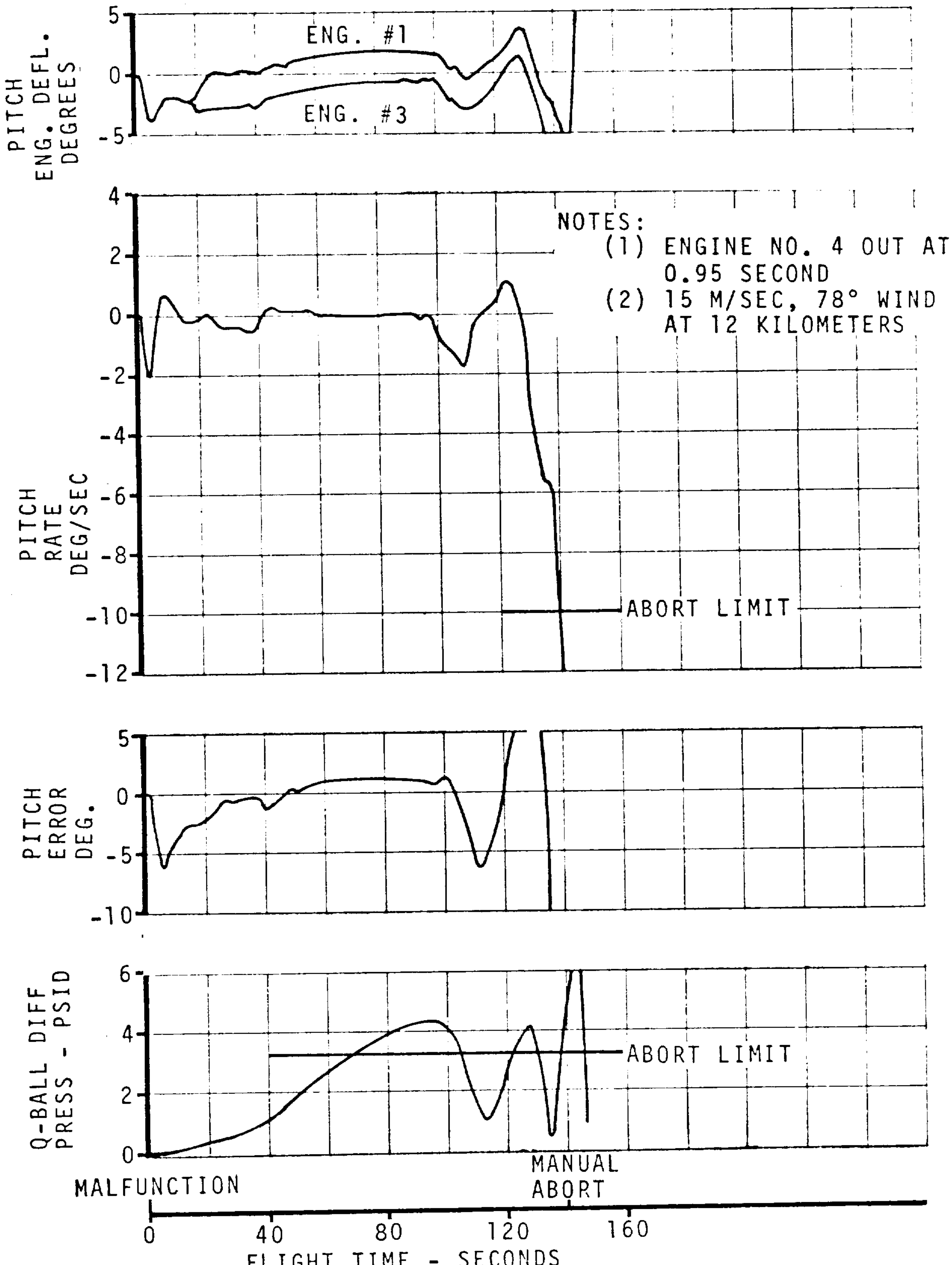
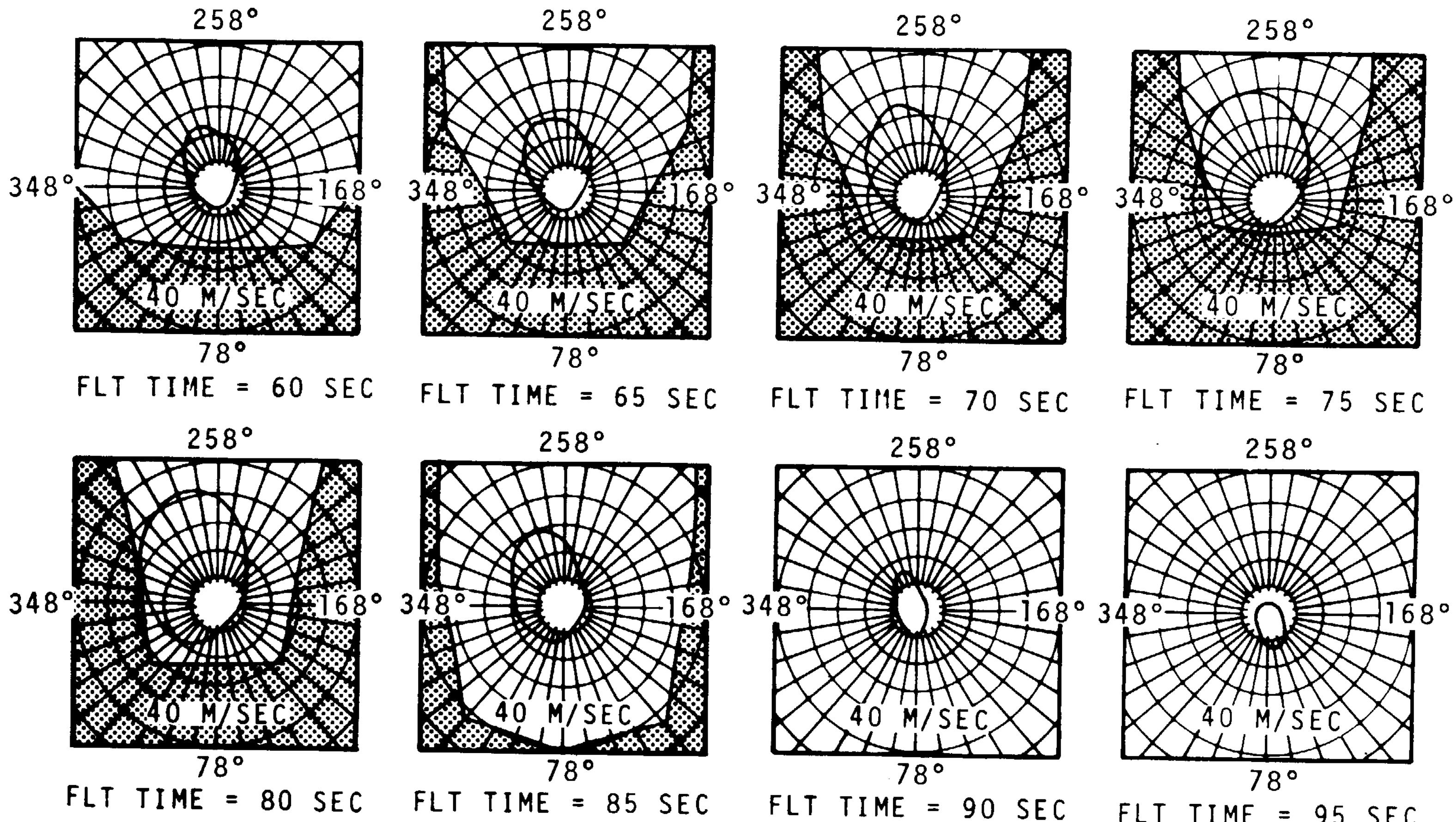


FIGURE 4-3 TIME HISTORY OF DYNAMIC VARIABLES FOR AN EARLY ENGINE-OUT MALFUNCTION



LEGEND: WINDS THAT CAUSE LOSS OF CONTROL

NOTES:
- 78° LAUNCH AZIMUTH
- 99 PERCENTILE SHEAR AND GUST REDUCED BY 15%
- ROSE REPRESENTS SEPTEMBER-OCTOBER 95 PERCENTILE WINDS
- CONCENTRIC CIRCLES REPRESENT WIND VELOCITY
- RADIAL LINES REPRESENT WIND DIRECTION

FIGURE 4-4 S-IC ENGINE-OUT CONTROL CAPABILITY

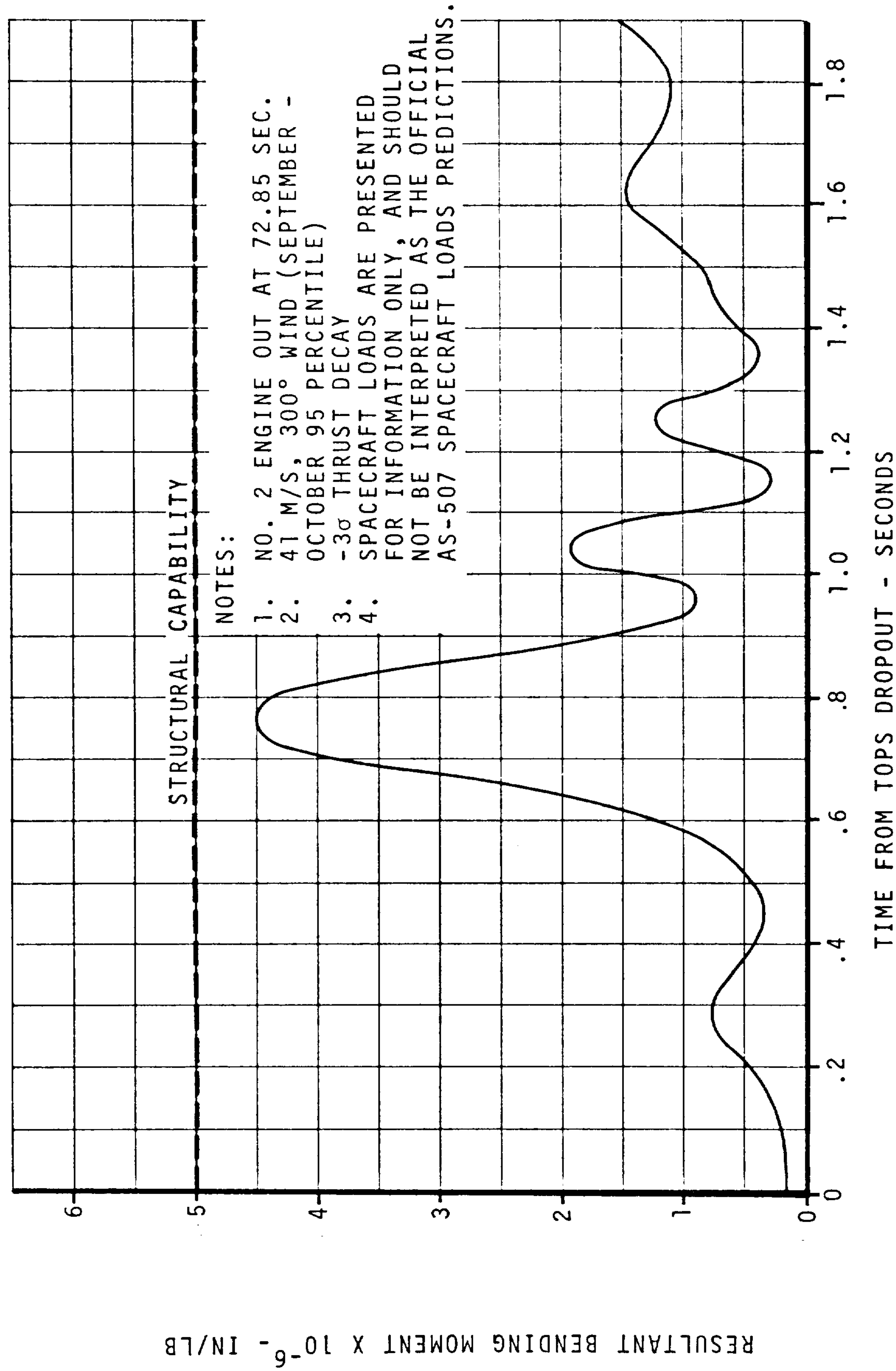


FIGURE 4-5 CSM JOINT BENDING MOMENT TIME HISTORY FOLLOWING ENGINE SHUTDOWN

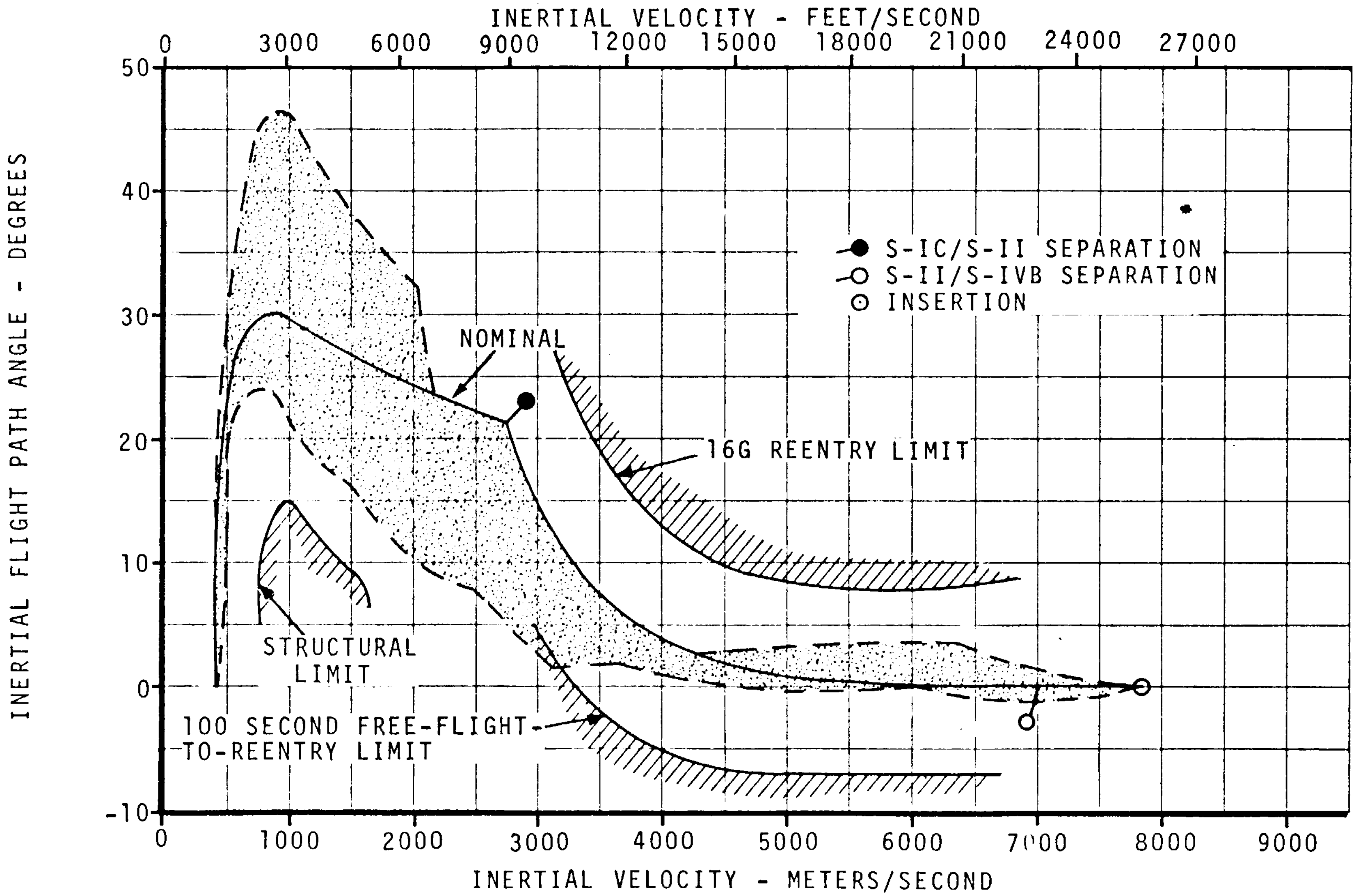


FIGURE 4-6 INERTIAL FLIGHT PATH ANGLE VS INERTIAL VELOCITY ENVELOPE FOR S-IC SINGLE ENGINE FAILURES

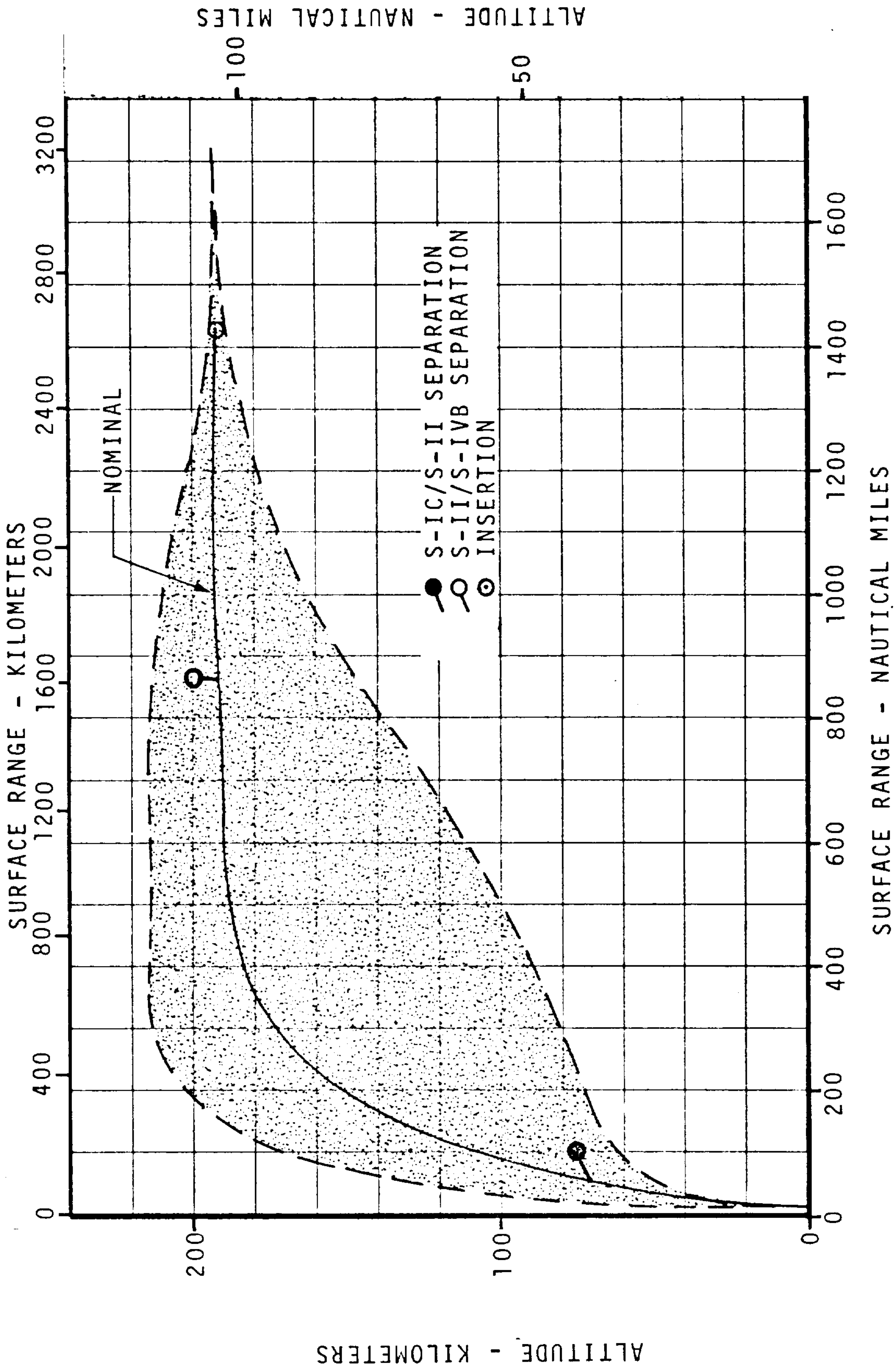


FIGURE 4-7 ALTITUDE VS SURFACE RANGE FOR S-IC SINGLE ENGINE FAILURES

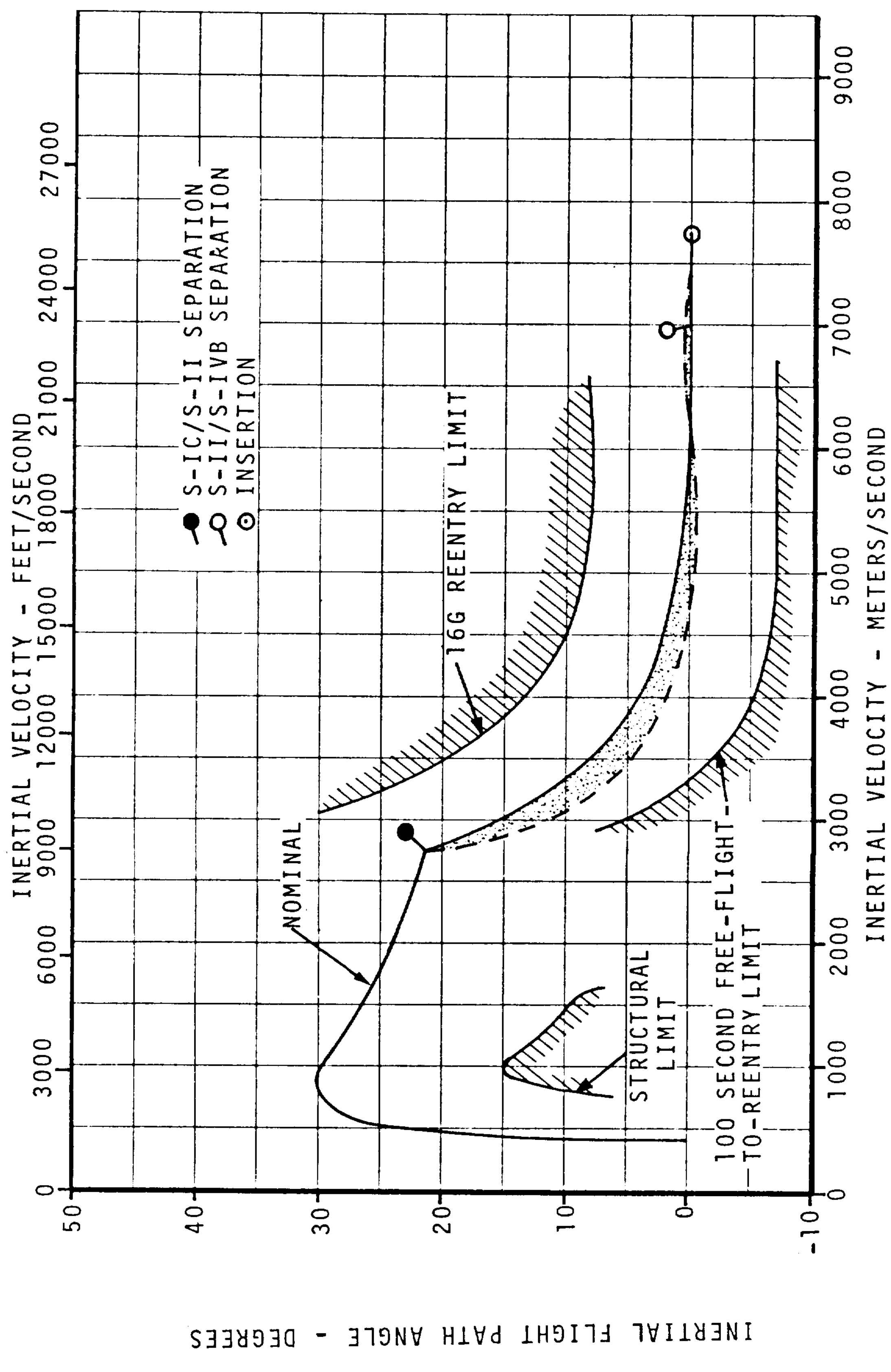


FIGURE 4-8 INERTIAL FLIGHT PATH ANGLE VS INERTIAL VELOCITY ENVELOPE FOR S-II SINGLE ENGINE FAILURES

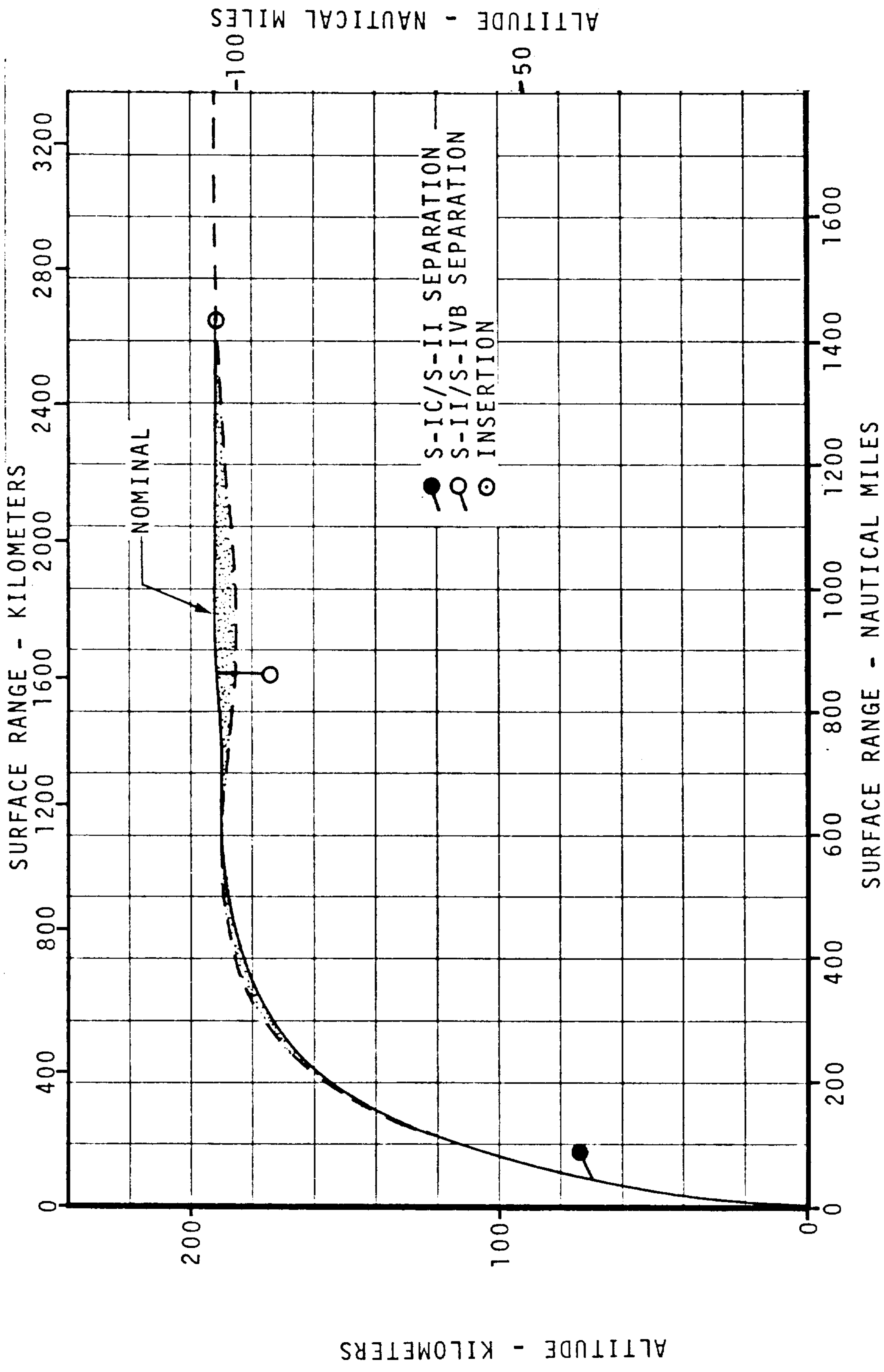


FIGURE 4-9 ALTITUDE VS SURFACE RANGE FOR S-II SINGLE ENGINE FAILURES

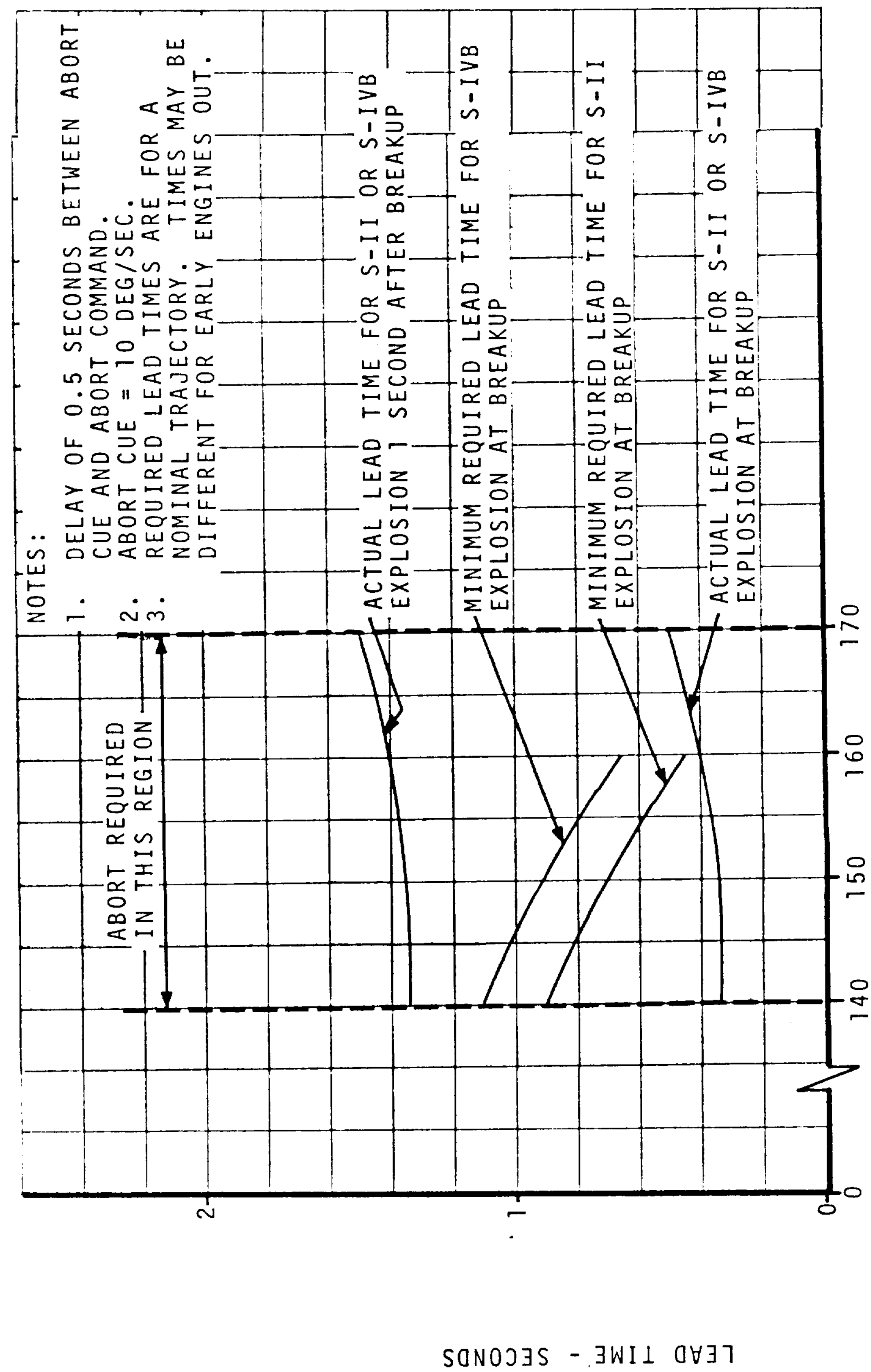


FIGURE 4-10 LEAD TIMES FOR ABORT AFTER EARLY S-IC ENGINE FAILURES

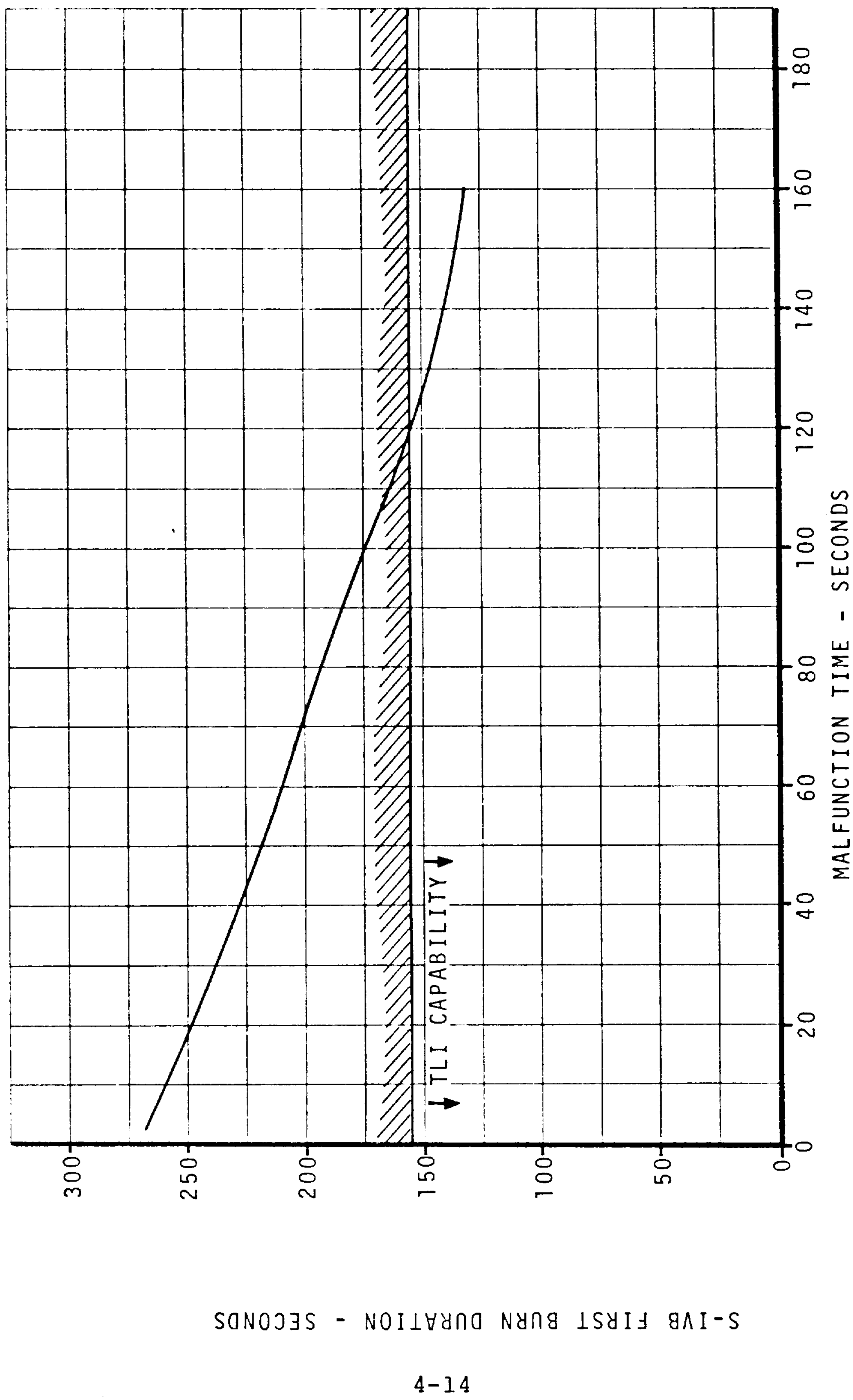


FIGURE 4-11 S-IVB FIRST BURN DURATION FOR S-IC SINGLE LOWER ENGINE SHUTDOWNS ACHIEVING POI

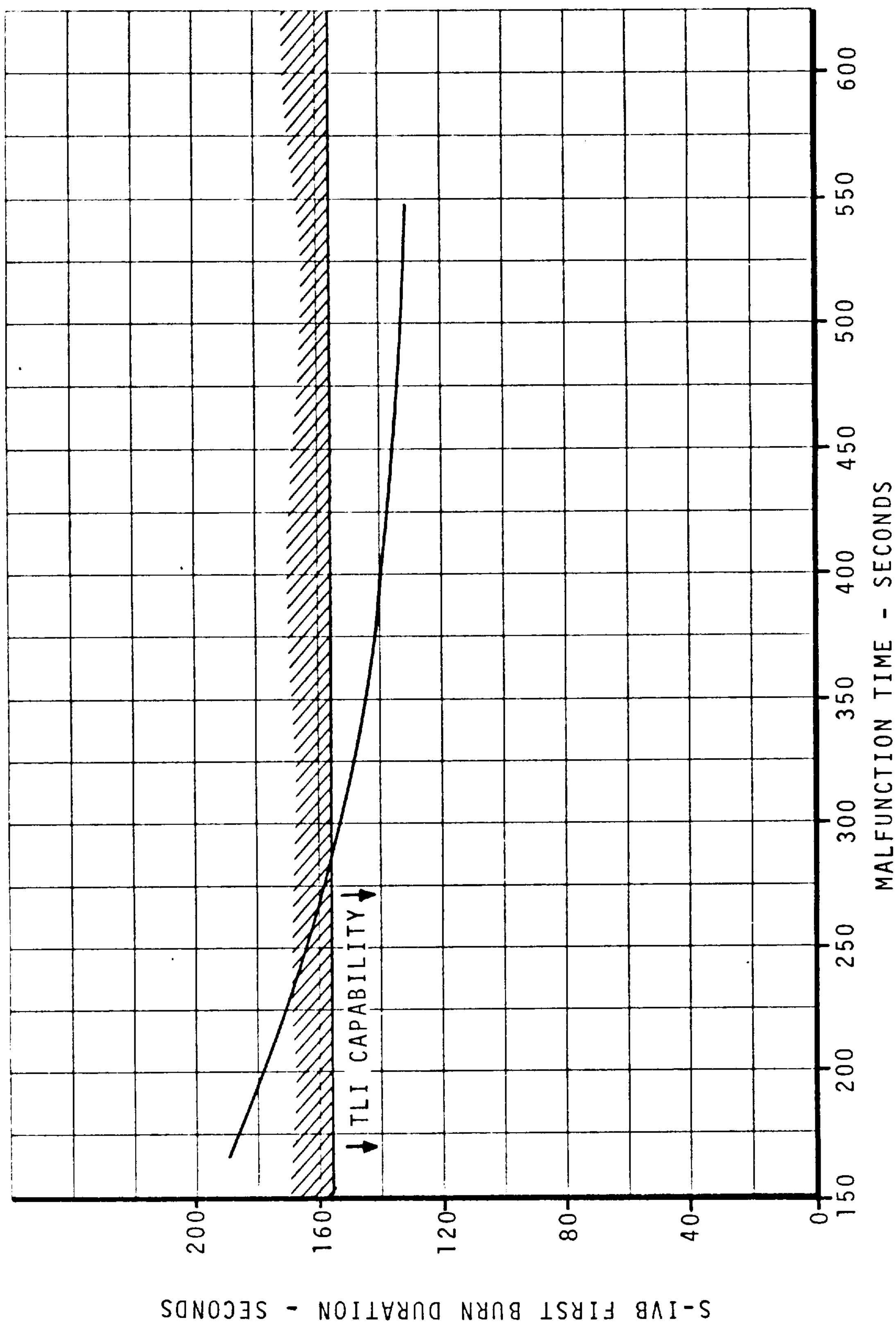


FIGURE 4-12 S-IVB FIRST BURN DURATION FOR S-II SINGLE LOWER ENGINE SHUTDOWNS ACHIEVING POI

4.2 DUAL ENGINE LOSS OF THRUST

4.2.1 Malfunction Description

Loss of thrust engine failures are described in Section 4.1. When loss of thrust on two adjacent control engines is detected during S-IC flight, after Automatic Abort Cutoff (AACO) is enabled, the remaining engines are automatically shut down.

4.2.2 Malfunction Dynamics

Ability to maintain control after failure of two engines in S-IC or S-II flight depends on which engines have failed and the times at which they fail. Figures 4-13 through 4-26 show the failure times which result in loss of control for each pair of engines. Figure 1-8 summarizes these times for simultaneous engine failures.

The regions shown in Figures 4-13 through 4-16 for adjacent control engines include the effects of early S-IC shutdown. Significant events during the time from 134.8 seconds to S-II ignition are:

EVENT	FLIGHT TIME (SECONDS)
AACO Enable	134.8
CECO & Timebase 2 Set	135.0
Timebase 3 Enable	152.0
OECO & Timebase 3 Set (Nominal)	159.9
S-II at 90% Thrust (Nominal)	164.3

Note that timebase 3 cannot be set until it is enabled 17.0 seconds after CECO. If shutdown occurs at 134.8, there are 21.6 seconds of coast between S-IC shutdown and the time S-II reaches 90% of full thrust. This extended coast period, coupled with a significant attitude rate, causes loss of control. Figure 4-27 shows typical dynamics for two adjacent control engines out after CECO.

No loss of control occurs for a single S-IC engine out followed by a single S-II engine out.

4.2.3 EDS Effectiveness and Crew Safety

During S-IC flight automatic abort is initiated if the thrust OK pressure switches on two engines drop out. This dual engine out automatic abort may be inhibited manually by the crew, but is inhibited automatically just prior to CECO.

4.2.3 (Continued)

The recommended time to inhibit automatic abort is 120 seconds in nominal flight because failure of the CSM joint can cause crew loss for simultaneous failures prior to this time.

After 120 seconds, manual abort cues are:

a. Engine-out lights
b. Ten degrees/second overrate light and spacecraft rate indicator
c. Abort request light

An additional abort cue of ±20 degrees attitude error is recommended for two control engines out after CECO. With this cue, no crew loss occurs for this malfunction.

4.2.4 Mission Completion Capability

Mission completion capability for two engines out is shown in Figure 1-8 and in Figures 4-13 through 4-27.

For adjacent S-IC control engines out after AACO enable, the vehicle fails to reach a 75 nautical mile orbit if shutdown occurs prior to 149 seconds. Loss of thrust acceleration causes failure of the accelerometer reasonableness test. The boost navigator uses backup F/M tables to calculate velocity resulting in a position error. Three failures of the accelerometer reasonableness test cause the orbit to have a perigee less than 75 nautical miles. Therefore, if the second engine fails prior to 149 seconds, the orbit is unsafe. A possible method of eliminating the long coast is to begin timebase 3 at shutdown. This eliminates most loss of control cases and permits a safe orbit for all cases that do not lose control.

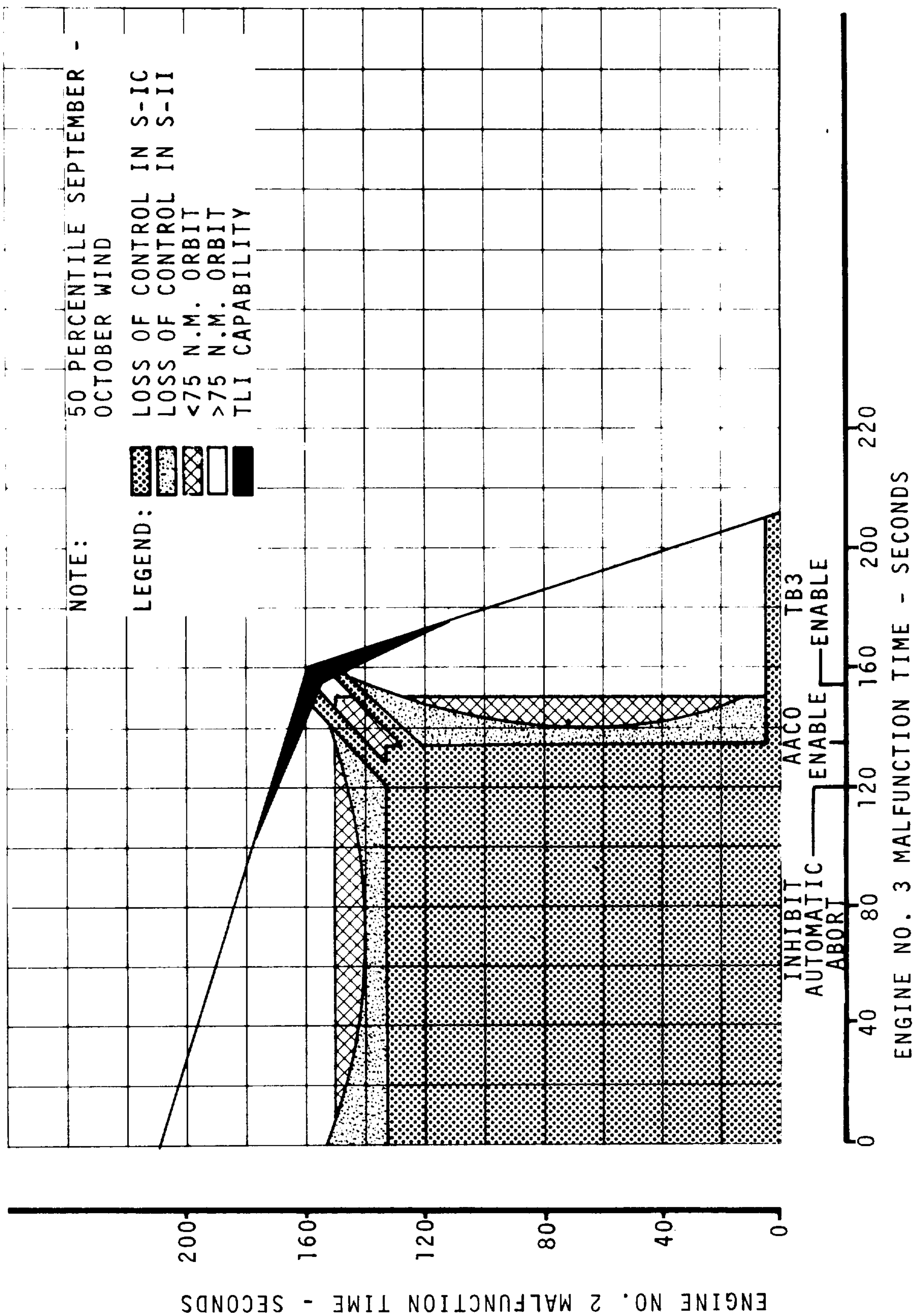


FIGURE 4-13 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 2 AND NO. 3

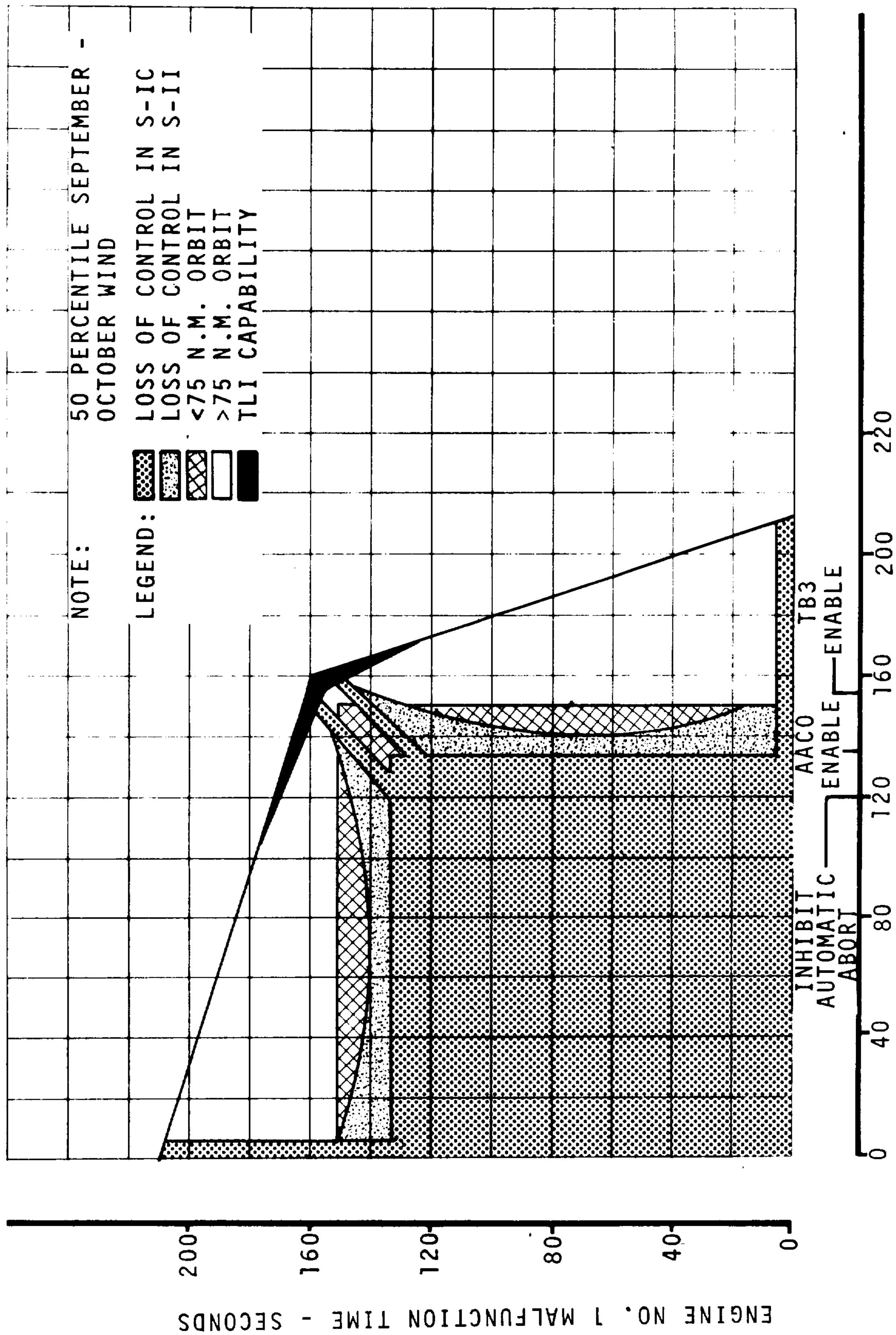


FIGURE 4-14 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 1 AND NO. 2

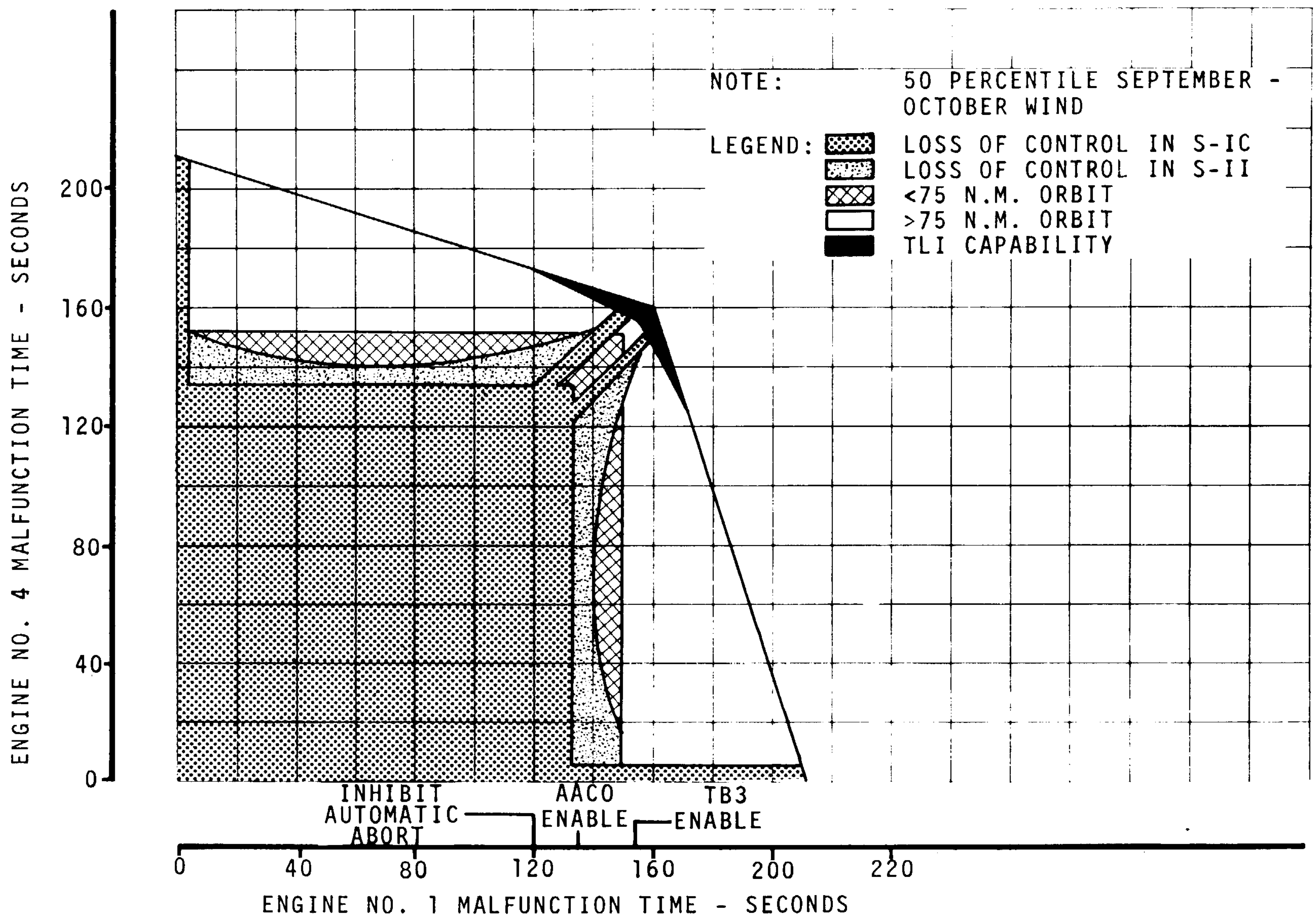


FIGURE 4-15 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 1 AND NO. 4

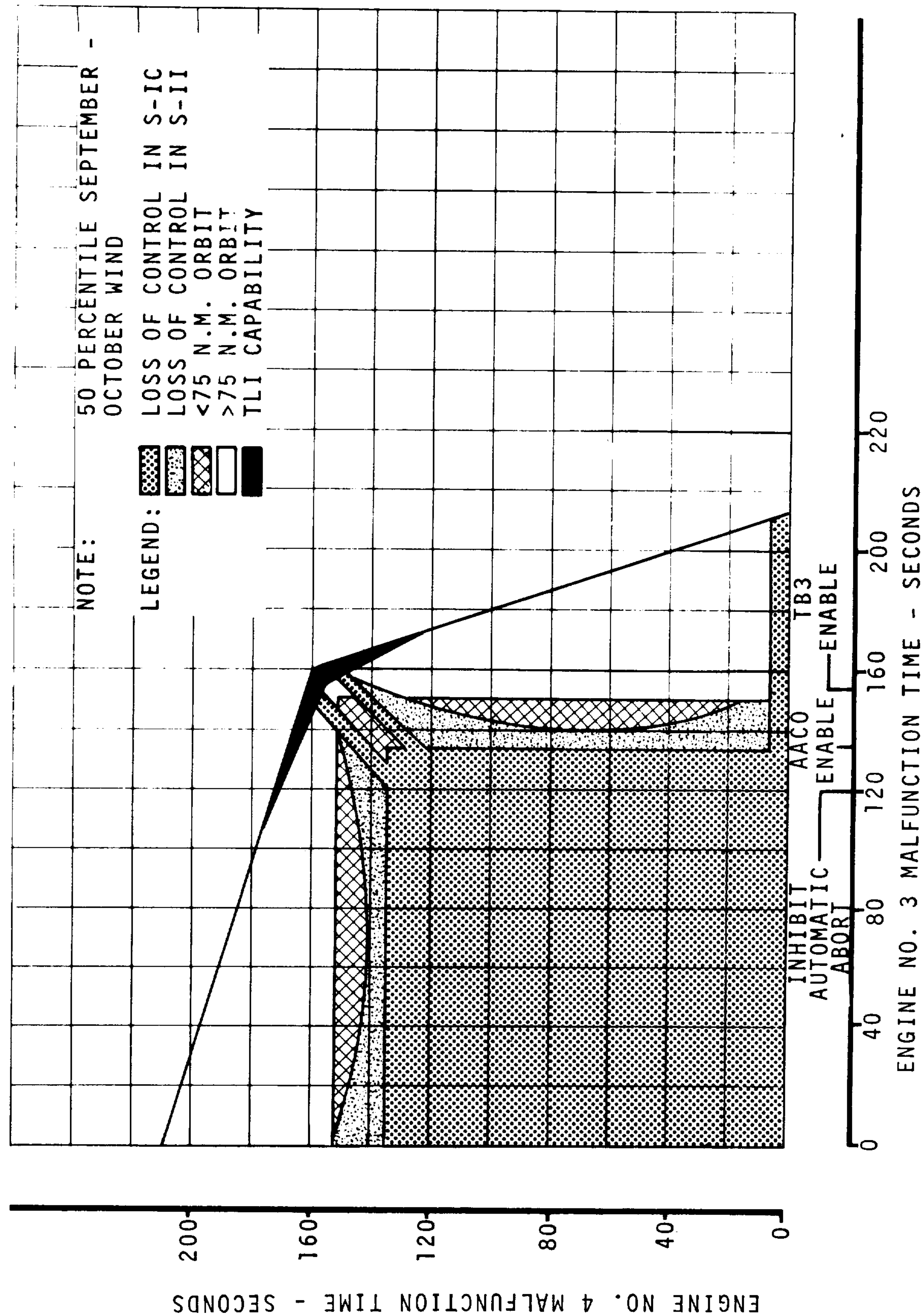


FIGURE 4-16 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 3 AND NO. 4

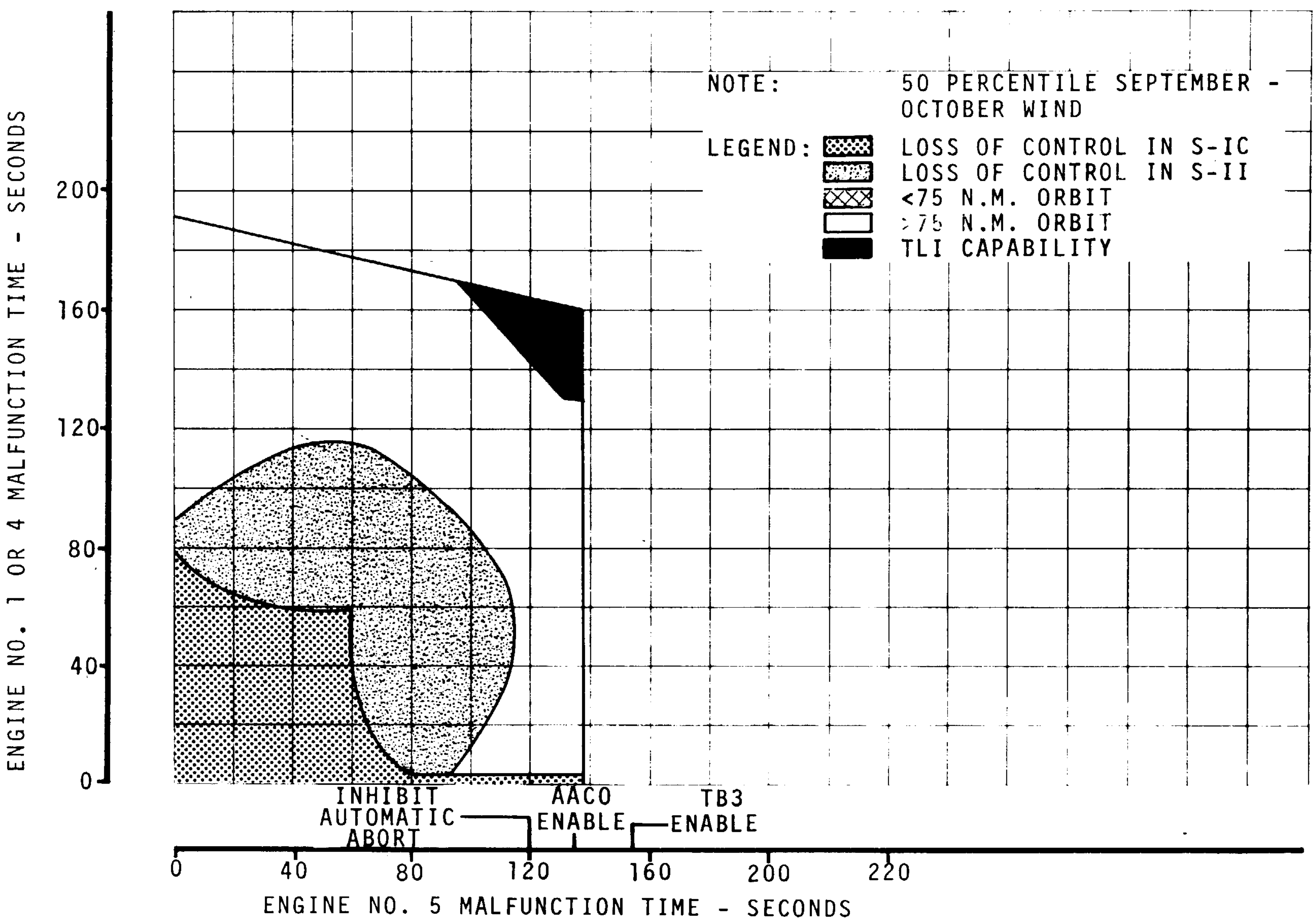


FIGURE 4-17 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 1 AND NO. 5 OR ENGINES NO. 4 AND 5

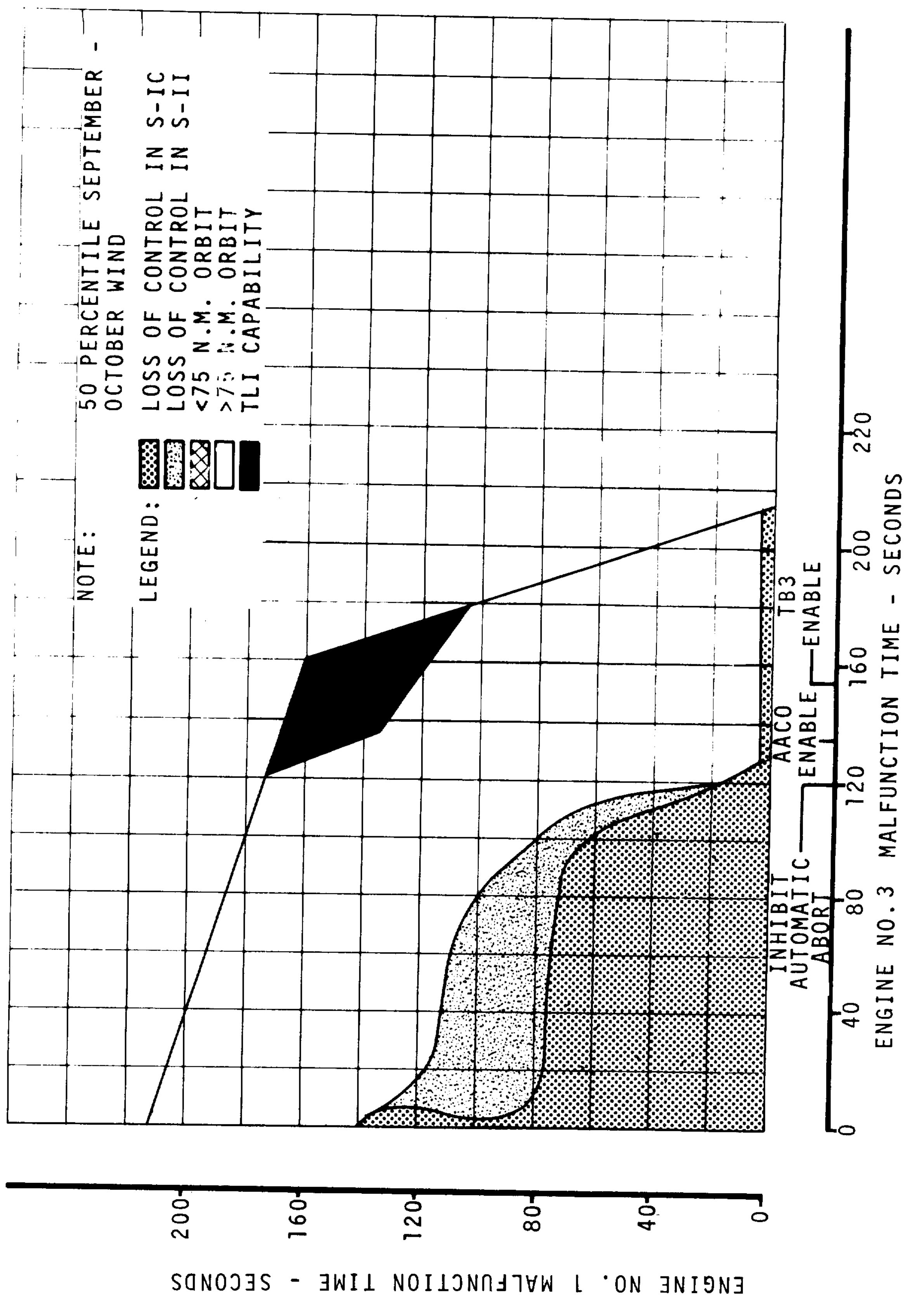


FIGURE 4-18 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 1 AND NO. 3

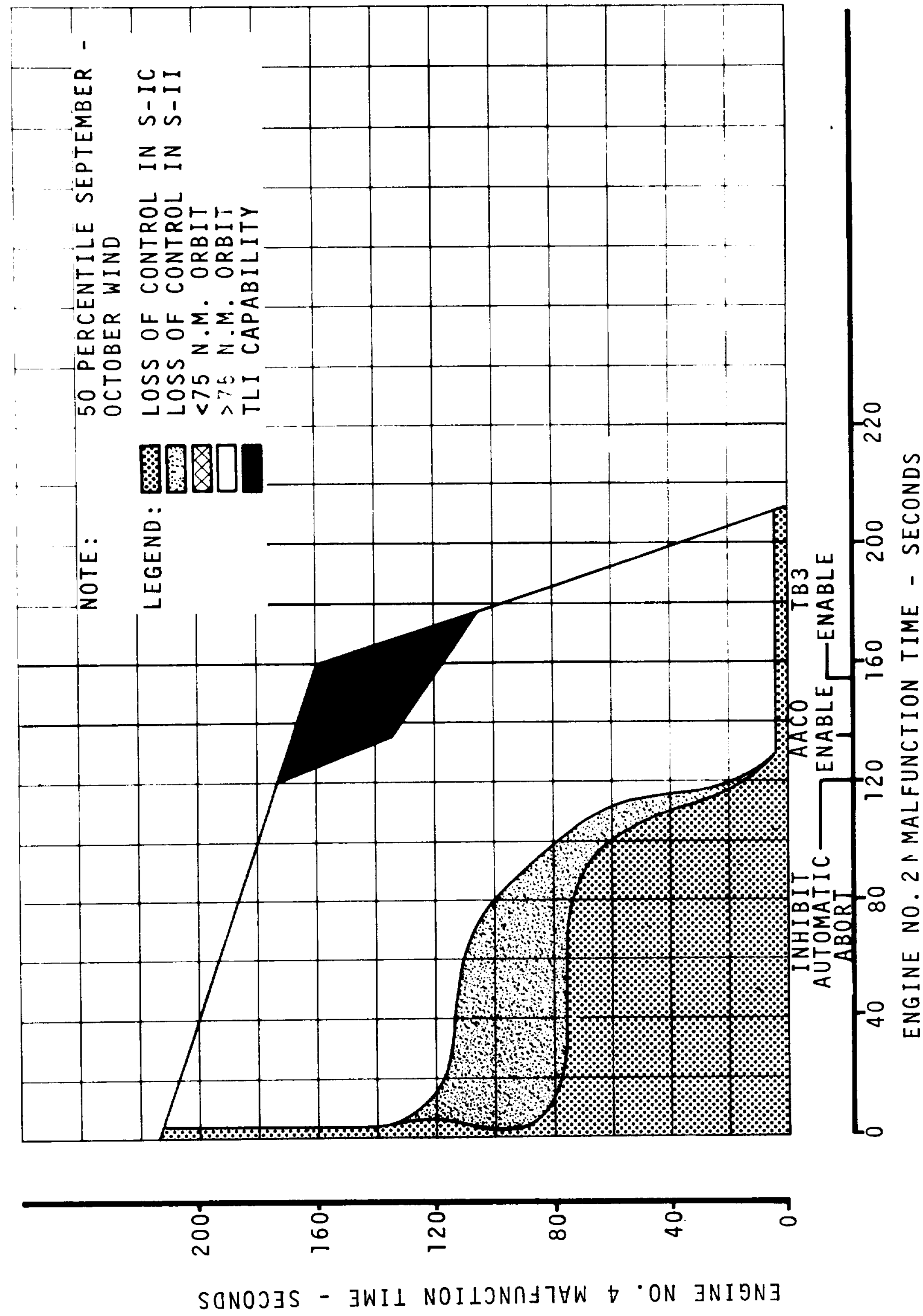


FIGURE 4-19 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 2 AND NO. 4

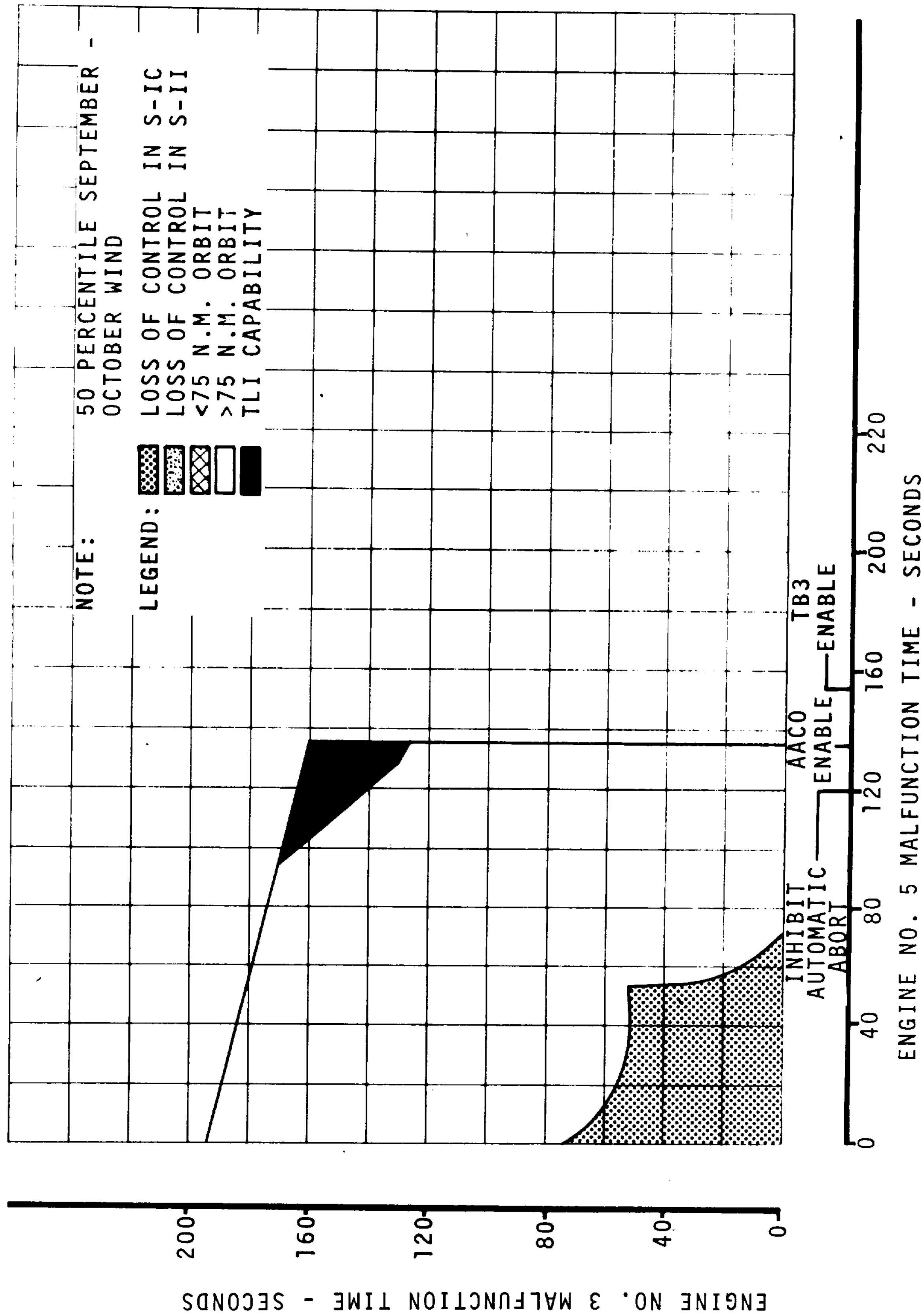


FIGURE 4-20 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 3 AND NO. 5

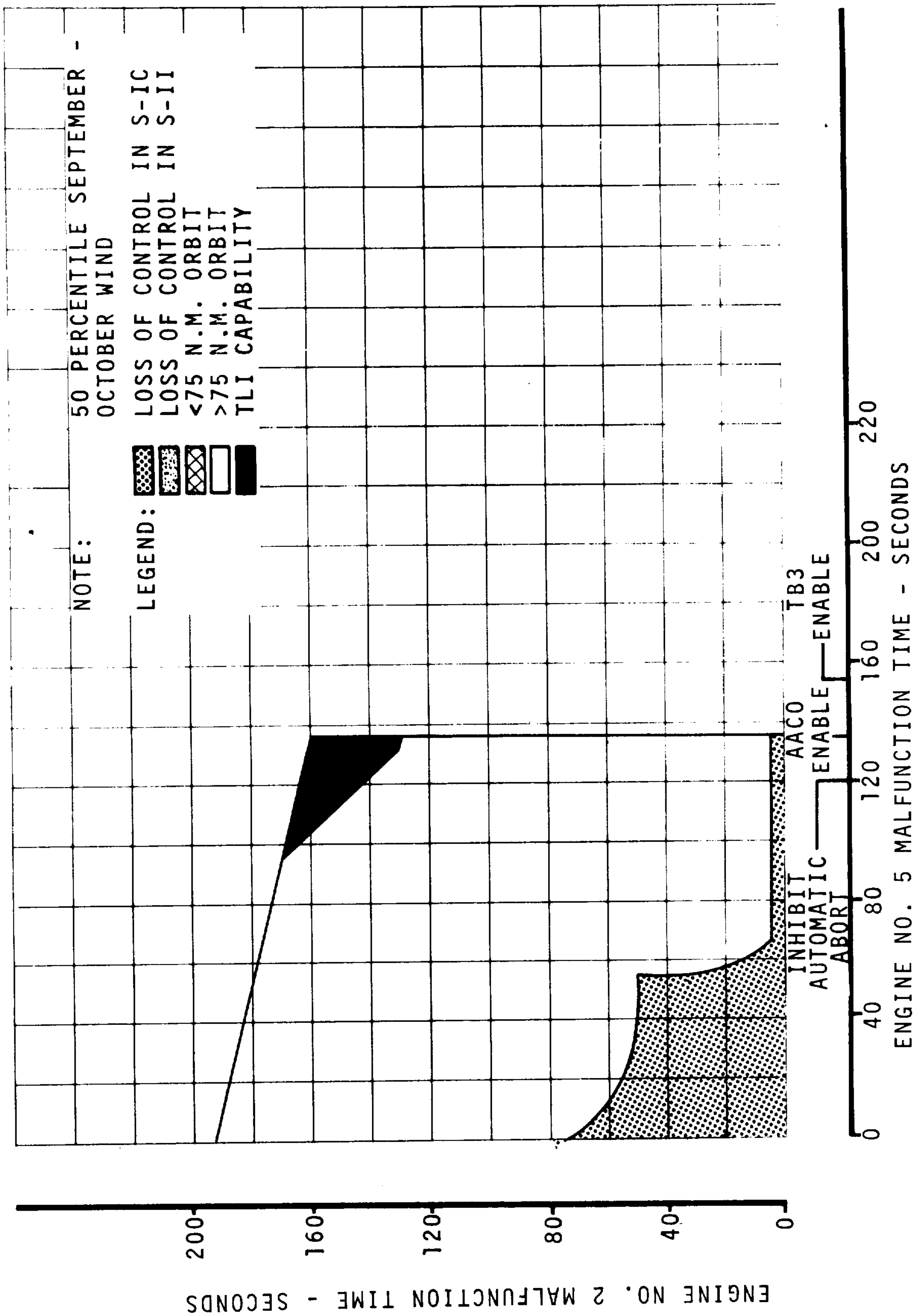


FIGURE 4-21 S-IC SEQUENTIAL ENGINE-OUT CAPABILITY - ENGINES NO. 2 AND NO. 5

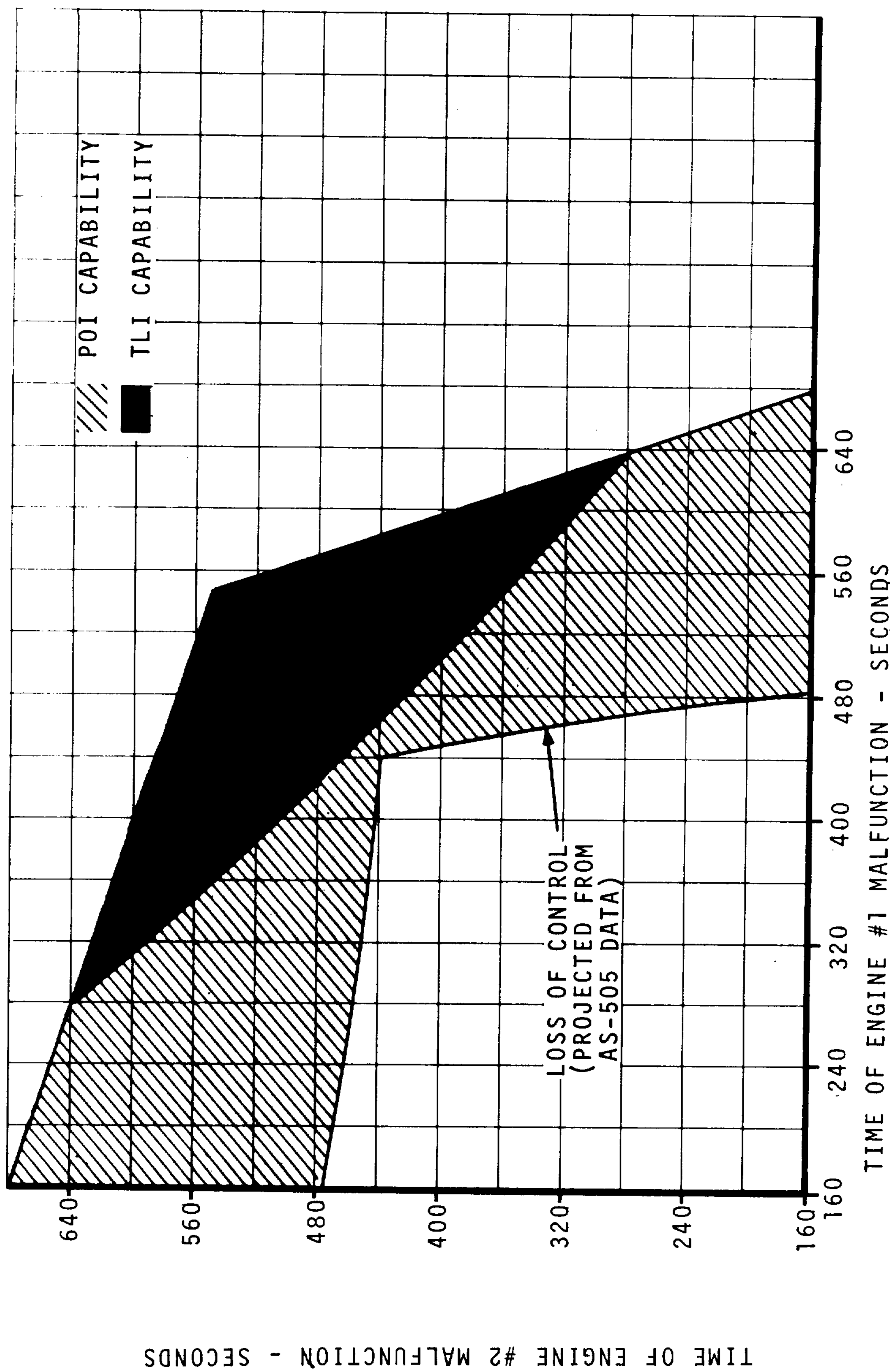


FIGURE 4-22 MISSION COMPLETION CAPABILITY FOR S-II NO. 1 & 2 ENGINE SHUTDOWN

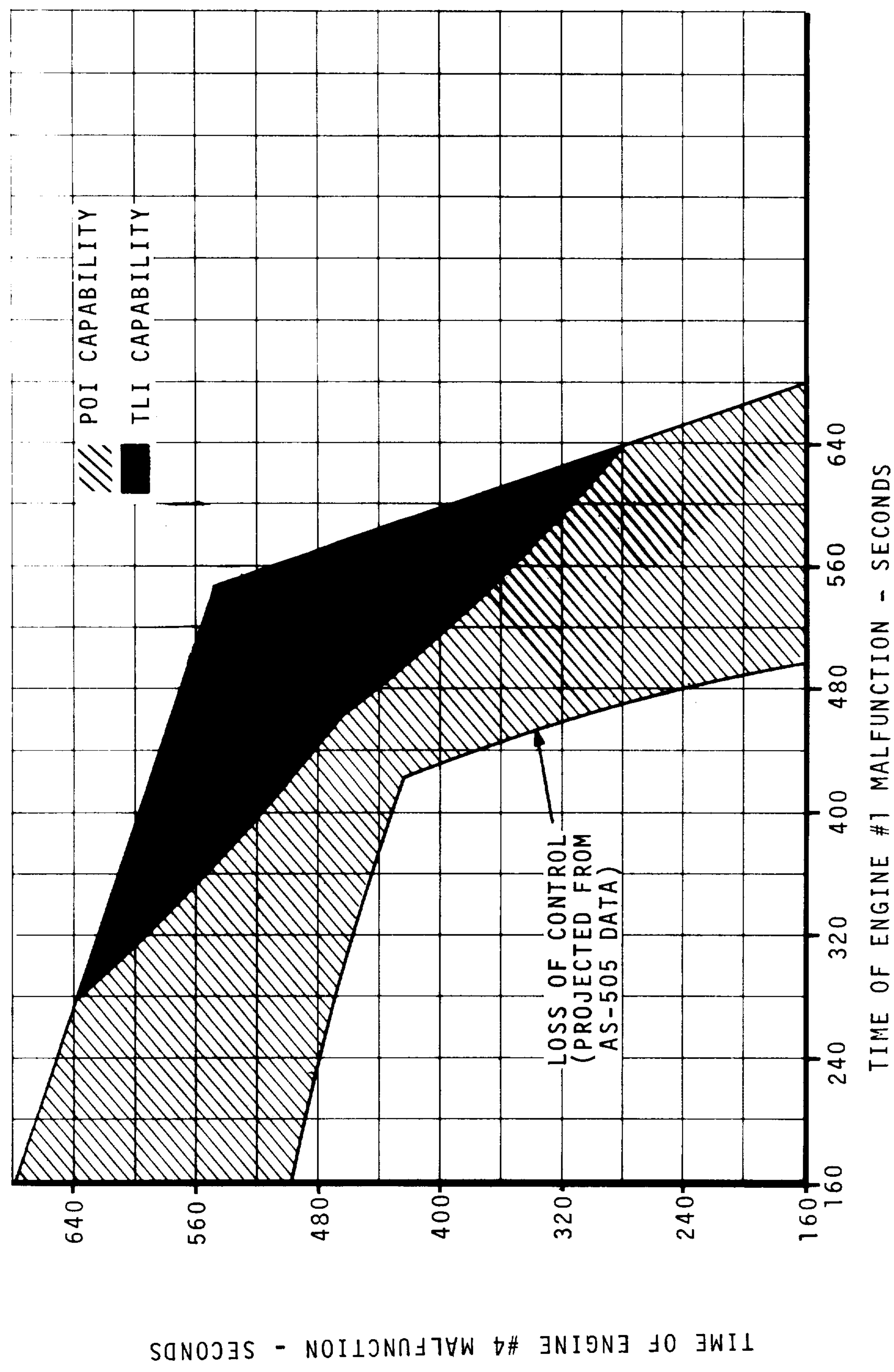


FIGURE 4-23 MISSION COMPLETION CAPABILITY FOR S-II NO. 1 & 4 ENGINE SHUTDOWN

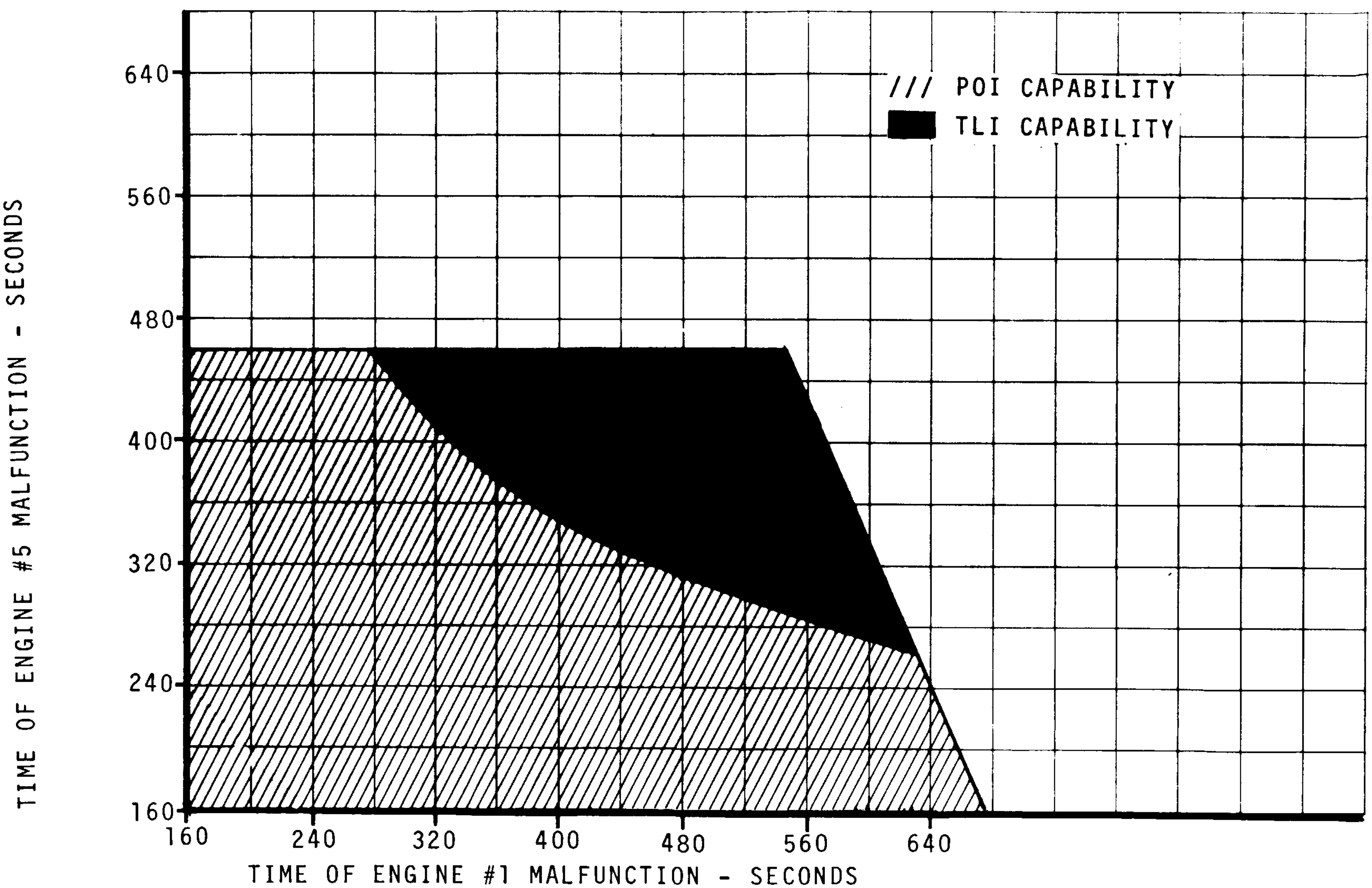


FIGURE 4-24 MISSION COMPLETION CAPABILITY FOR S-II NO. 1 & 5 ENGINE SHUTDOWN

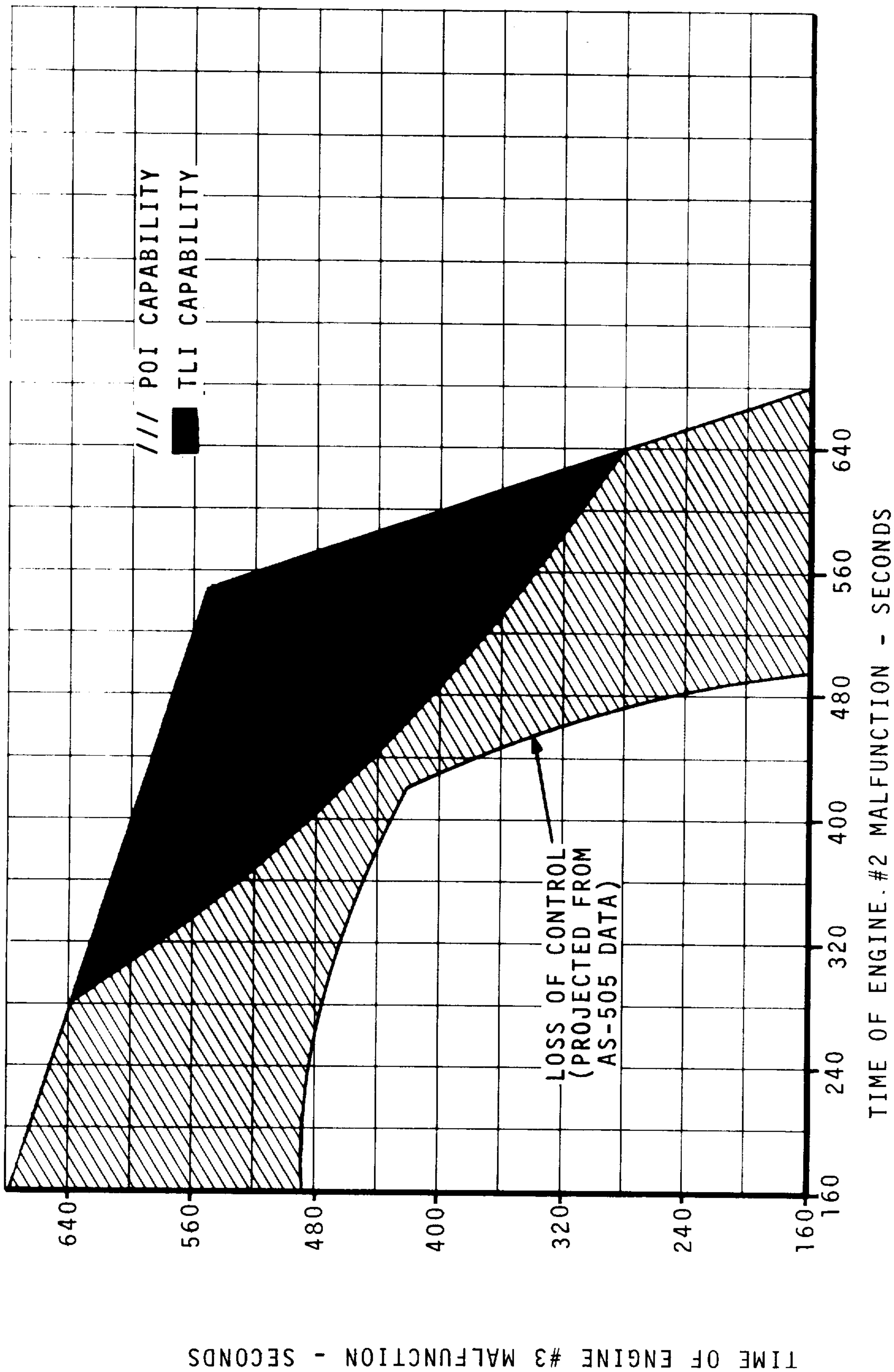


FIGURE 4-25 MISSION COMPLETION CAPABILITY FOR S-II NO. 2 & 3 ENGINE SHUTDOWN

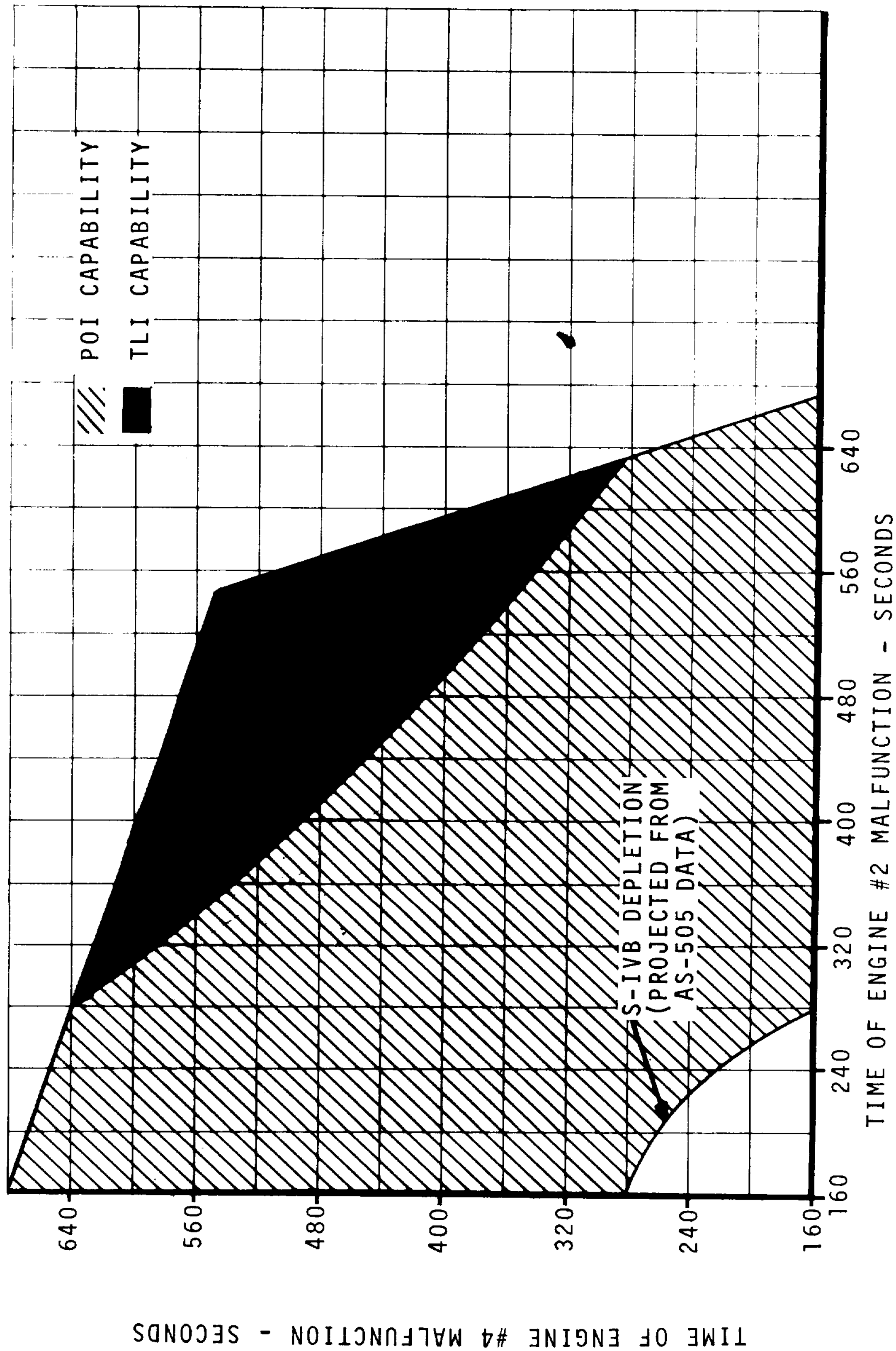


FIGURE 4-26 MISSION COMPLETION CAPABILITY FOR S-II NO. 2 & 4

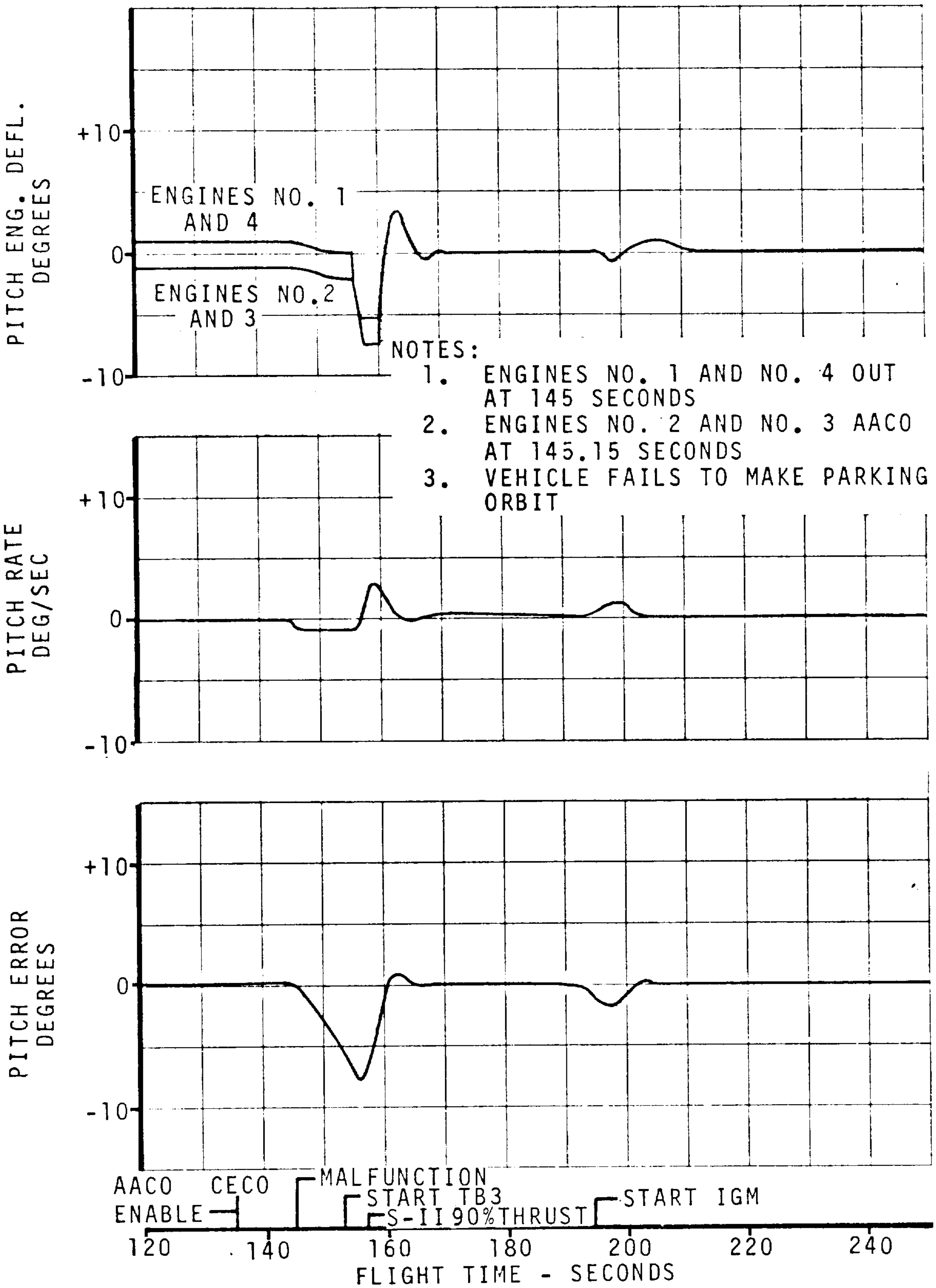


FIGURE 4-27 TIME HISTORY OF DYNAMIC VARIABLES FOR ENGINES NO. 1 AND NO. 4 OUT AFTER CECO

4.3 ACTUATOR HARDOVER

4.3.1 Malfunction Description

Actuator hardover is any failure that causes an engine actuator to either extend or retract to its limit.

4.3.2 Malfunction Dynamics

Analysis shows that during S-IC flight, an actuator fully extended (engine deflected inboard) gives worst case results due to the S-IC outboard engine cant of two degrees.

An actuator hardover near the pad does not cause pad fallback, but any actuator hardover before 0.7 seconds results in an engine bell colliding with the holddown posts. An actuator hardover in the positive yaw direction before 3.2 seconds results in tower collision.

Figure 4-28 shows control capability for an actuator hardover in the high-q region. An actuator hardover causes loss of control in the 95 percentile wind rose between 68 and 88 seconds (high-q region). Figure 4-29 shows typical dynamics for an actuator hardover at 80 seconds. Loss of control results from excessive aerodynamic moments exceeding the control capability.

Vehicle tension loads due to the abrupt loss of thrust at abort cutoff cause structural failure for actuator hardover malfunctions which require abort between 70 and 100 seconds flight time. Figure 4-30 shows tension loads for this malfunction occurring at 80 seconds.

An S-IC actuator hardover occurring between 100 and 107 seconds flight time can cause loss of control due to the excessive aerodynamic moment and the control gain switching transient. Figures 4-31 and 4-32 show typical dynamics and corresponding tension loads, respectively. No control loss occurs and no staging criteria are violated by an S-IC single actuator hardover between 108 and 160 seconds.

During S-II flight, there is no loss of control for an actuator hardover malfunction.

A single S-II actuator hardover inboard causes a large inboard deflection of another control engine, as shown in Figure 4-33 . This exposes the base of the S-II vehicle to excessive radiation from the engine plumes with the following possible results:

a. Collapse of thrust structure due to induced thermal stress

4.3.2 (Continued)

b. Loss of engine thrust and/or S-II/S-IVB separation capability due to wiring harness damage.

S-II stage damage may occur within 15 to 25 seconds after actuator hardover.

Any actuator hardover in the S-IVB stage results in tumbling.

4.3.3 EDS Effectiveness and Crew Safety

Since there are no abort cues for early failures resulting in holddown post or tower collision, it is assumed that these failures result in crew loss.

The manual abort cues and the automatic abort setting of 4 degrees per second provide positive abort lead times for all S-IC actuator hardover failures in the high-q region which require abort. Abort lead times are shown in Figure 4-34. The figure shows that abort can be unsafe if an explosion occurs at breakup, but is safe for all explosions occuring one second after breakup. This results in the probability of crew loss of 5 times per million flights. In most cases the manual abort cues, attitude error ($\pm$5.0 degrees) and q-ball ΔP (3.2 psid), occur before the automatic abort requirement is reached.

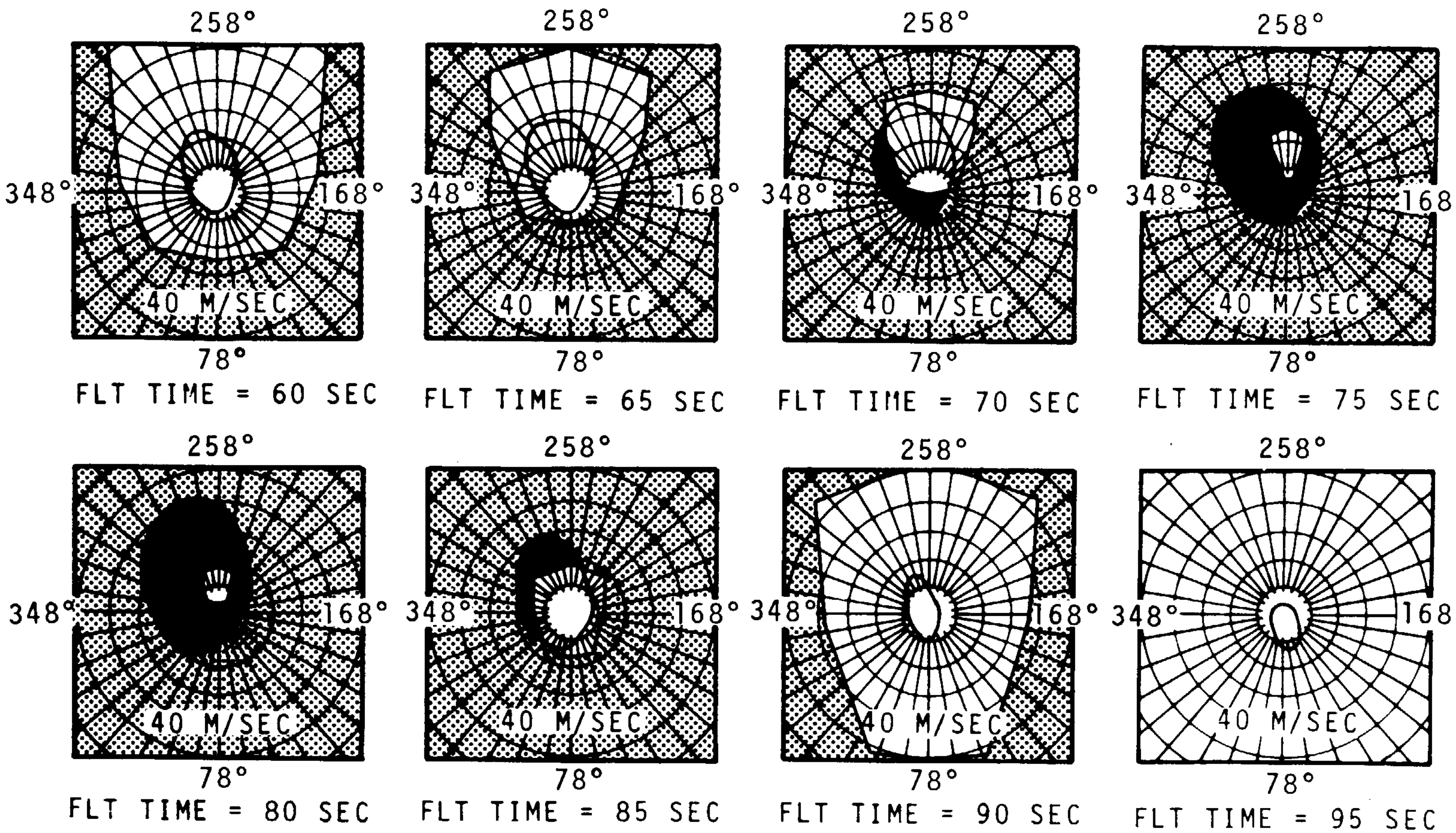
S-IC actuator hardover between 108 and 120 seconds can result in false automatic abort on overrate. Failures after 120 seconds do not require abort, and no false aborts are indicated.

While there is no loss of control for a single S-II engine actuator hardover, abort cues for this malfunction are necessary because of heating problems (Reference 6). These cues are voice request and the Abort Request Light.

For actuator failures during S-IVB flight, the 10 degree/second rate limit is the manual abort cue. All aborts are safe.

4.3.4 Mission Completion Capability

Any single actuator hardover in S-IC or S-II which does not require abort for reasons discussed in Section 4.3.3 has primary mission completion capability. Actuator hardover in S-IVB results in mission loss.



LEGEND:
- WINDS THAT CAUSE LOSS OF CONTROL
- WINDS LESS THAN SEPTEMBER-OCTOBER 95 PERCENTILE THAT CAUSE LOSS OF CONTROL

NOTES:
- 78° LAUNCH AZIMUTH
- 99 PERCENTILE SHEAR AND GUST REDUCED BY 15%
- ROSE REPRESENTS SEPTEMBER-OCTOBER 95 PERCENTILE WINDS
- CONCENTRIC CIRCLES REPRESENT WIND VELOCITY
- RADIAL LINES REPRESENT WIND DIRECTION

FIGURE 4-28 ACTUATOR HARDOVER CONTROL CAPABILITY

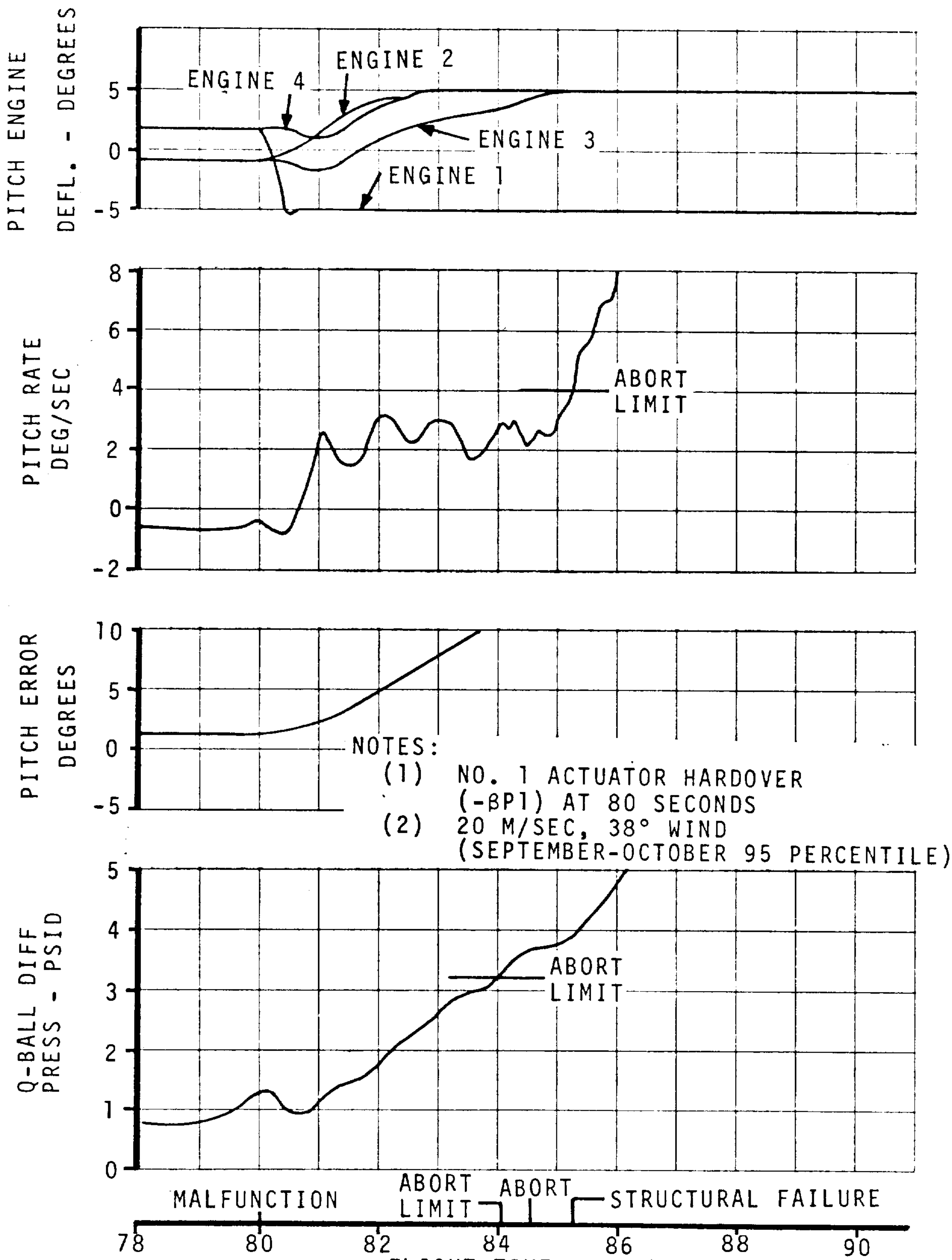


FIGURE 4-29 TIME HISTORY OF DYNAMIC VARIABLES FOR ACTUATOR HARDOVER AT 80 SECONDS

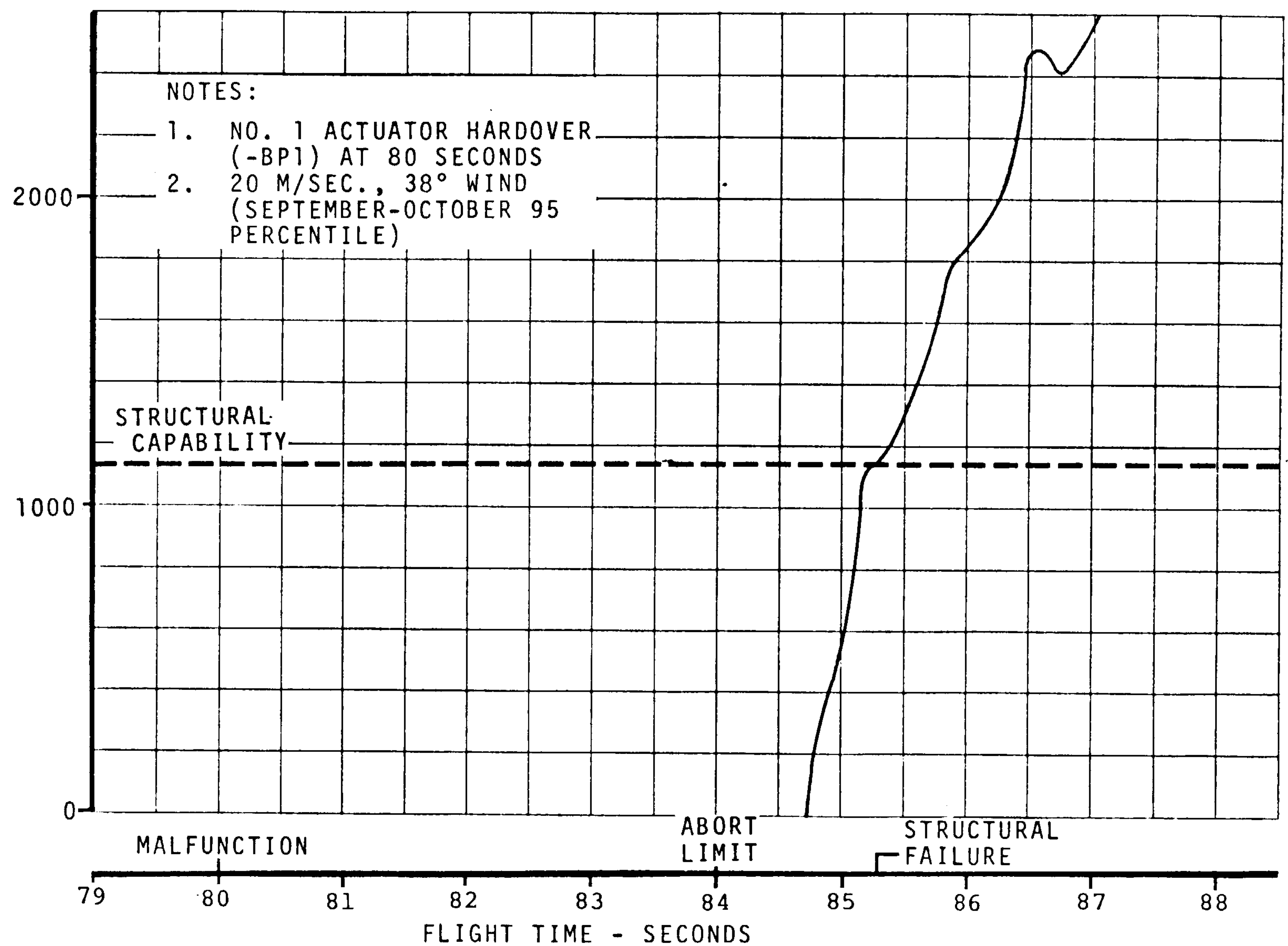


FIGURE 4-30 COMBINED TENSION LOADS FOLLOWING ACTUATOR HARDOVER AT 80 SECONDS (STA. 1564)

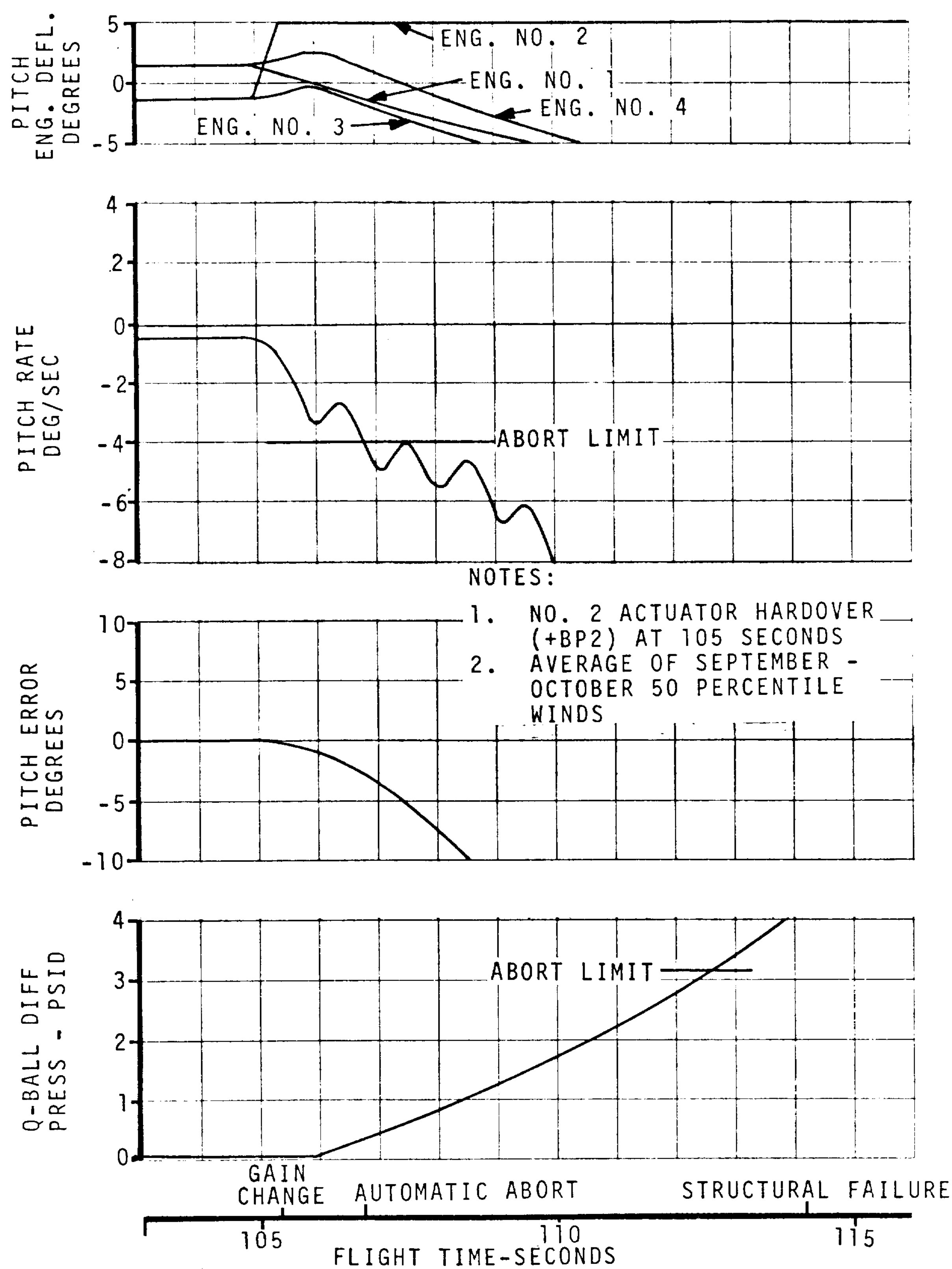


FIGURE 4-31 TIME HISTORY OF DYNAMIC VARIABLES FOR ACTUATOR HARDOVER NEAR GAIN CHANGE

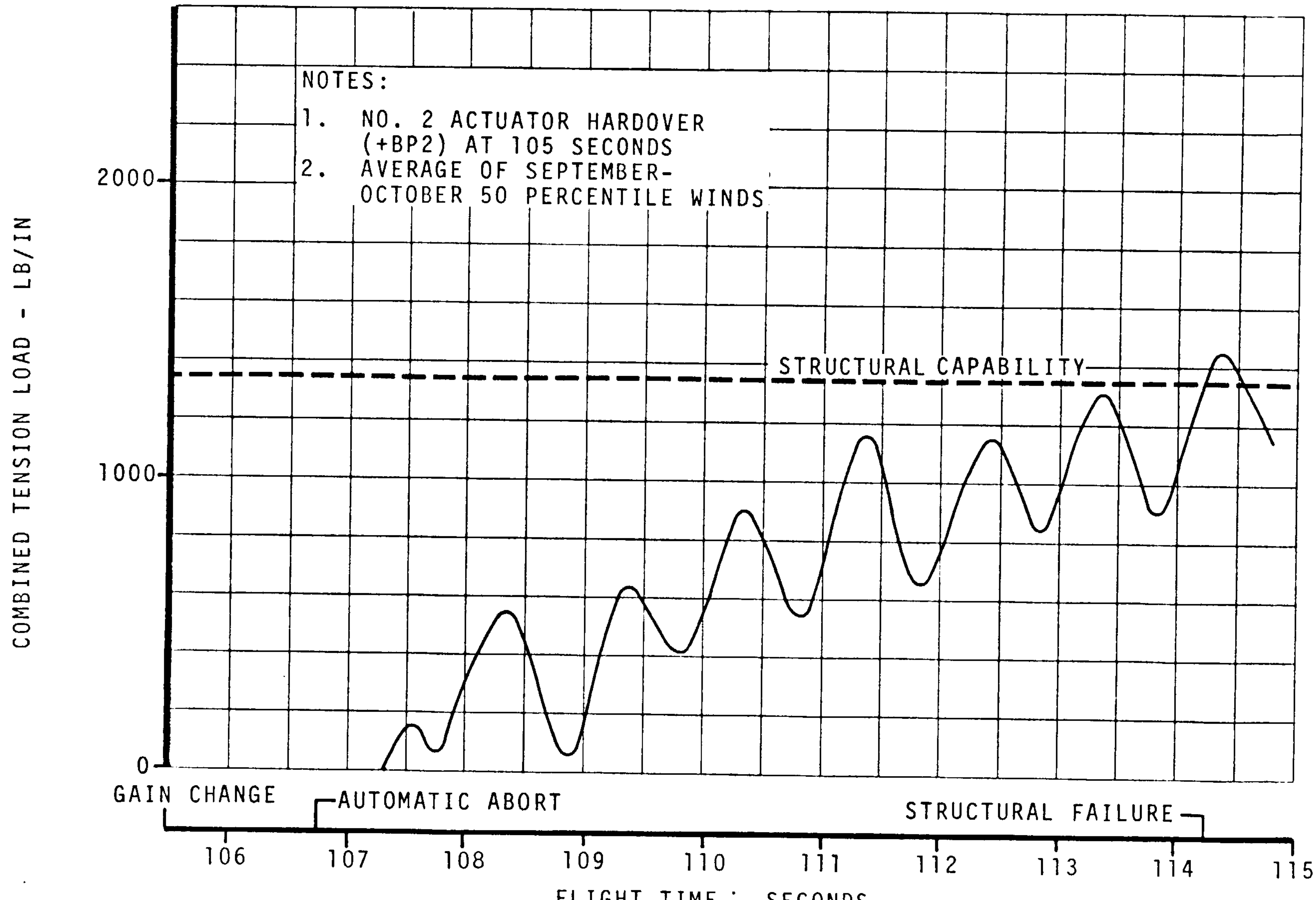


FIGURE 4-32 COMBINED TENSION LOADS FOLLOWING ACTUATOR HARDOVER NEAR GAIN CHANGE (STA. 1760)

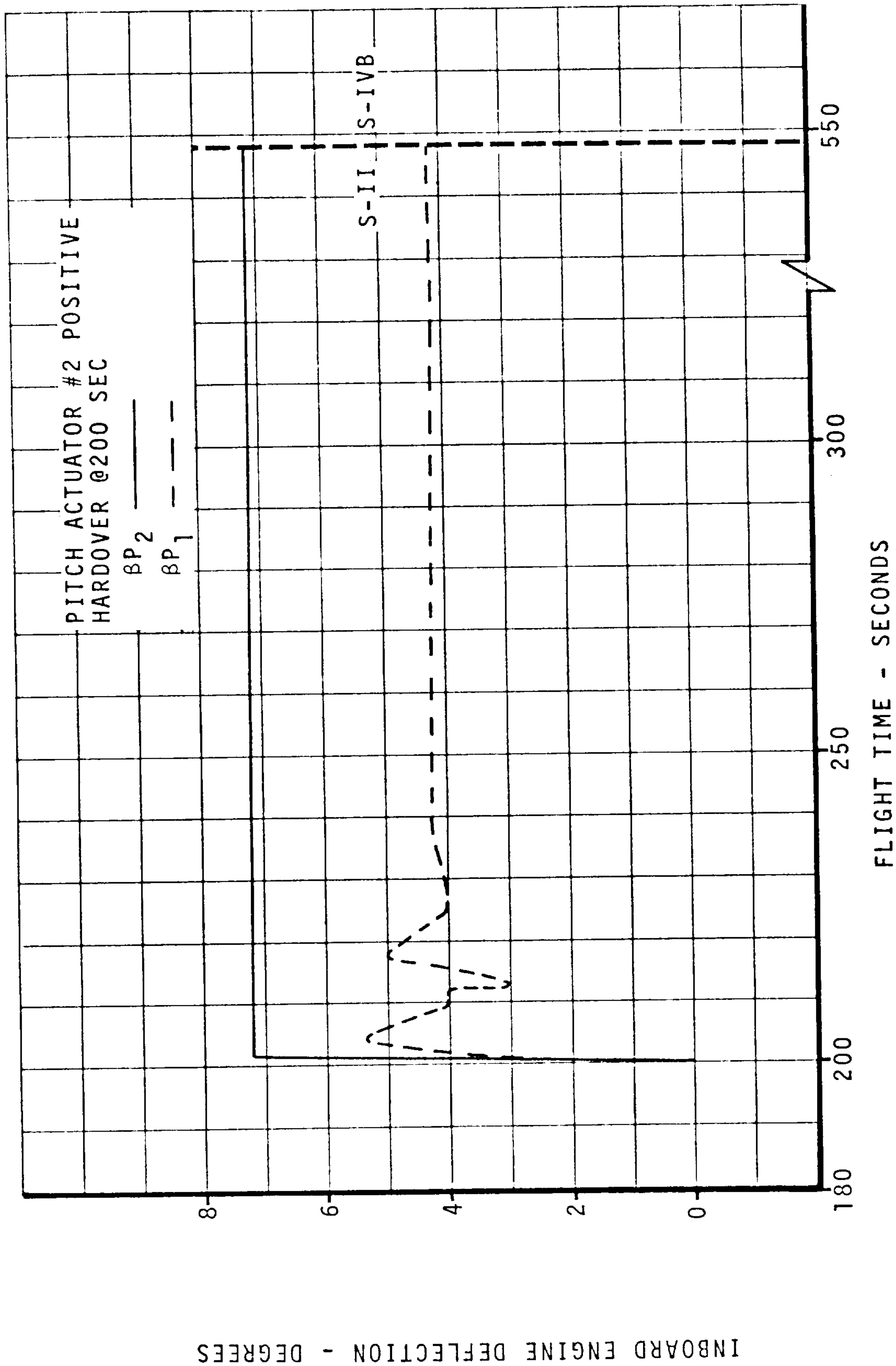


FIGURE 4-33 S-II PITCH ENGINE RESPONSE TO AN ACTUATOR HARDOVER

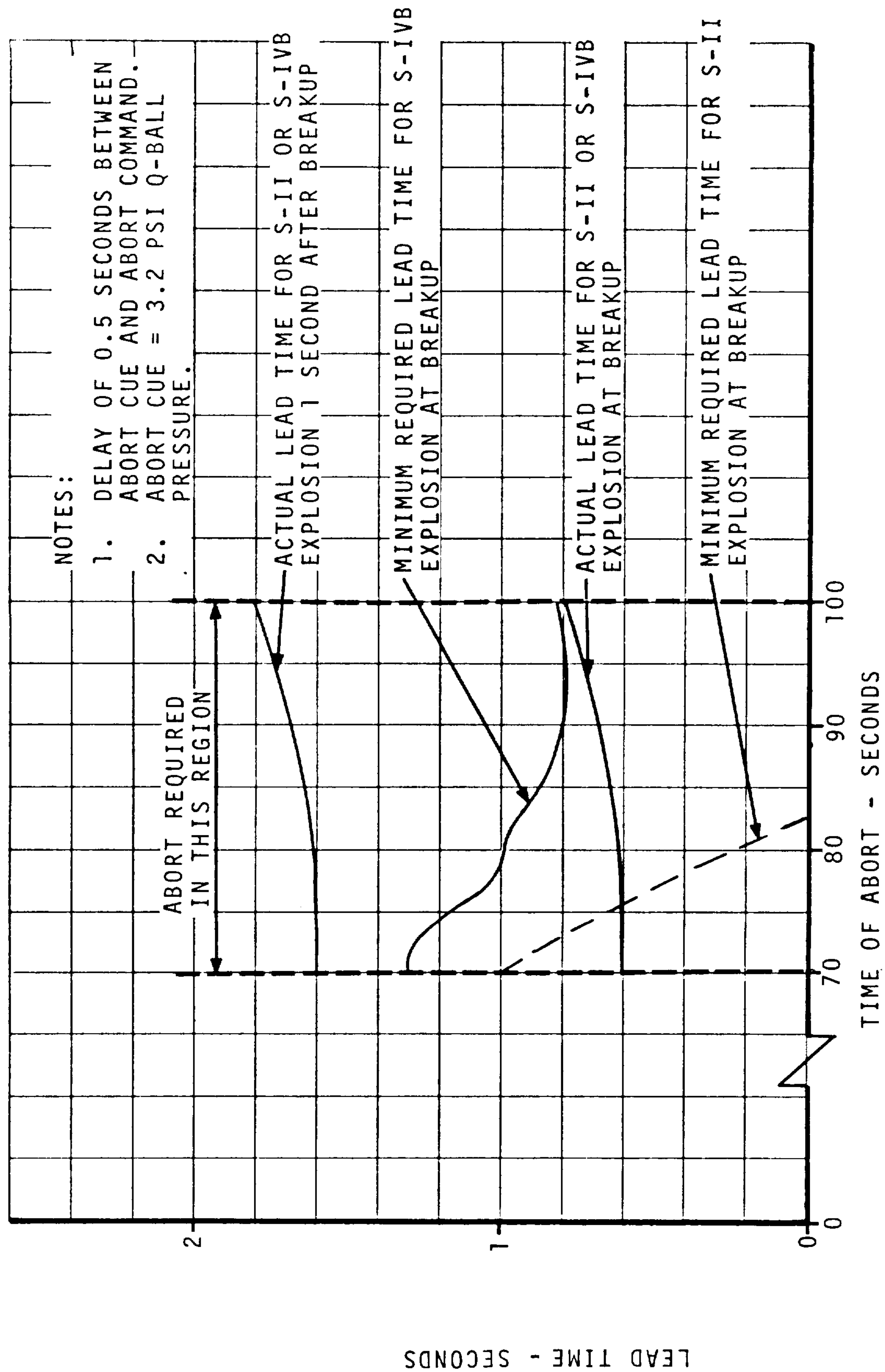


FIGURE 4-34 LEAD TIMES FOR ABORT AFTER AN S-IC ACTUATOR HARDOVER

4.4 SINGLE ACTUATOR INOPERATIVE

4.4.1 Malfunction

An actuator inoperative malfunction is any failure which causes a single actuator to remain at or near null (±0.7 degree) regardless of control commands or external forces placed on the actuator by the engine.

4.4.2 Malfunction Dynamics

In S-IC and S-II flight, only minor dynamics due to the vehicle rotating to a trim attitude result from actuator failures to null.

In S-IVB flight, an inoperative actuator causes attitude divergence in the plane of the failed actuator. Figure 4-35 shows a typical case for a null pitch actuator. As the failed actuator position approaches the limit for this malfunction (±0.7 degree of null position), vehicle divergence becomes more rapid.

4.4.3 EDS Effectiveness and Crew Safety

Abort is not necessary for an actuator inoperative during S-IC and S-II flights. First abort cue for S-IVB first and second burns is attitude deviation. Second cue is one of the following:

a. Exceeding the ±45 degree yaw attitude limit
b. Exceeding the rate limits
c. Ground confirmation of attitude deviation (abort request light) if received prior to the above.

Abort lead time is important for yaw actuator inoperative because of the spacecraft platform yaw tumble limit of 90 degrees. Figure 4-36 shows sufficient lead time exists for abort based upon the above cues.

4.4.4 Mission Completion Capability

Full mission completion capability is maintained for an actuator inoperative during S-IC and S-II flight.

Abort is required during S-IVB first and second burns if the malfunction occurs earlier than approximately 30 seconds before nominal cutoff. If the malfunction occurs after this time in second burn, full mission completion capability is maintained; if it occurs before this time, TLI capability is lost.

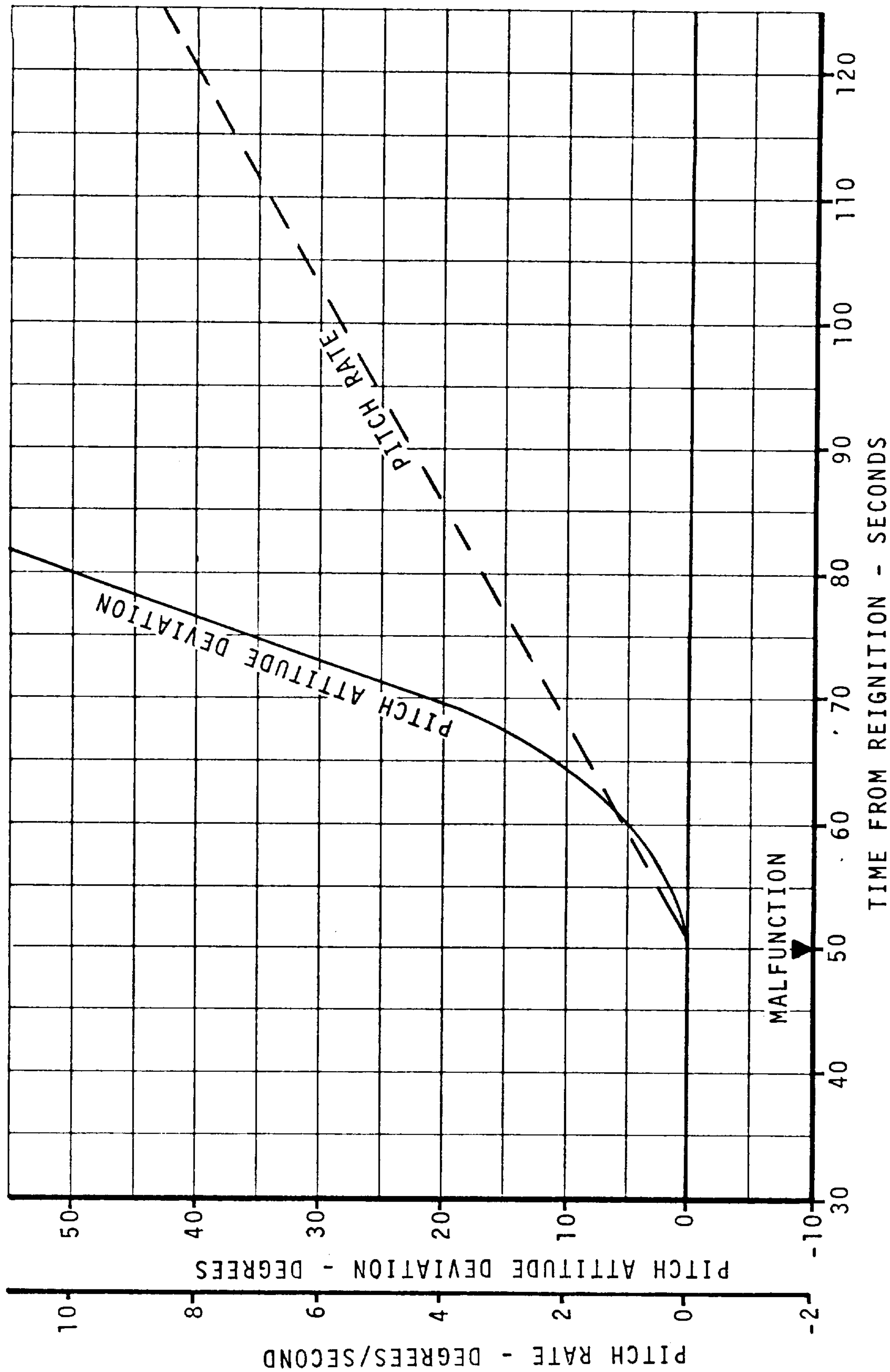


FIGURE 4-35 DIVERGENCE CAUSED BY PITCH ACTUATOR NULL IN S-IVB STAGE

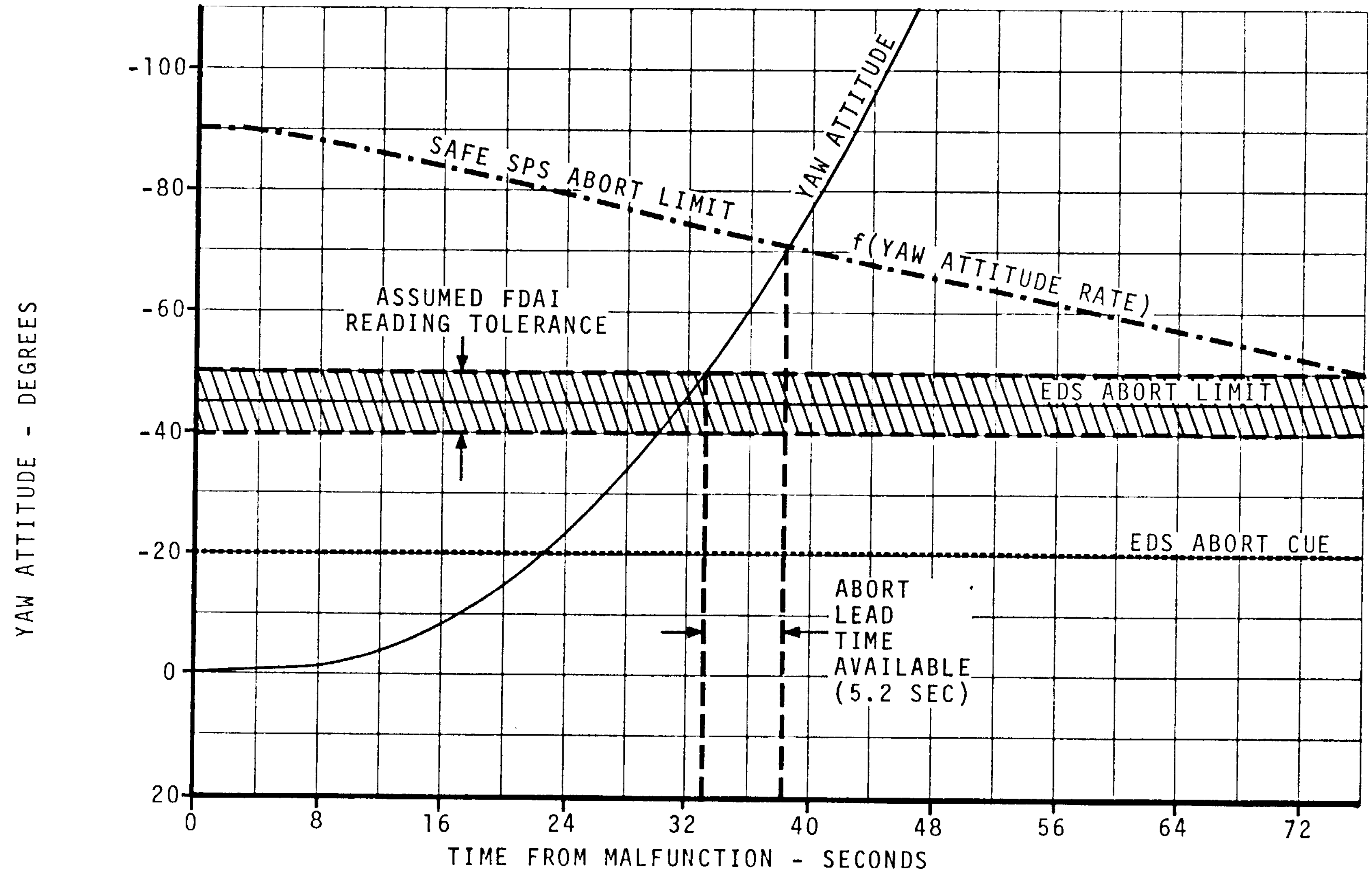


FIGURE 4-36 ABORT LEAD TIME FOR YAW ACTUATOR NULL IN S-IVB FLIGHT

4.5 SATURATED ERROR SIGNAL

4.5.1 Malfunction Description

Any failure that causes an erroneous 15.3 degree attitude error signal is a saturated attitude error.

4.5.2 Malfunction Dynamics

Figure 1-1 shows the times of malfunction that cause pad fallback, liftoff interference, and tower collision. Since the engines are not shut down in the abort sequence before 30 seconds, a saturated attitude error can turn the vehicle and cause it to impact in the pad area.

Saturated attitude error in S-IC flight causes tumbling at all flight times. Typical dynamics for this malfunction are shown in Figure 4-37. Worst case launch vehicle tension loads that result from a saturated control signal in the high-q region are shown in Figure 4-38.

Saturation of the pitch or yaw error signal in the S-II stage initially causes actuators hardover in the affected plane; actuators hardover cause the vehicle attitude rate to increase until the control equation ($\beta = a_o\psi + a_1\dot{\phi}$) becomes balanced. This occurs when the actuators approach the null position, and the attitude rate reaches a maximum value and remains at that value. Shown below are the values used to calculate the maximum rate for S-II saturated error signal for the first set of S-II gains:

$$0 = (1.12)(15.3) + (1.89)\ \dot{\phi}_{MAX}\ \text{(pitch and yaw)}$$

$$\dot{\phi}_{MAX}\ \text{(pitch and yaw)} = -9.1\ \text{degrees/second}$$

This maximum vehicle rate does not violate the EDS rate limit of 10 degrees/second. However, the launch vehicle overrate light may be activated due to the ±0.8 degree tolerance on the ±9.2 degrees/second limit setting. Attitude deviation of 20 degrees is exceeded within 5 seconds after time of malfunction. Saturated roll error signal causes a large engine deflection in both pitch and yaw planes. The vehicle roll rate reaches a value which stabilizes the control equation. This value is calculated below:

$$0 = (.25)(15.3) + (.2)\ \dot{\phi}\ \text{(roll)}$$

$$\dot{\phi}_{MAX}\ \text{(roll)} = -19\ \text{degrees/second}$$

4.5.2 (Continued)

Saturated error signal in S-IVB produces the same effects as those described for the S-II. Because of the difference in control gains a_o and a_1, however, the EDS rate limit of 10 degrees/second is exceeded for pitch and yaw malfunctions.

$$0 = (.81)(15.3) + (.97)\ \dot{\phi}_{MAX} \text{ (pitch and yaw)}$$

$$\dot{\phi}_{MAX} \text{(pitch and yaw)} = -12.8 \text{ degrees/second}$$

$$0 = (1)(15.3) + (1)\ \dot{\phi}_{MAX} \text{ (roll)}$$

$$\dot{\phi}_{MAX} \text{ (roll)} = -15.3 \text{ degrees/second}$$

Figure 4-39 shows S-II and S-IVB pitch rate time histories resulting from a saturated error signal malfunction.

4.5.3 EDS Effectiveness and Crew Safety

Failures prior to 1.25 seconds result in the engines colliding with the holddown posts. Crew loss is assumed for these cases, since there are no valid cues for this situation. Failures after 1.25 seconds result in aborts on angular rate cues. Abort lead times for saturated error signal malfunctions during S-IC flight are shown in Figure 4-40 . The figure shows a comparison of actual lead time and required lead time for the most critical period of flight, from 30 seconds to 90 seconds. Lead times during the remaining period of S-IC flight are much greater than required. The figure indicates that if an explosion occurs one second after breakup, the crew would be unsafe from explosions following aborts occuring between 30 and 65 seconds.

In S-II, a launch vehicle overrate light may be used as a crew abort cue; however, due to the balancing of the control equation, as described in Paragraph 4.5.2, a marginal condition exists for the overrate light cue, necessitating reliance on ground cues. Abort is by the launch escape system prior to tower jettison and by the SPS after jettison; abort cues are the abort request light and FDO limits.

Cues for saturated error signal in S-IVB are the same as those in S-II. No automatic abort is provided for this malfunction in S-II or S-IVB stages.

4.5.4 Mission Completion Capability

Saturated error signal in any stage of flight results in mission loss.

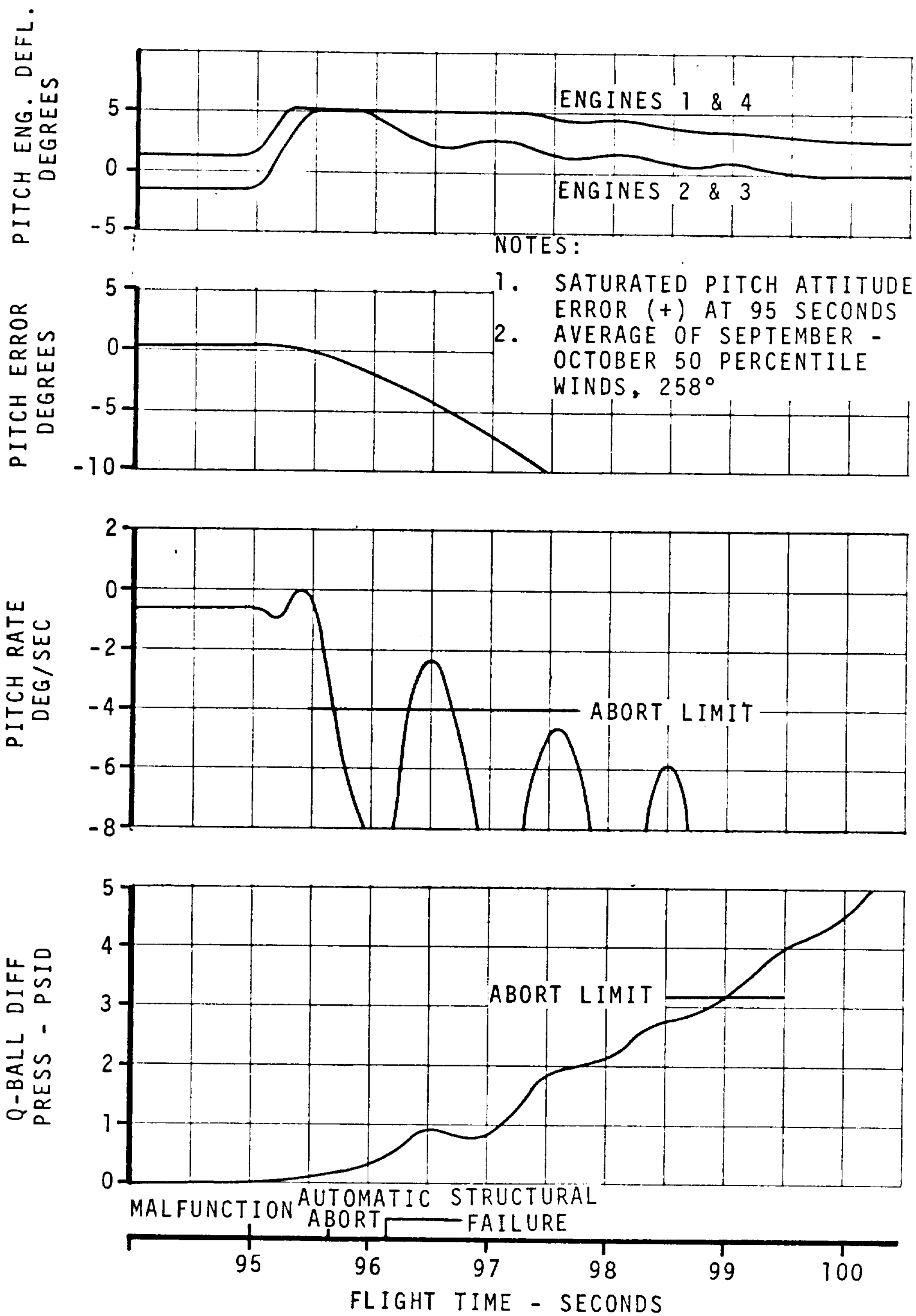


FIGURE 4-37 TIME HISTORY OF DYNAMIC VARIABLES FOR SATURATED ERROR SIGNAL (S-IC)

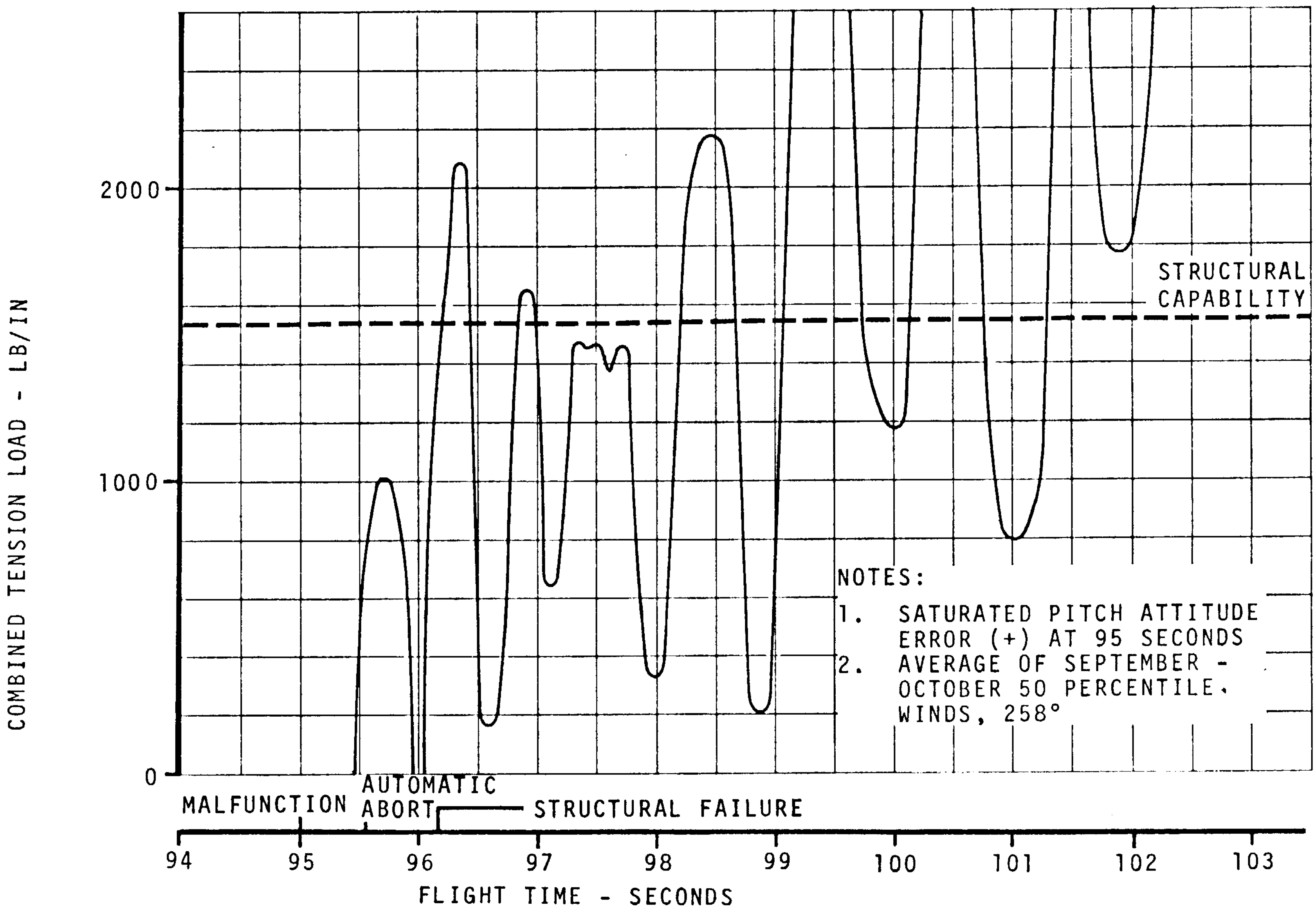


FIGURE 4-38 TENSION LOAD AT STATION 2832 AFT (SATURATED ERROR SIGNAL)

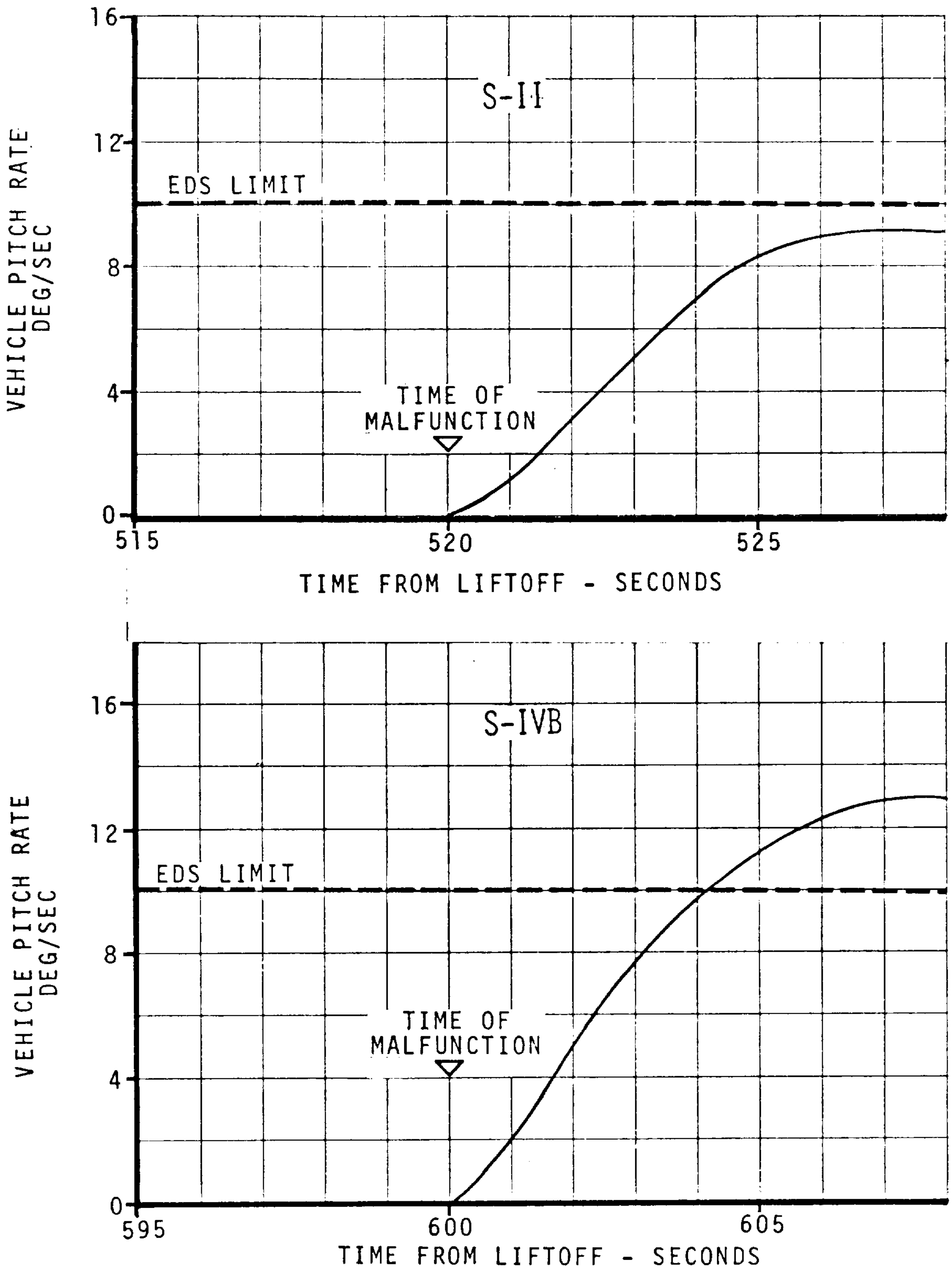


FIGURE 4-39 VEHICLE PITCH RATE HISTORY FOLLOWING SATURATED ERROR SIGNAL IN S-II AND S-IVB

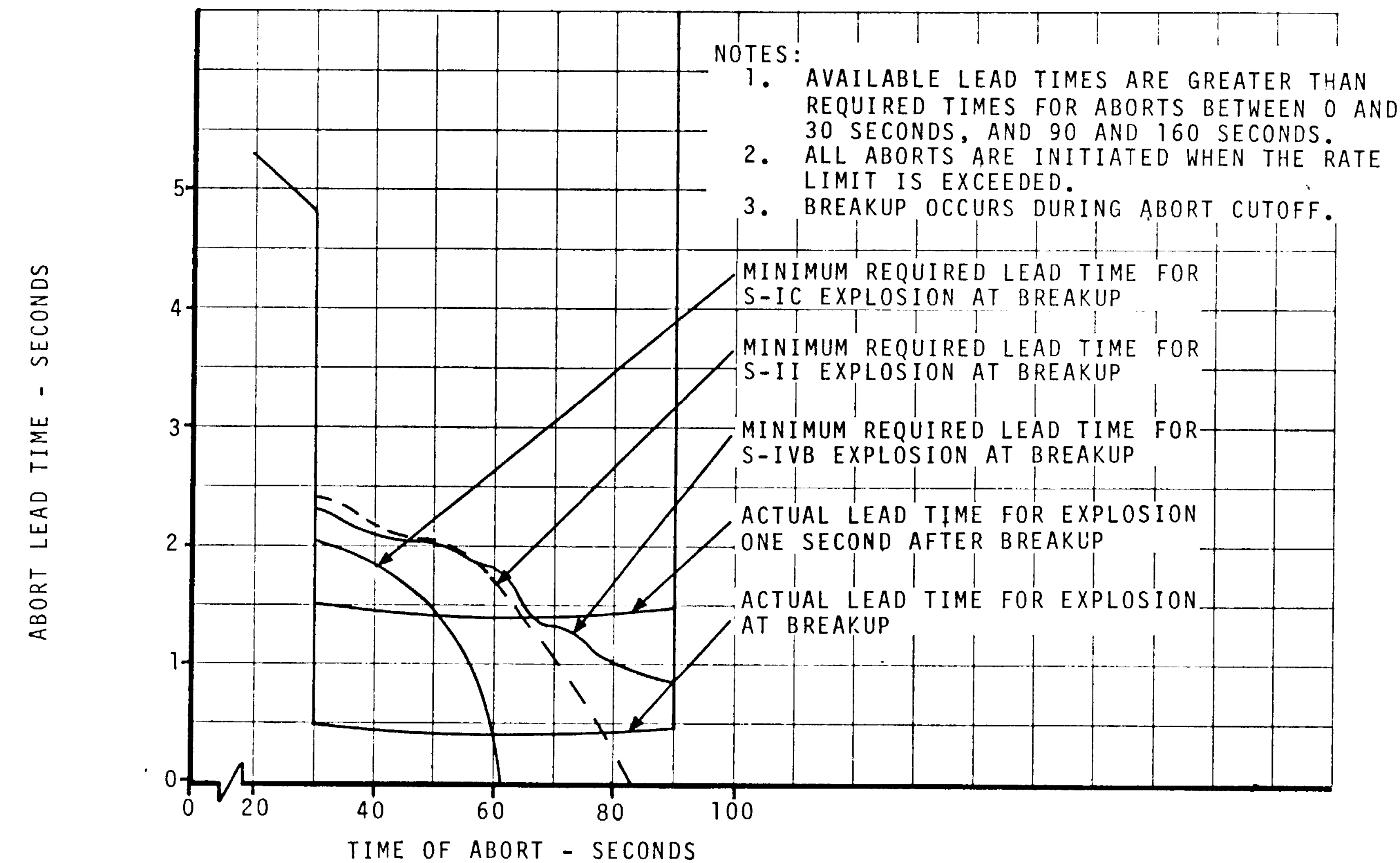


FIGURE 4-40 ABORT LEAD TIMES FOR ABORTS FOLLOWING A SATURATED ATTITUDE ERROR MALFUNCTION IN S-IC

4.6 SATURATED RATE SIGNAL

4.6.1 Malfunction Description

Any failure that causes a false attitude rate signal of 10 degrees/second to be sent to the flight control computer is referred to as a saturated attitude rate signal.

4.6.2 Malfunction Dynamics

Saturated rate signal causes loss of control at all times during S-IC flight. Typical dynamics following this malfunction are shown in Figure 4-41.

Worst case launch vehicle tension loads that result from a saturated rate signal in the high-q region are shown in Figure 4-42.

A saturated pitch or yaw rate signal in the S-II stage causes the actuators in the affected plane to go hardover and the attitude error increases until it reaches the limit of 15.3 degrees. The actuators then return the control engines to a deflection of between 1 to 2 degrees, which results in uncontrolled tumbling. A similar situation occurs for a saturated roll rate signal. Figure 4-43 shows a vehicle pitch rate history for a saturated rate signal during S-II flight.

The effects of a saturated rate signal during S-IVB flight are similar to those in S-II. Figure 4-43 shows an S-IVB pitch rate history resulting from a saturated pitch rate signal.

4.6.3 EDS Effectiveness and Crew Safety

Failures prior to 1.25 seconds result in the engines colliding with the holddown posts. Crew loss is assumed for these cases, since there are no valid cues for this situation. Failures after 1.25 seconds result in aborts on angular rate cues.

EDS abort cues and logic for a saturated rate signal are identical to those for a saturated attitude error (see Paragraph 4.5.3).

4.6.4 Mission Completion Capability

Saturated rate signal in any stage of flight results in mission loss.

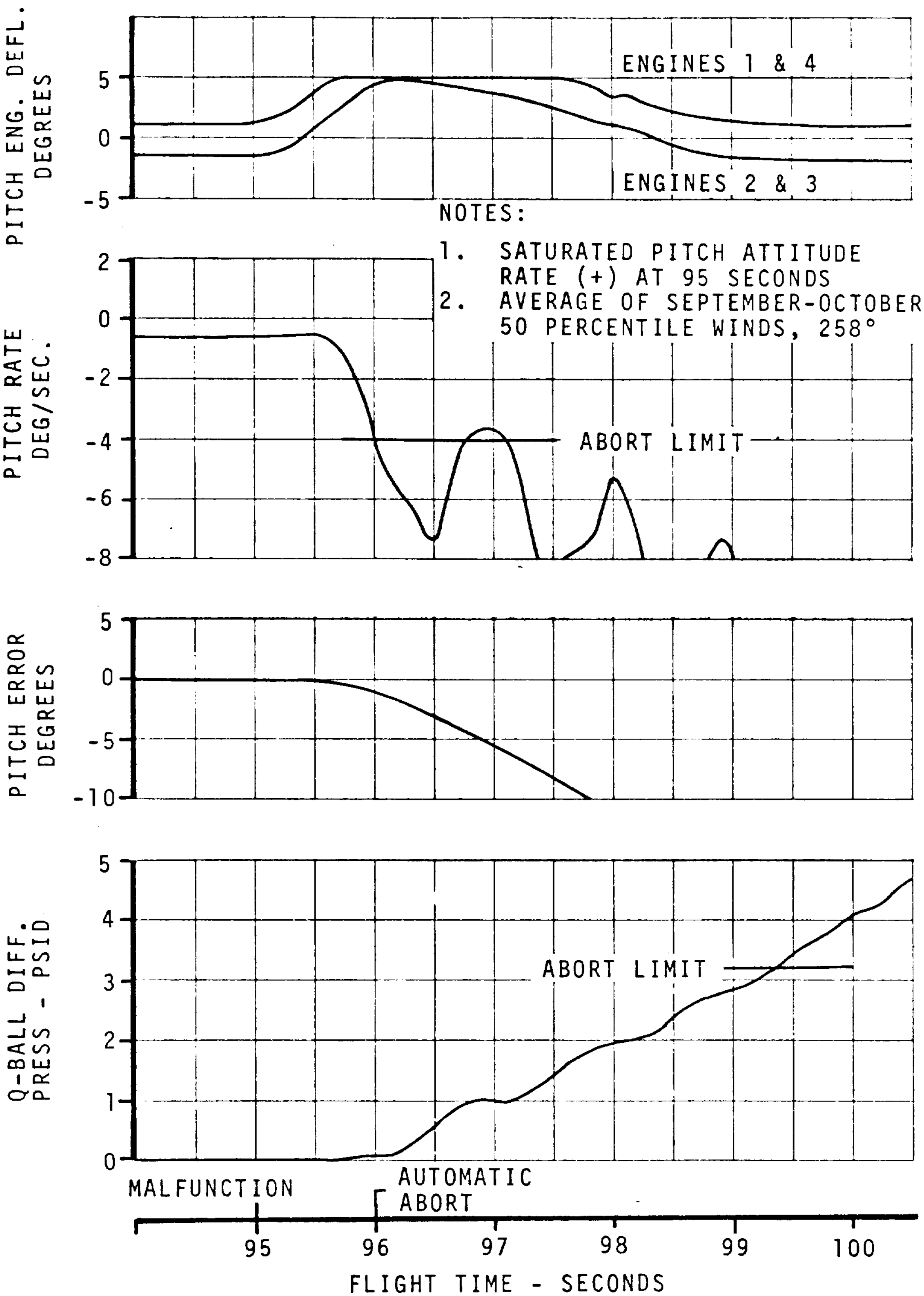


FIGURE 4-41 TIME HISTORY OF DYNAMIC VARIABLES FOR SATURATED RATE SIGNAL(S-IC)

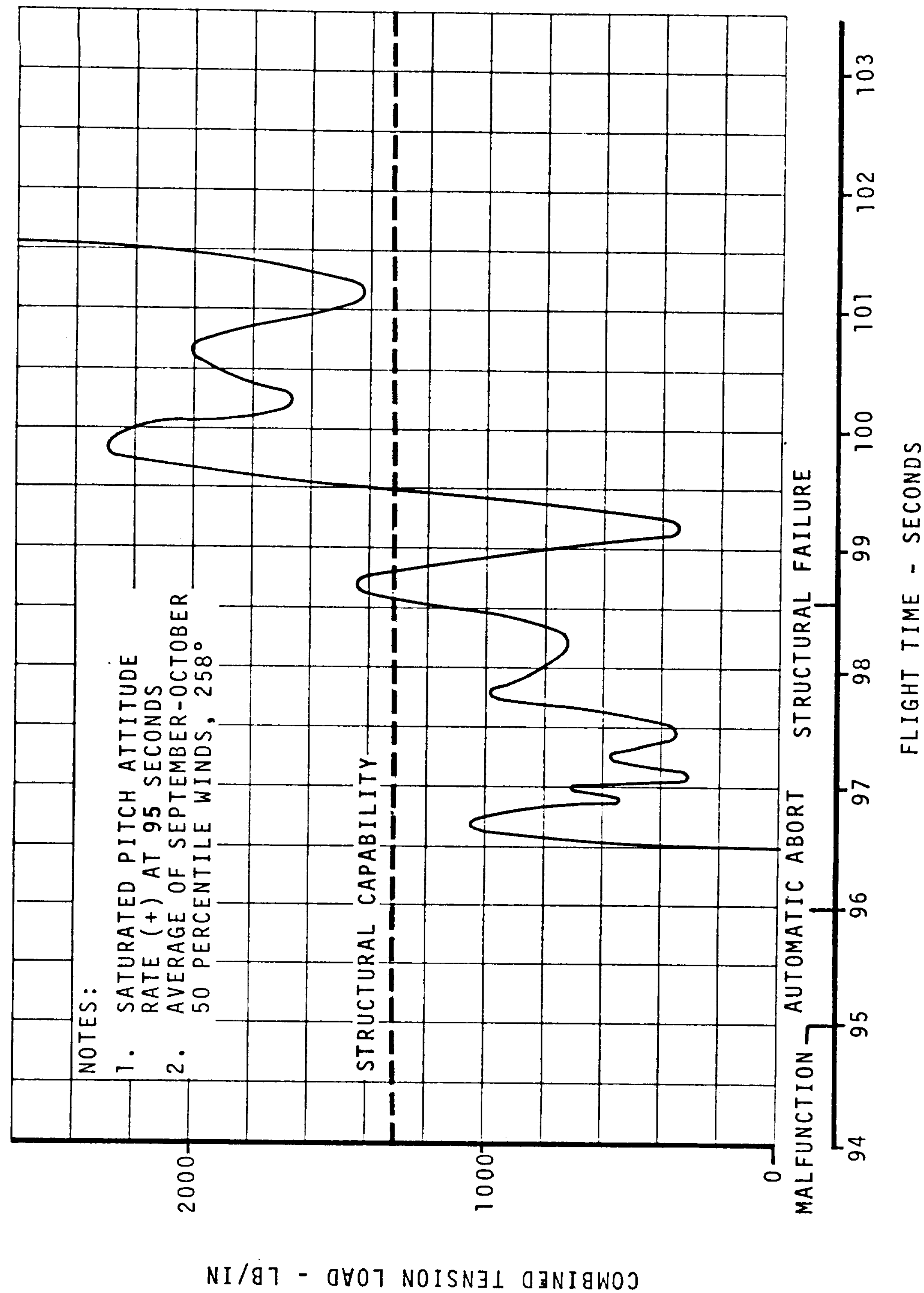


FIGURE 4-42 TENSION LOAD AT STATION 1760 (SATURATED RATE SIGNAL)

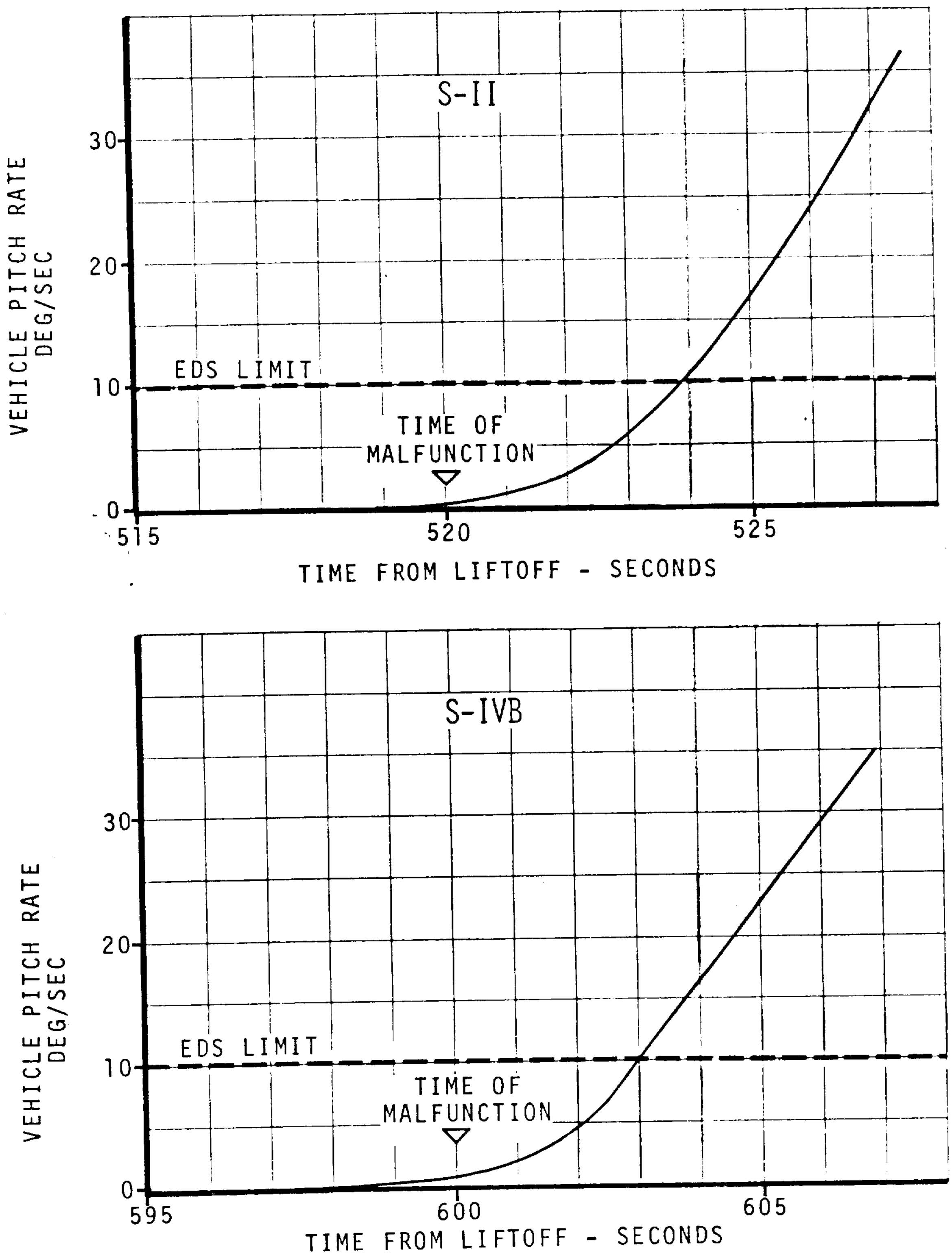


FIGURE 4-43 VEHICLE PITCH RATE HISTORY FOLLOWING SATURATED RATE SIGNAL IN S-II AND S-IVB

4.7 LOSS OF INERTIAL ATTITUDE

4.7.1 Malfunction Description

An ST-124 inertial platform, located in the Instrument Unit, is used as the inertial attitude reference during boost. Platform outputs are monitored by the LVDC and tested for reasonableness. On successive failures of the reasonableness test, the LVDC issues a guidance failure discrete. This discrete lights the LV GUID light on the spacecraft instrument panel. It also turns on the LV overrate light prior to automatic abort deactivation.

A backup system permits spacecraft takeover of guidance on AS-507. Switchover to this backup system is performed manually upon detection of the LV GUID light.

In this analysis it is assumed that the platform fails in either the pitch, yaw, or roll plane at a constant angular acceleration or at a constant drift rate. The gimbal reasonableness test is failed when the difference between successive gimbal readings indicates a rate of change of gimbal angle greater than 15 degrees per second. Figure 4-44 shows a typical attitude error time history during the guidance switchover sequence. Time t_1 depends on the angular acceleration of the failed platform. Times between events are based on the sequence given in Reference 7.

4.7.2 Malfunction Dynamics

The vehicle attempts to follow a failed platform. Thus, a high acceleration failure results in the vehicle accelerating rapidly to follow it. Body rates in excess of abort rate limits, which in this study are assumed to be equivalent to loss of control, are thus reached in a very short time. In all cases studied, aborts on rate preceded aborts on q-ball $\Delta P\alpha$. Guidance switchover following platform failures in roll can be accomplished throughout flight without loss of control.

Figures 4-45, 4-46, and 4-47 show the minimum platform accelerations that are required to maintain control after guidance switchover following positive pitch, negative pitch, and yaw platform failures, respectively.

Sensed attitude error at guidance switchover is not a dependable abort cue since loss of control can result for an error as small as 1.0 degree. Figures 4-48 and 4-49 show the minimum attitude errors that can result at switchover for cases that lose control. Errors are much larger during the manual EDS mode because of the 10 degree per second abort rate limit.

4.7.2 (Continued)

Minimum accelerations for safe switchover during S-II flight cannot be defined because no dependable onboard EDS limits are violated prior to activation of the LV GUID light. S-II platform failure detection delays cause rapid buildup of attitude errors to the saturation limit of 15.3 degrees. This results in actuator trimming at vehicle rates just below the 10 degree/second rate limit as described in Section 4.5. Excessive attitude deviations at the time of LV GUID light activation can cause violation of FDO limits subsequent to S/C guidance switchover.

The maximum platform failure delay in S-IVB flight without violating EDS abort limits is approximately 1.7 seconds. Rate and error control gains in S-IVB flight do not cause actuator trimming below the EDS rate abort limits as in S-II.

Very slow platform failures that do not violate the guidance failure tests can result in off-nominal orbits. Figure 4-50 shows maximum platform drift rates, with resulting degradation of accelerometer data, that the vehicle can tolerate and still attain a 75-100 nautical mile parking orbit. All non-immediate platform failures with drift rates in excess of those defined in Figure 4-50 and with guidance failure delays exceeding 1.7 seconds in S-IVB result in loss of vehicle.

4.7.3 EDS Effectiveness and Crew Safety

As is shown by lead times in Figure 4-51, existing EDS automatic and manual abort limits are adequate to ensure crew safety during the transition from primary guidance to S/C guidance. Slow platform failures in S-II may not violate onboard EDS limits. These failures must rely on ground monitoring of trajectory for violation of FDO limits and attitude deviations.

4.7.4 Mission Completion Capability

The launch vehicle can be guided to a safe parking orbit (perigee altitude greater than 75 NMI) in the S/C backup guidance mode and TLI can be achieved. Performance losses due to manual guidance to insertion and translunar injection are within the AS-507 reserves. An injection is achieved requiring less than a 305 meter/second (1000 feet/second) midcourse correction budget. These results are based on the study of Reference 7.

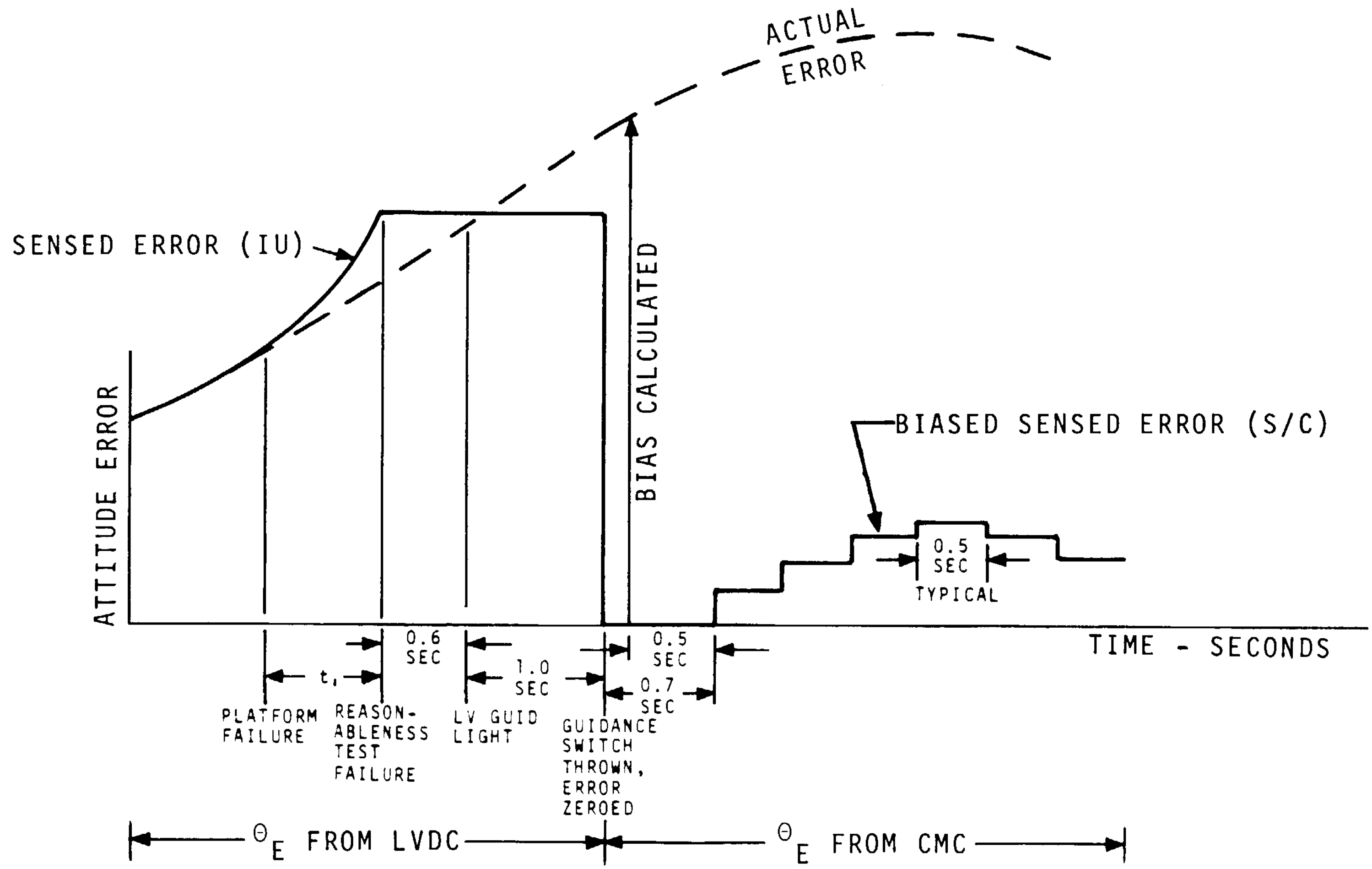


FIGURE 4-44 GUIDANCE SWITCHOVER SEQUENCE

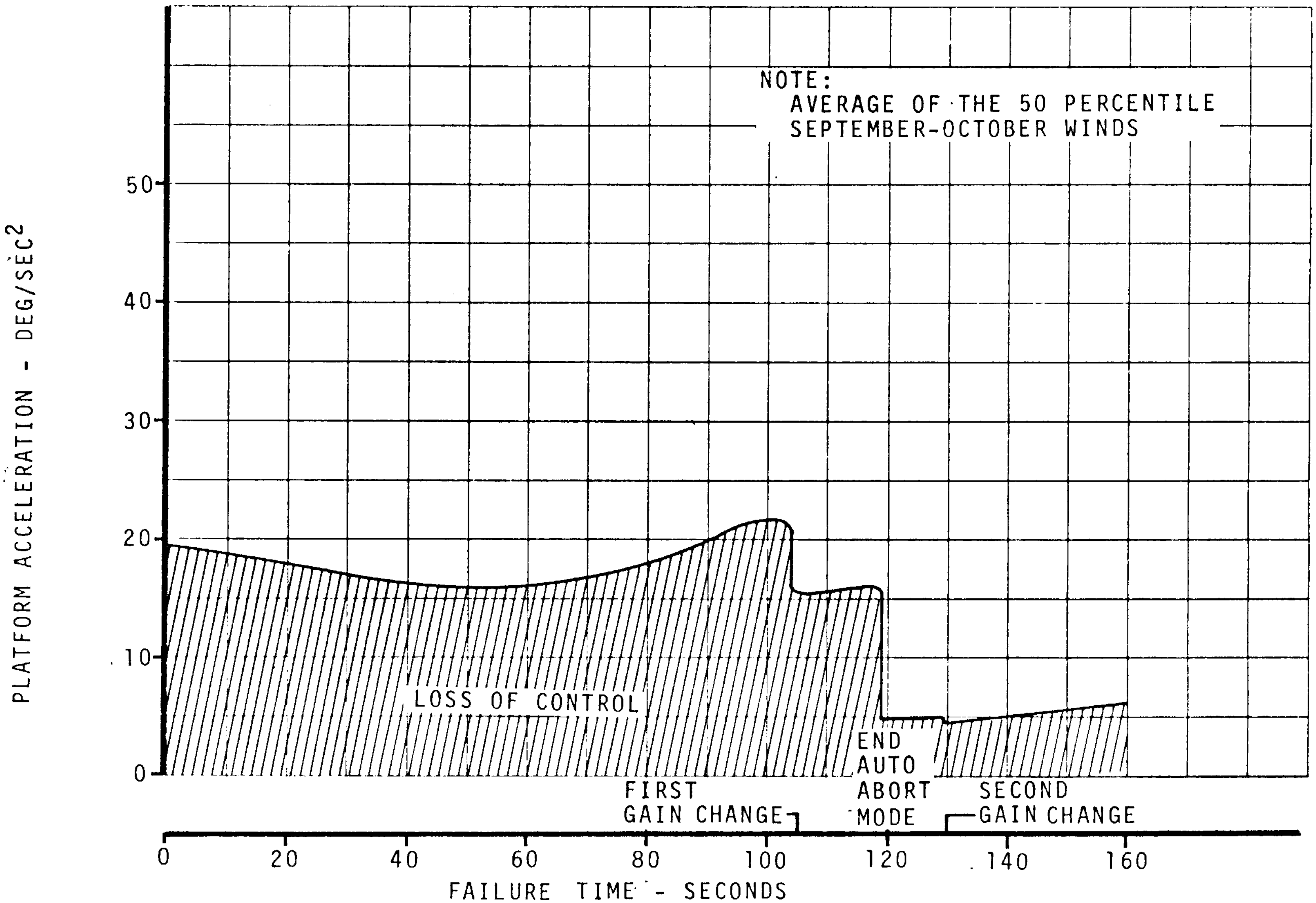


FIGURE 4-45 POSITIVE PITCH PLATFORM FAILURE CAPABILITY WITH SPACECRAFT BACKUP

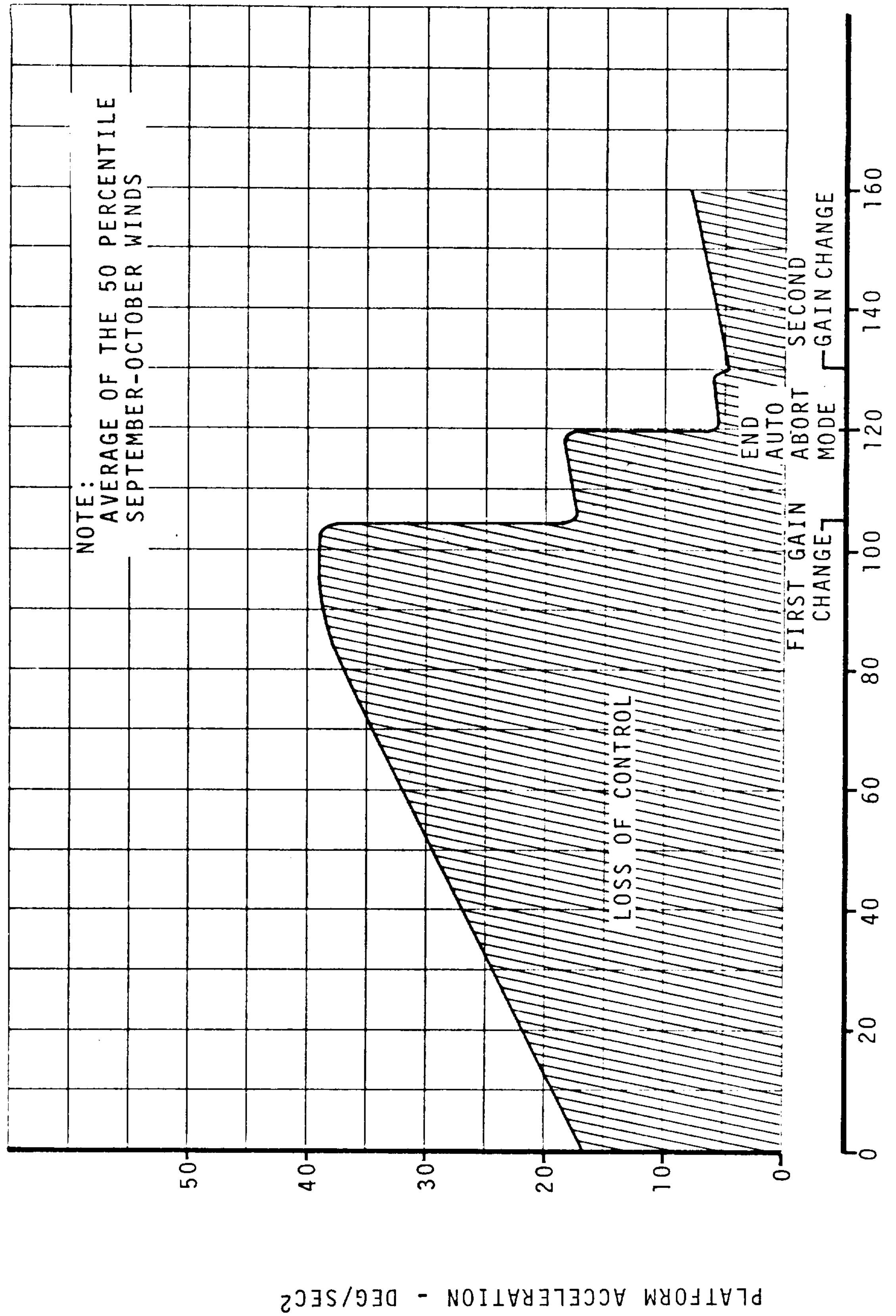


FIGURE 4-46 NEGATIVE PITCH PLATFORM FAILURE CAPABILITY WITH SPACECRAFT BACKUP

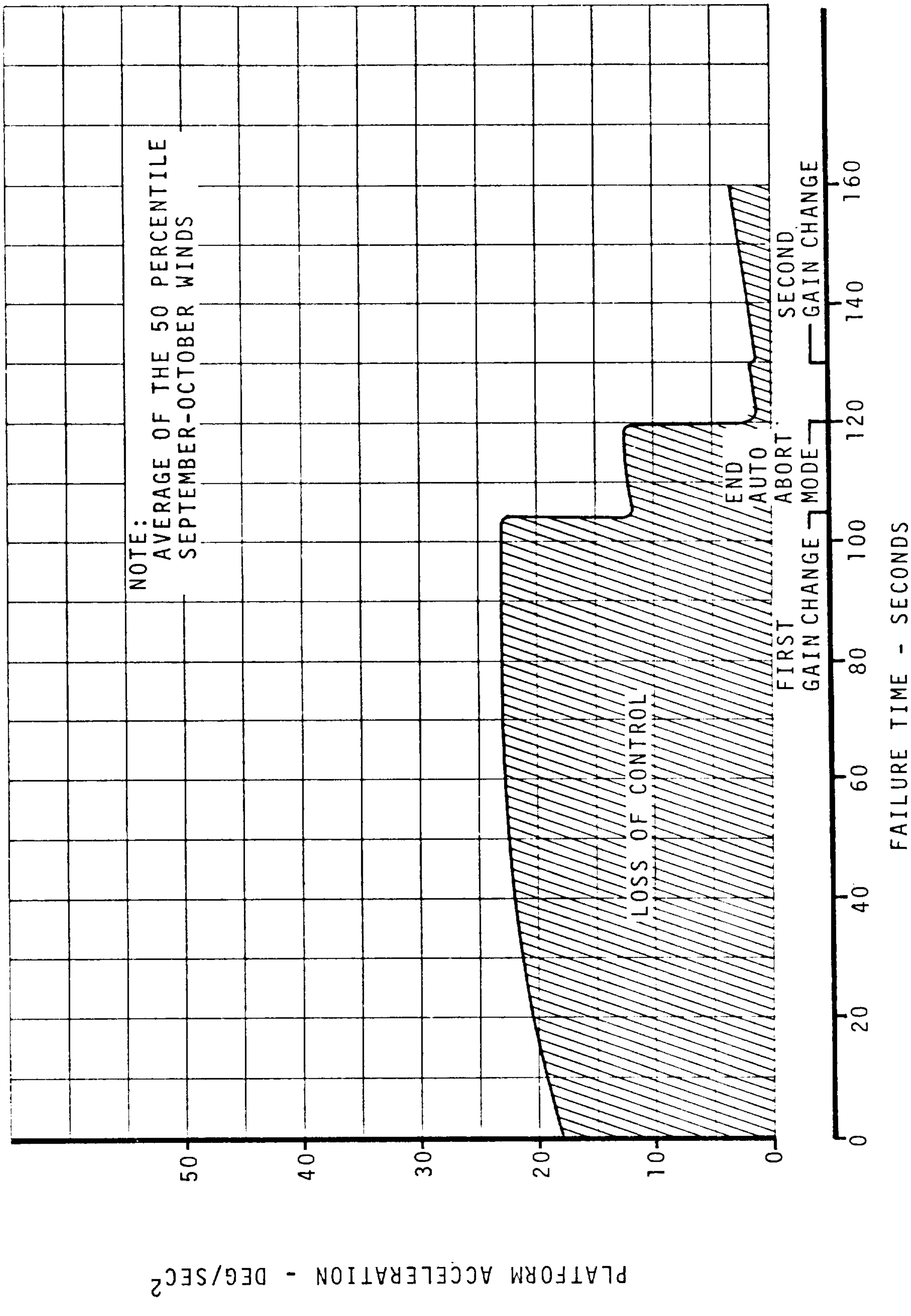


FIGURE 4-47 YAW PLATFORM FAILURE CAPABILITY WITH SPACECRAFT BACKUP

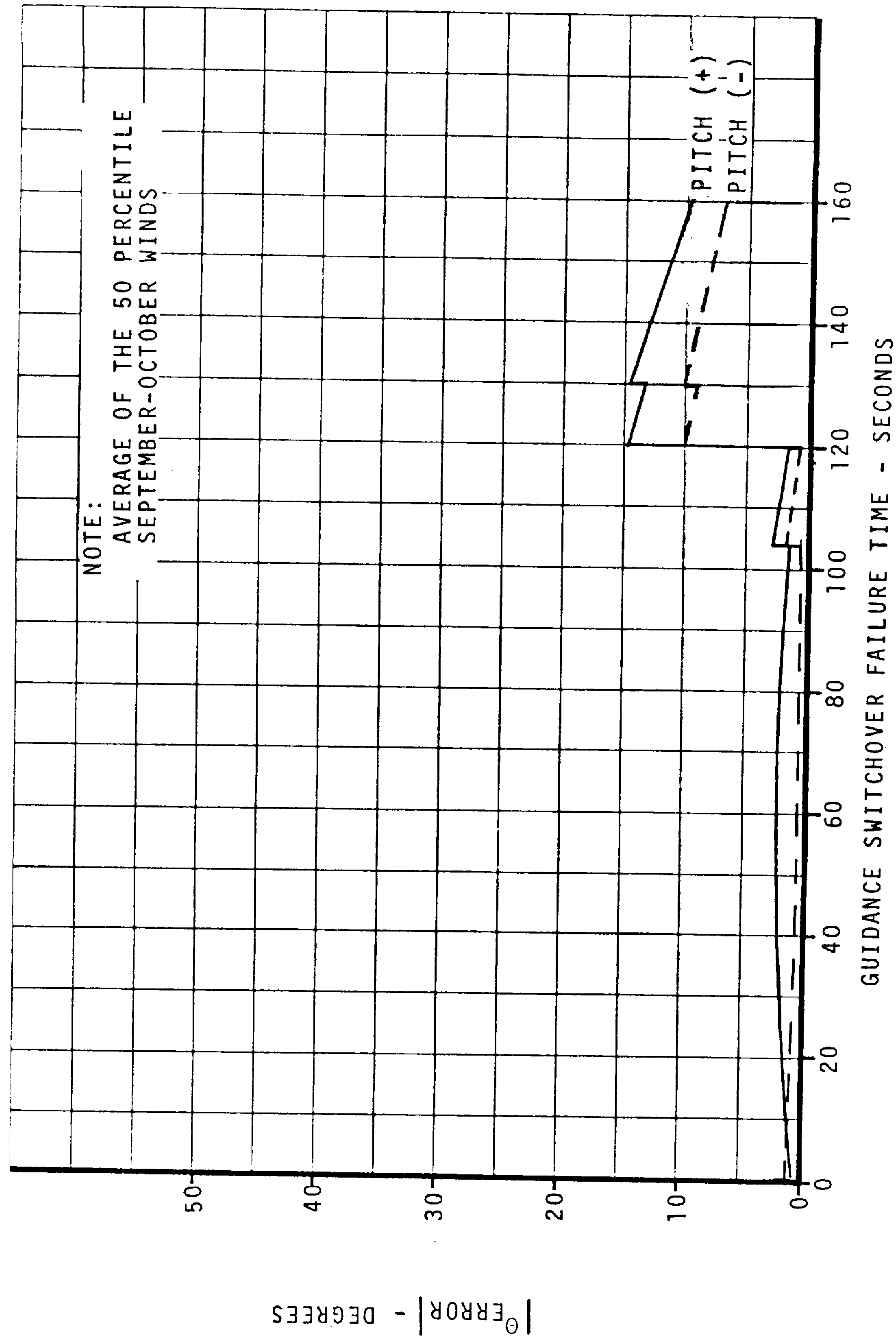


FIGURE 4-48 MINIMUM PITCH ATTITUDE ERRORS AT SWITCHOVER FOR PLATFORM FAILURES THAT CAUSE LOSS OF CONTROL

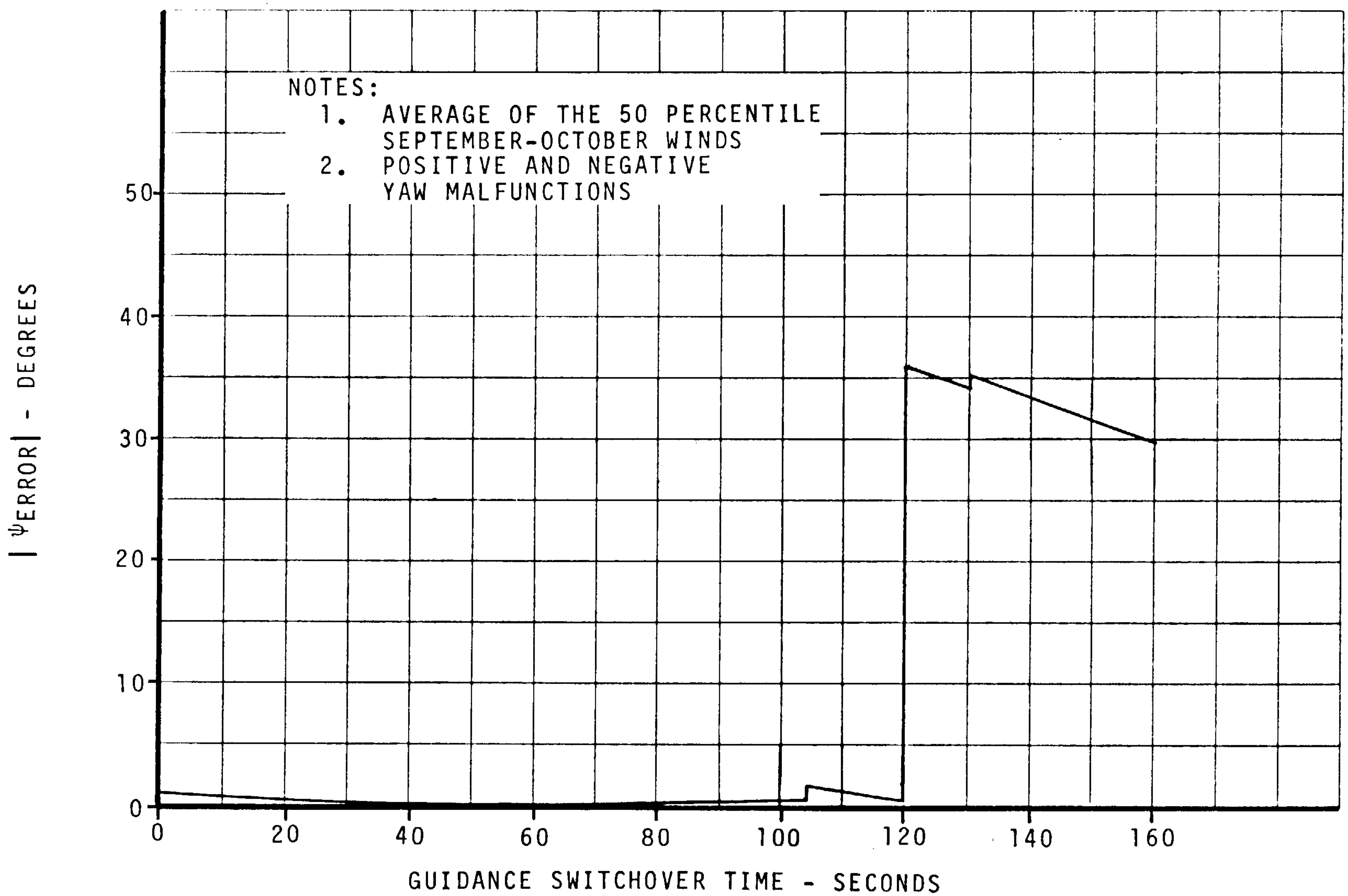


FIGURE 4-49 MINIMUM YAW ATTITUDE ERRORS AT SWITCHOVER FOR PLATFORM FAILURES THAT CAUSE LOSS OF CONTROL

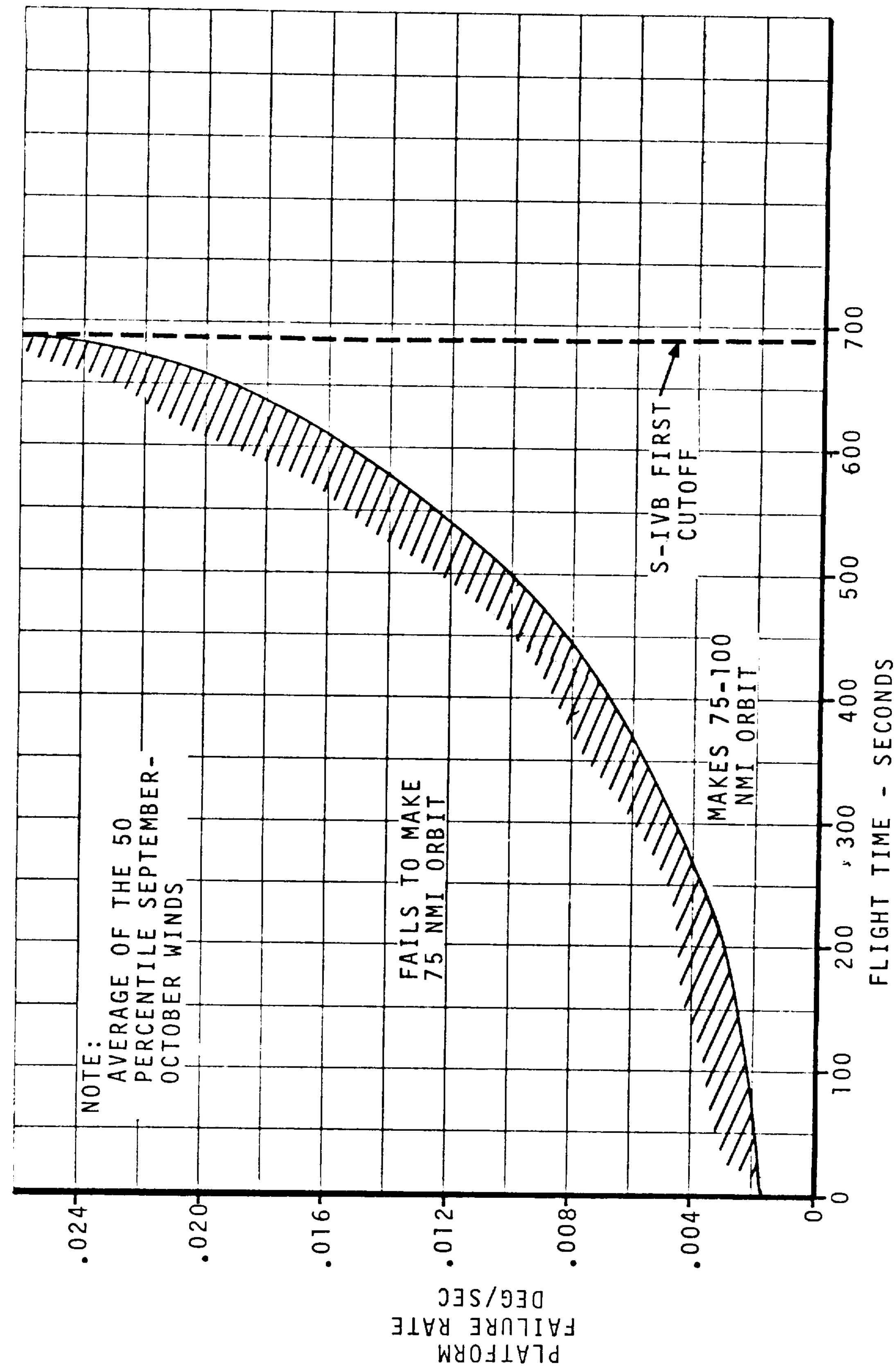


FIGURE 4-50 MAXIMUM TOLERABLE DRIFT RATE WITH 75 NMI ORBIT CAPABILITY

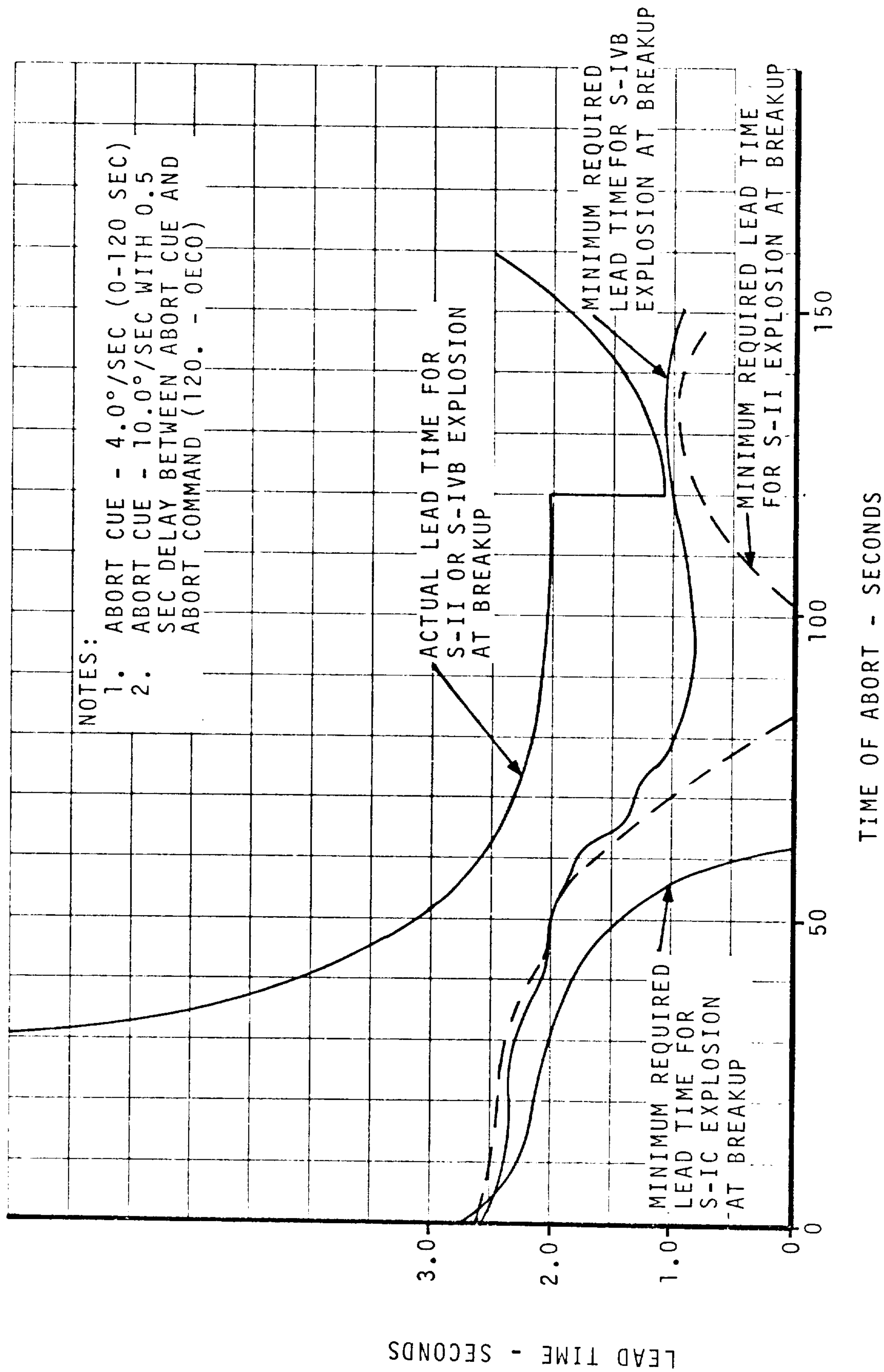


FIGURE 4-51 LEAD TIMES FOR ABORT AFTER S-IC LOSS OF PLATFORM

4.8 LOSS OF ATTITUDE COMMAND

4.8.1 Malfunction Description

Loss of attitude command includes all failures that result in the attitude command remaining constant or becoming zero at the time of failure.

4.8.2 Malfunction Dynamics

Loss of roll attitude command can occur at all flight times without causing mission loss.

Loss of yaw attitude command prior to approximately 350 seconds can result in excessive out-of-plane conditions such that IGM fails to shut down the S-IVB engine at POI velocity. This necessitates shutdown by ground command.

Constant pitch attitude command failures cause mission loss during all flight stages except during the latter portions of S-IVB first and second burns.

Failure of major loop pitch command to zero causes a one degree/second pitch-up for boost-to-orbit. This malfunction causes vehicle loss except for late in S-IVB burns. If the failure occurs prior to 90 seconds in S-IC flight, rapid vehicle tumbling results.

Failure of minor loop pitch command to zero causes a 12 degrees/second pitch-up resulting in loss of control during S-IC flight. Failures result in large attitude deviations during S-II and S-IVB burns. Vehicle loss occurs for all failure times.

4.8.3 EDS Effectiveness and Crew Safety

The abort cues for loss of yaw and pitch attitude command failures are:

a. Attitude deviation
b. Exceeding FDO limits, or ground confirmation of attitude deviation (abort request light)

Failure of the major loop pitch command to zero prior to 90 seconds results in automatic abort based on 4 degrees/second pitch rate. Manual abort cues are:

a. Pitch attitude deviation
b. Exceeding FDO limits or ground confirmation of attitude deviation (abort request light)

4.8.3 (Continued)

Failure of the minor loop pitch command to zero before 120 seconds flight time results in automatic abort based on 4 degrees/second pitch rate. After that time, manual abort cues are 10 degrees per second rate, and ±5 degrees attitude deviation during S-IC flight or ±20 degrees attitude deviation during S-II or S-IVB flight.

False abort or crew loss does not occur for loss of attitude command malfunctions.

4.8.4 Mission Completion Capability

Mission completion capability exists for loss of roll command at all flight times. Parking orbit insertion is achieved for loss of yaw command failures after 350 seconds. Mission loss occurs for a failure of pitch command to zero in boost-to-orbit. For a constant pitch command occurring after 580 seconds of flight time, POI is achieved, but TLI is not possible. In S-IVB second burn, TLI is achieved for failure of pitch command to zero after 320 seconds from reignition and for constant pitch command after 270 seconds from reignition.

4.9 LOSS OF ATTITUDE ERROR SIGNAL

4.9.1 Malfunction Description

Any failure that causes a false attitude error signal of zero degrees to the flight control computer is referred to as loss of attitude error signal.

4.9.2 Malfunction Dynamics

For any malfunction time, loss of attitude error causes attitude divergence. Under pitch or yaw rate control only, the control system positions the thrust vector such that it compensates for angular disturbances. This results in a vehicle rate dictated by the control law. During flight through the atmosphere, aerodynamic moments resulting from increasing angle-of-attack cause increasing rate. In the high-q region, uncontrolled tumbling followed by structural failure occurs when the aerodynamic moment exceeds control authority. During upper stage flight, attitude divergence is slower and no structural breakup occurs. Pitch and yaw failures cause loss of vehicle except for the latter portion of S-IVB burn. No vehicle loss occurs for roll failures. Figure 4-52 shows typical dynamics for loss of pitch attitude error in S-IC. Worst case loads for this case are shown in Figure 4-53.

4.9.3 EDS Effectiveness and Crew Safety

During S-IC flight up to 120 seconds, safe aborts are provided by the 4 degree per second overrate setting. Manual abort cues are:

a. 5 degrees attitude error
b. 3.2 psi q-ball pressure (between 50 and 120 seconds)
c. 10 degree per second overrate (after 120 seconds)

Abort cues for failures during S-II and S-IVB flight are:

a. ±20 degrees attitude deviation
b. FDO limits or ground confirmation of attitude deviation (abort request light)

False abort or crew loss is not expected for loss of attitude error signal malfunctions.

4.9.4 Mission Completion Capability

Loss of attitude error (pitch or yaw) during S-IC, S-II, or early S-IVB flight results in loss of mission. Parking orbit insertion is attained for loss of pitch or yaw error after 620 seconds. In S-IVB 2nd burn, full mission completion capability exists for loss of pitch or yaw error signal after approximately 310 seconds from reignition.

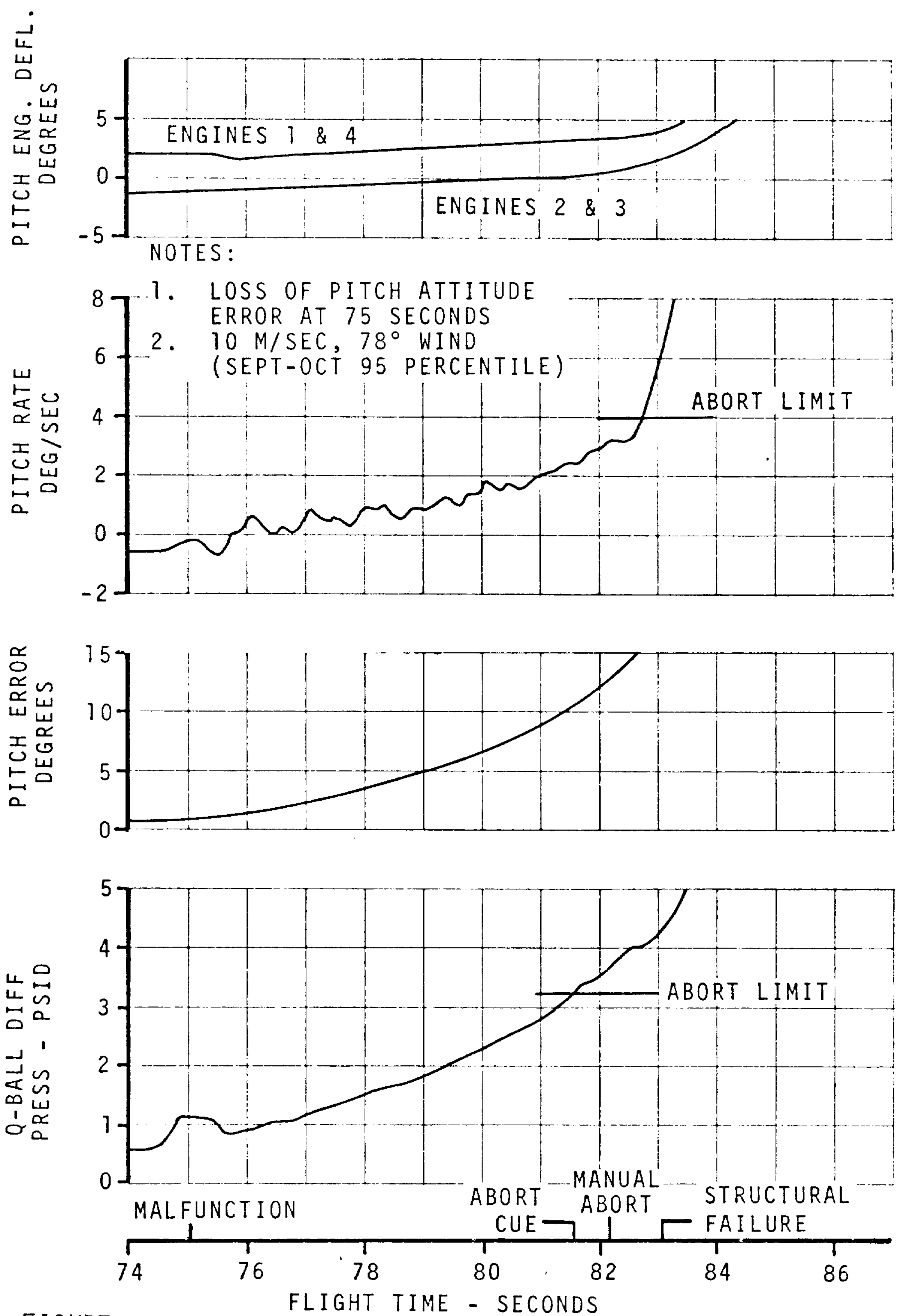


FIGURE 4-52 TIME HISTORY OF DYNAMIC VARIABLES FOR LOSS OF ATTITUDE ERROR (S-IC)

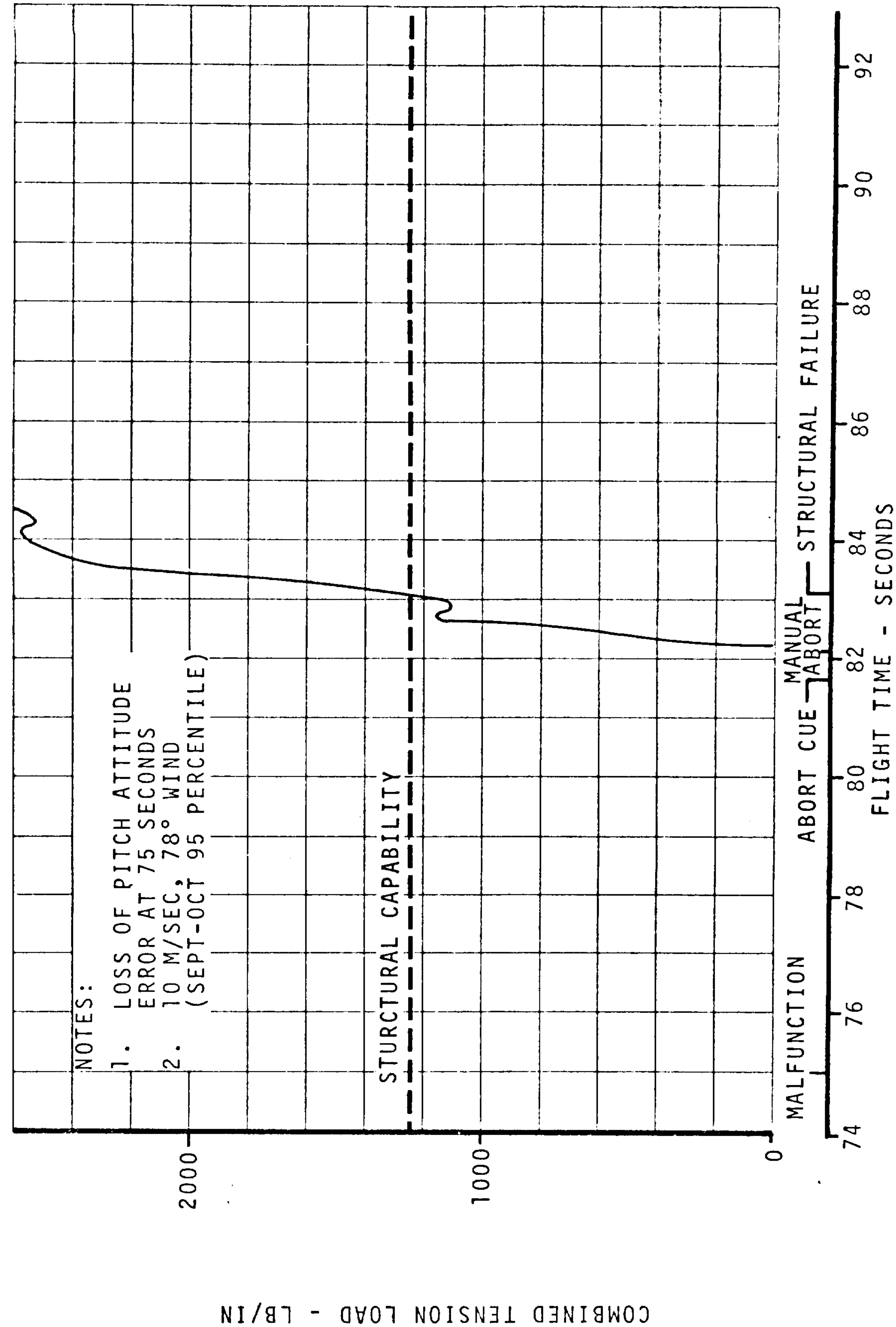


FIGURE 4-53 TENSION LOAD AT STA. 1564 (LOSS OF ATTITUDE ERROR)

4.10 LOSS OF ATTITUDE RATE SIGNAL

4.10.1 Malfunction Description

Any failure that causes a false vehicle rate indication of zero degrees per second to the flight control computer is referred to as a loss of attitude rate signal.

4.10.2 Malfunction Dynamics

Except for roll failures during S-IVB flight, the loss of attitude rate signal causes oscillatory divergence of vehicle attitude due to the absence of rate damping. During S-IC flight in the high-q region, this divergence is aggravated by the buildup of aerodynamic moments, eventually resulting in structural breakup. Dynamic variables for a loss of rate signal at 65 seconds are shown in Figure 4-54 . Loads that result from this malfunction are shown in Figure 4-55 . Figure 4-56 shows dynamics for loss of pitch rate signal during S-II flight.

Loss of roll rate signal in S-IVB produces only small attitude oscillations which do not diverge. This results from the different method of roll axis control used during S-IVB flight, namely, the Auxiliary Propulsion System (APS). The roll attitude of the vehicle is maintained within the 1 degree deadband of the APS control system as shown in Figure 4-57.

4.10.3 EDS Effectiveness and Crew Safety

During S-IC flight, safe automatic aborts are provided by the 4 degree per second overrate setting and rate indicator. After automatic abort is disabled, manual abort cues are 10 degrees per second overrate and 5 degrees attitude error.

Manual abort cues for failures during S-II and S-IVB flight are the 10 degree per second overrate light and spacecraft rate indicator.

False aborts or crew loss do not occur for loss of rate signal malfunctions.

4.10.4 Mission Completion Capability

Loss of attitude rate during S-IC and S-II stage flight results in vehicle and mission loss. For loss of pitch rate signal as early as 35 seconds before S-IVB engine cutoff, and loss of yaw rate signal as early as 100 seconds before S-IVB engine cutoff, insertion or injection may be achieved with residual rates less than 10 degrees per second.

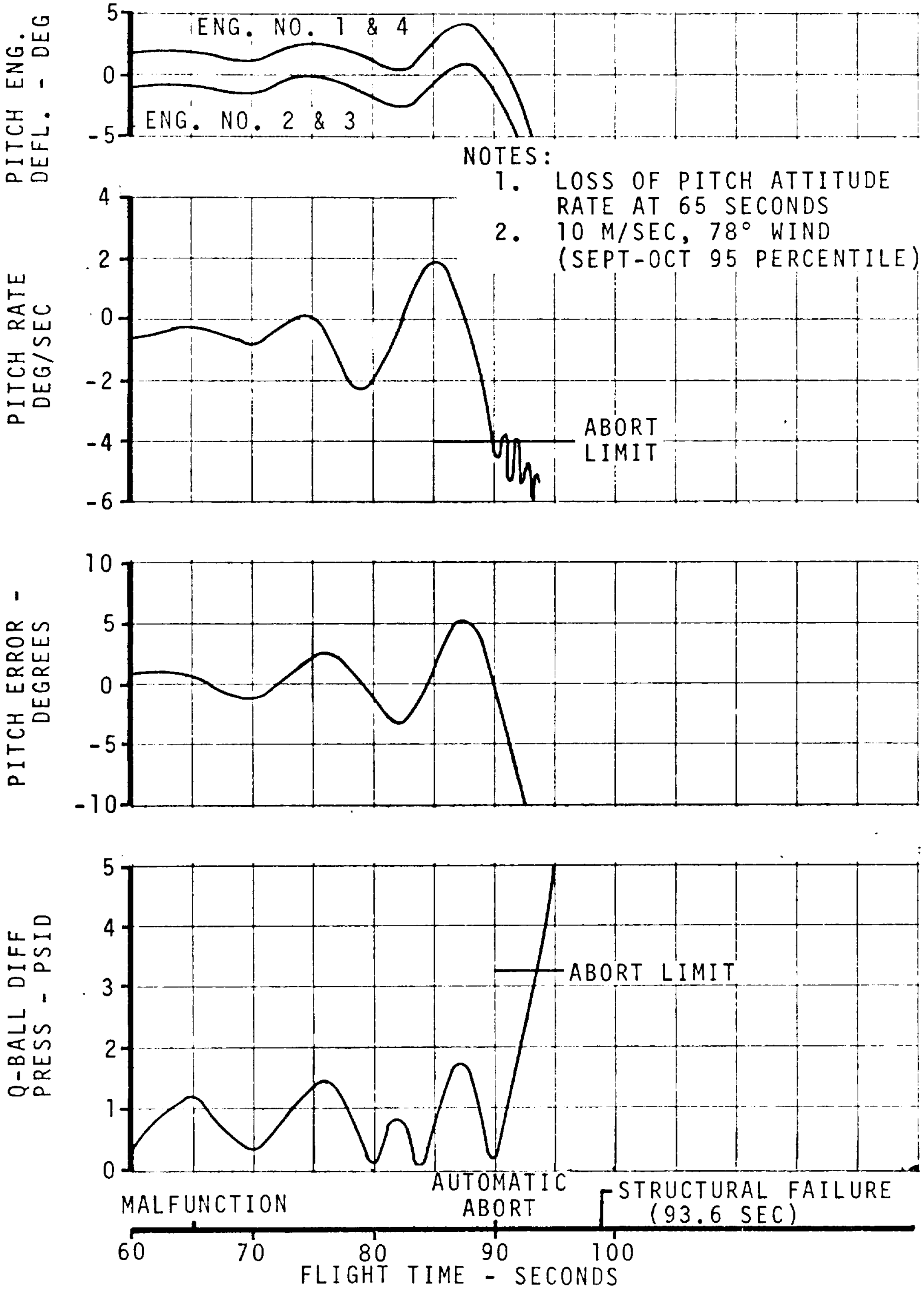


FIGURE 4-54 TIME HISTORY OF DYNAMIC VARIABLES FOR LOSS OF ATTITUDE RATE (S-IC)

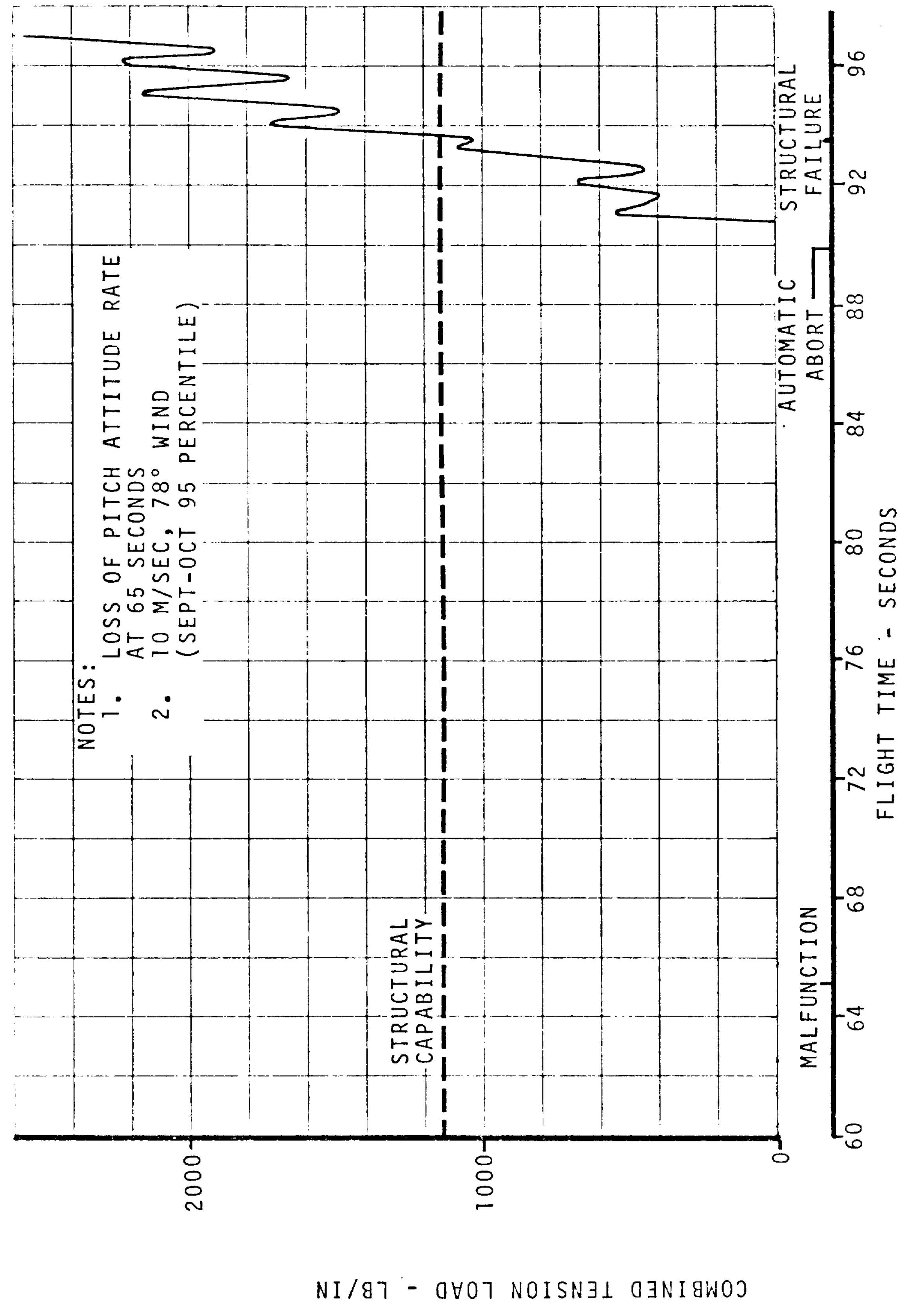


FIGURE 4-55 TENSION LOAD AT STA. 1564 (LOSS OF ATTITUDE RATE)

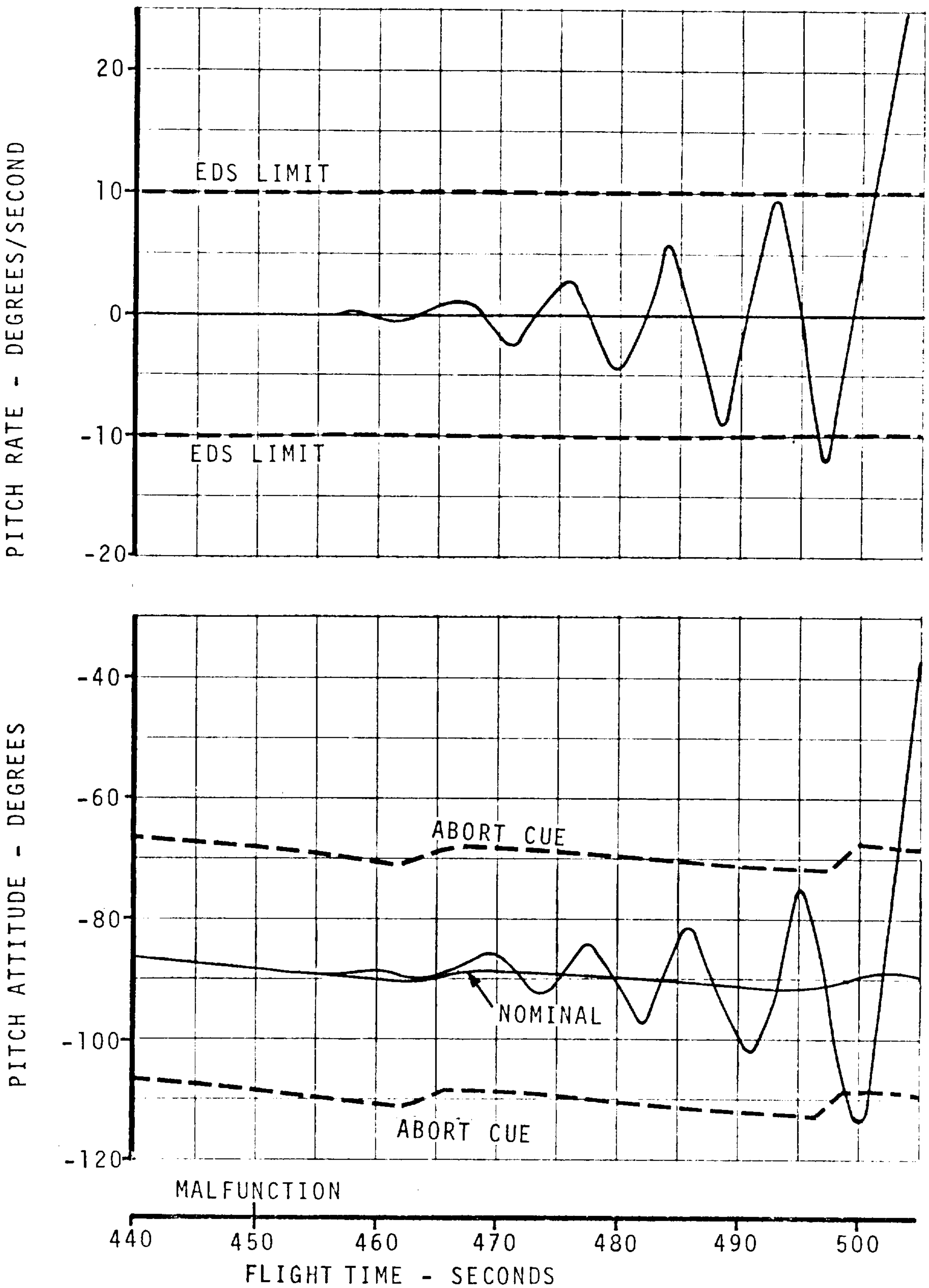


FIGURE 4-56 TIME HISTORY OF DYNAMIC VARIABLES FOR LOSS OF ATTITUDE RATE (S-II)

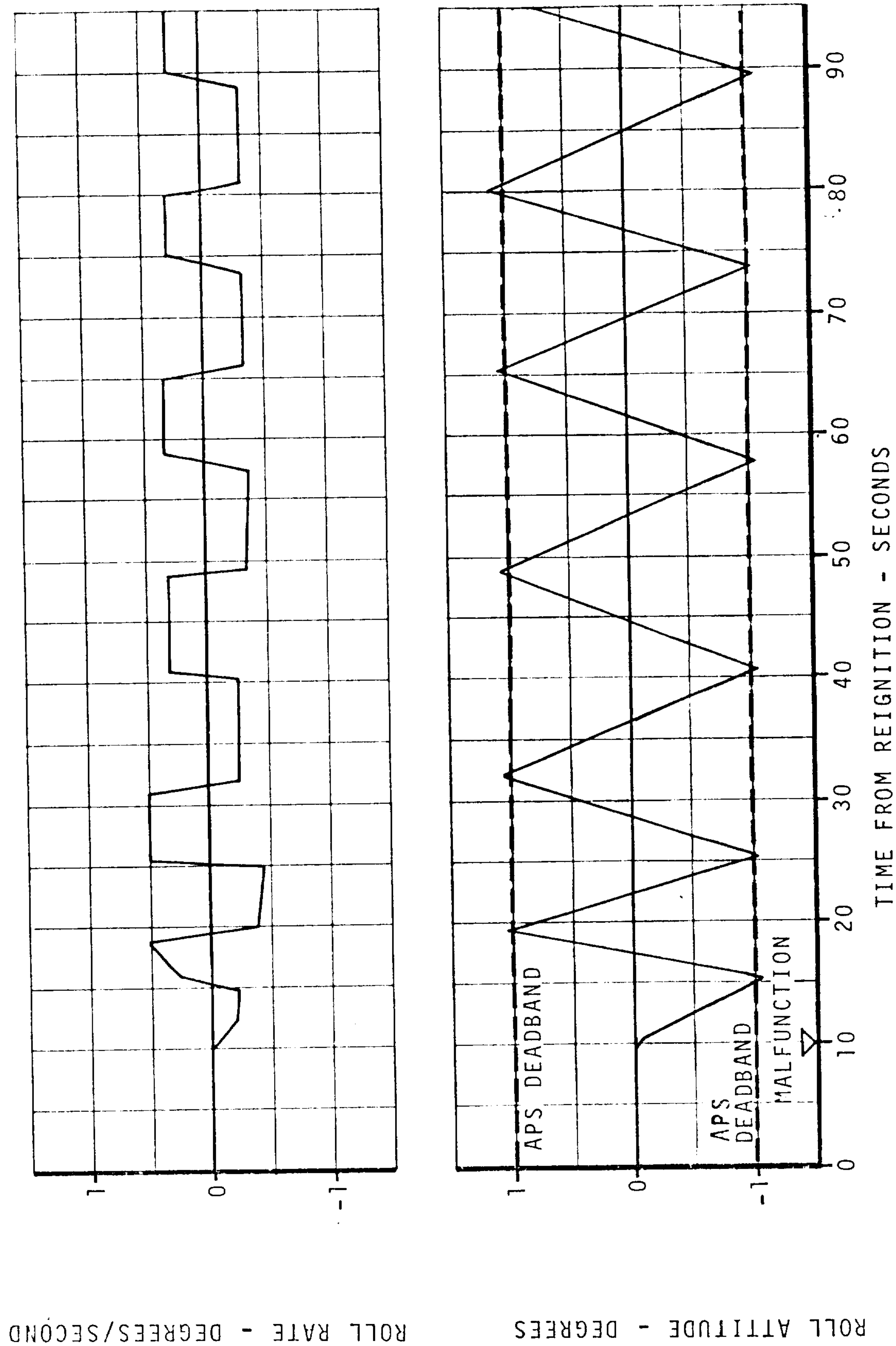


FIGURE 4-57 S-IVB ROLL DYNAMICS FOLLOWING LOSS OF ROLL RATE SIGNAL DURING SECOND BURN

4.11 ACCELEROMETER MALFUNCTIONS

4.11.1 Malfunction Description

Failures considered are those which result in a zero accelerometer reading on a single axis with and without thrust misalignment and the effects of manual S-IVB shutdown and navigator update.

The accelerometer output is tested for reasonableness in each major cycle. When the zero test is enabled (during each stage mainstage burn) an accelerometer output of less than $|0.05|$ meters/second is accepted when the vehicle longitudinal axis is within $\Delta\theta$ degrees of normal to that accelerometer input axis. $\Delta\theta$ is 2 degrees except during S-IC and S-II burn with a premature outboard engine shutdown, when it becomes 6 degrees. Backup data is substituted by the accelerometer processing logic when the accelerometer reading is rejected.

4.11.2 Malfunction Without Thrust Misalignment

Into-orbit X-accelerometer failures for a nominal vehicle result in perigee altitudes greater than 75 nautical miles. For $+3\sigma$ and -3σ vehicles (3σ acceleration profile), X-accelerometer failures result in perigee altitudes greater than 75 nautical miles after 135 and 30 seconds, respectively. The worst case is a -3σ X-accelerometer failure at 2.0 seconds which results in a perigee altitude of 53.5 nautical miles. Apogee/perigee altitudes for X-accelerometer failures are presented in Figure 4-58.

Into-orbit Z-accelerometer failures result in elliptical orbits with perigees near 100 nautical miles. Figure 4-59 shows apogee and perigee altitudes for nominal, $+3\sigma$, and -3σ Z-accelerometer failures.

Into-orbit Y-accelerometer failures result in nearly circular 100 nautical mile orbits. Yaw steering angle history for a Y-axis accelerometer failure at 2 seconds after first motion is presented in Figure 4-60 for boost to insertion and in Figure 4-61 for boost to TLI. For this failure, the earth orbit has a -0.1443 degree error in inclination and a -0.2177 degree error in descending node. The Y-axis velocity error at TLI is 20 meters/second.

The primary analysis for out-of-orbit accelerometer failures is for a September 13 (1969) launch date, a 78.051 degree launch azimuth, and a first opportunity. The velocity error in the failed axis (from desired cutoff hypersurface) at TLI for failure times of an X, Y, or Z-accelerometer is presented for the primary analysis in Figures 4-62 through 4-64. This error, rather than the perturbation from the nominal cutoff

4.11.2 (Continued)

velocity is presented because following an accelerometer failure, the hypersurface cutoff conditions differ from the nominal. The velocity components normal to the failed axis have no error from the targeted cutoff hypersurface other than normal accelerometer output error. Steering angle envelopes for all failure times and nominal histories for pitch angle and yaw angle are presented for the primary analysis in Figures 4-65 and 4-66, respectively.

The velocity error in the failed axis (see preceding paragraph) for first opportunity and across the launch window is shown in Figures 4-67 and 4-68 for X- and Z-accelerometer failures. Both maximum error and error as a function of failure time are obtainable from the figures for different days and azimuths. Velocity error for Y-accelerometer failures varies relative to how long IGM rides the 2° deadband and has a maximum predicted error of the order of 50 meters per second.

4.11.3 Malfunction With Thrust Misalignment

A thrust misalignment with no accelerometer failure has negligible effect on the flight because a calculation, Steering Misalignment Correction (SMC), is provided to correct the steering commands for the null input deflection error.

The insertion or injection error occurring with a pitch thrust misalignment concurrent with a pitch plane accelerometer failure is approximately equal to ($\pm$5 meters/second) the error for the accelerometer failure with no misalignment. The misalignment causes a slightly revised steering history resulting in a different boost duration compared with a similar accelerometer failure with no misalignment. This boost duration change ranges up to 2 seconds at POI, 3σ thrust misalignment in all three stages for into-orbit accelerometer failures, and 0.5 seconds at TLI, for S-IVB second burn failures.

The reason for these effects is that, while the misalignment causes the control system to set up a delta between the commanded (χ) and actual (θ) attitudes with which to zero out the misalignment, i.e.,

$$\beta = \beta_O + a(\Theta - \chi) + b\dot{\phi}$$

where:

a = altitude gain
b = rate gain
$\dot{\phi}$ = vehicle body rate
β = engine deflection
β_O = thrust misalignment,

4.11.3 (Continued)

the navigator sees the backup acceleration resolved through the actual attitude angles. The control system (with or without a misalignment) maintains the thrust through the vehicle CG. The angular error between this thrust acceleration and axial acceleration and also the magnitude error between the backup and actual acceleration is the same with or without a misalignment.

The insertion error with a yaw thrust misalignment concurrent with a Y-axis accelerometer failure depends on how long the 2° accelerometer output zero test limit is followed. A $\pm 3\sigma$ thrust misalignment causes up to 65 m/second additional error in Y at POI compared with a similar accelerometer failure with no misalignment and up to 0.5 second extension of the S-IVB first burn duration. The injection error with the same vehicle condition results in an additional error in Y at TLI of up to 5 m/second and up to 0.5 second extension to the S-IVB second burn. If a navigator update is used to correct an into-orbit Y error, then the into-and out-of-orbit Y errors are not additive. Note that the 2 degree zero test deadband will also affect certain day/azimuth S-IVB second burn X- and Z-accelerometer failures when the vehicle major axis lies within 2 degrees of the failed axis during second burn.

4.11.4 Manual S-IVB Shutdown

A manual shutdown on inertial velocity can reduce overspeed at POI and TLI. The manual cutoff at POI will only reduce the overspeed produced by an early Z-axis accelerometer failure. X-axis accelerometer failures have more of a flight path angle error than an overspeed and a manual cutoff does not reduce their POI error. Similarly, Y-axis failures have an orbit inclination error and little overspeed. A navigator update is required in EPO to correct disagreement between the actual and navigator state vectors.

At TLI with an LVDC cutoff, the overspeed is dependent on the S-IVB reignition angle (position vector in the X/Z plane) and the failure time (i.e., the component of the FOM-FOMC error in the failed axis). The overspeed reduction achieved by a manual shutdown on inertial velocity is dependent on the direction of the total acceleration immediately prior to TLI. The inertial velocity specified for the manual shutdown is obtained from a realtime S-IVB second burn simulation from the achieved EPO.

4.11.5 EDS Effectiveness and Crew Safety

Abort may be required for X-axis accelerometer failures prior to 30 seconds after first motion on a -3σ vehicle and 135

4.11.5 (Continued)

seconds after first motion on a $+3\sigma$ vehicle. The abort cues are FDO limits and abort request light. A Mode IV abort is required.

4.11.6 Mission Completion Capability

Mission completion capability for accelerometer failures is shown in Figure 1-9; mission continuance will be a real time decision. All out-of-orbit accelerometer failures result in perturbed TLI injection requiring midcourse correction and/or manual cutoff.

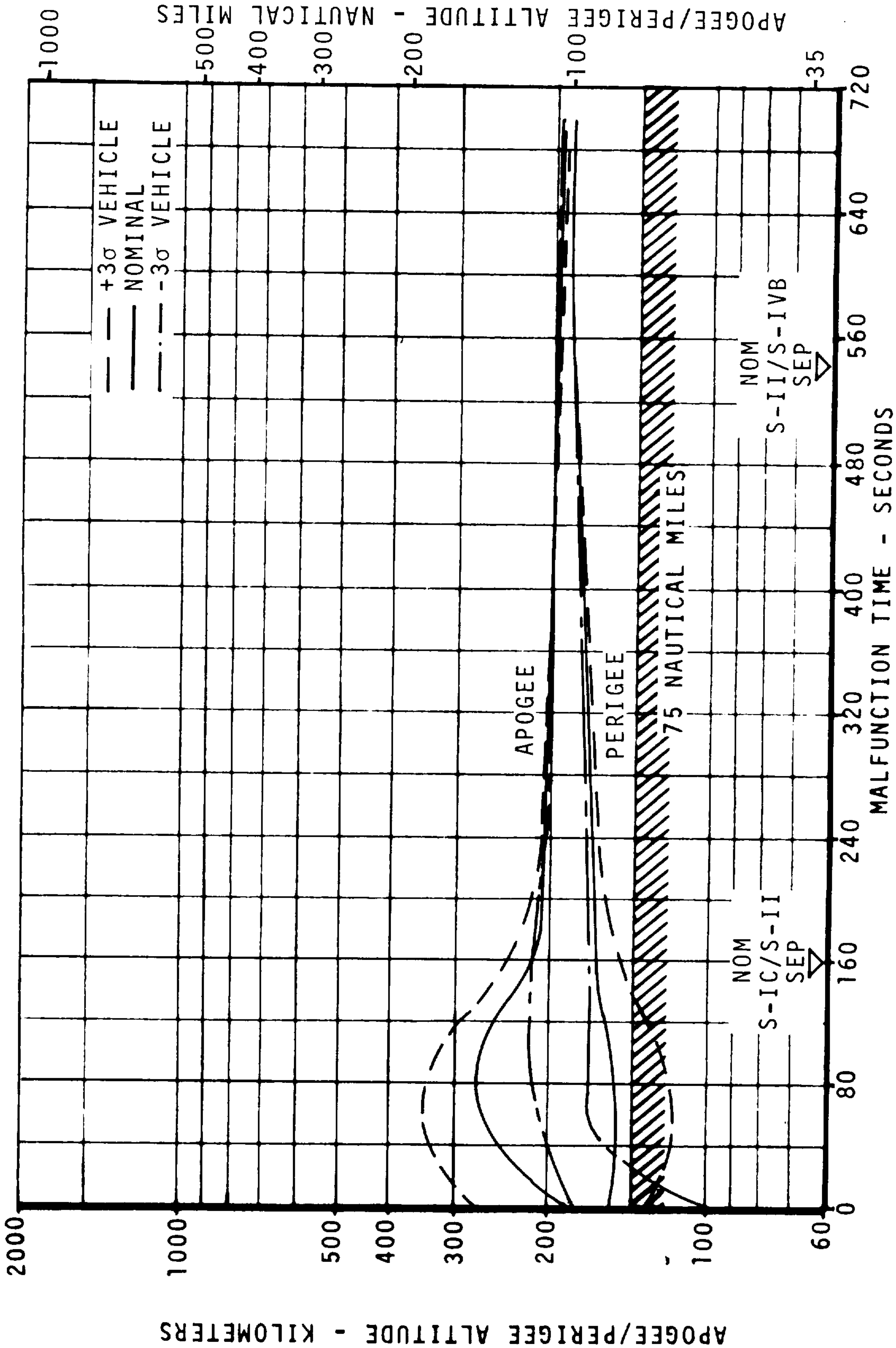


FIGURE 4-58 APOGEE/PERIGEE ALTITUDE FOR X-AXIS ACCELEROMETER FAILURES

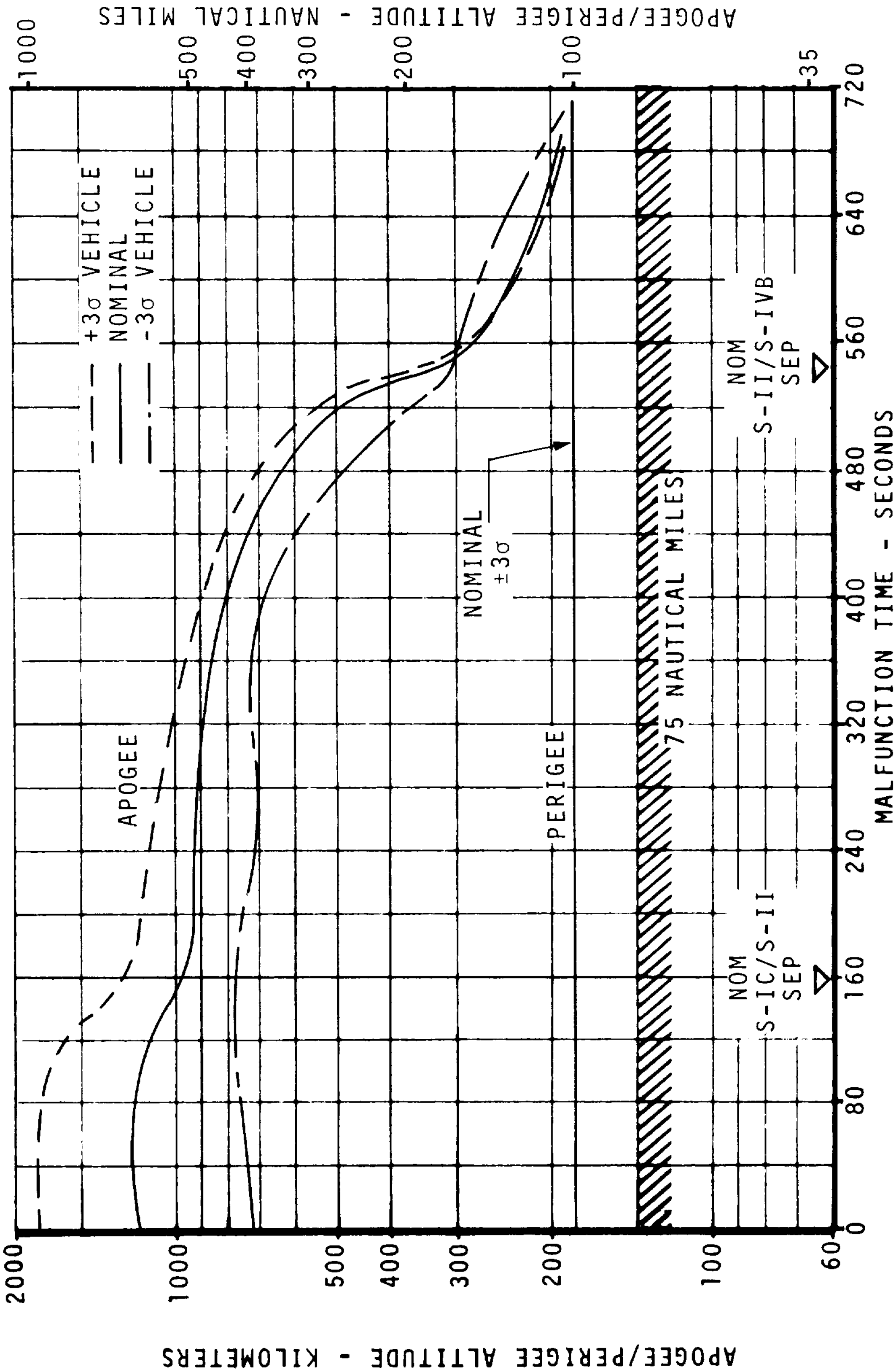


FIGURE 4-59 APOGEE/PERIGEE ALTITUDE FOR Z-AXIS ACCELEROMETER FAILURES

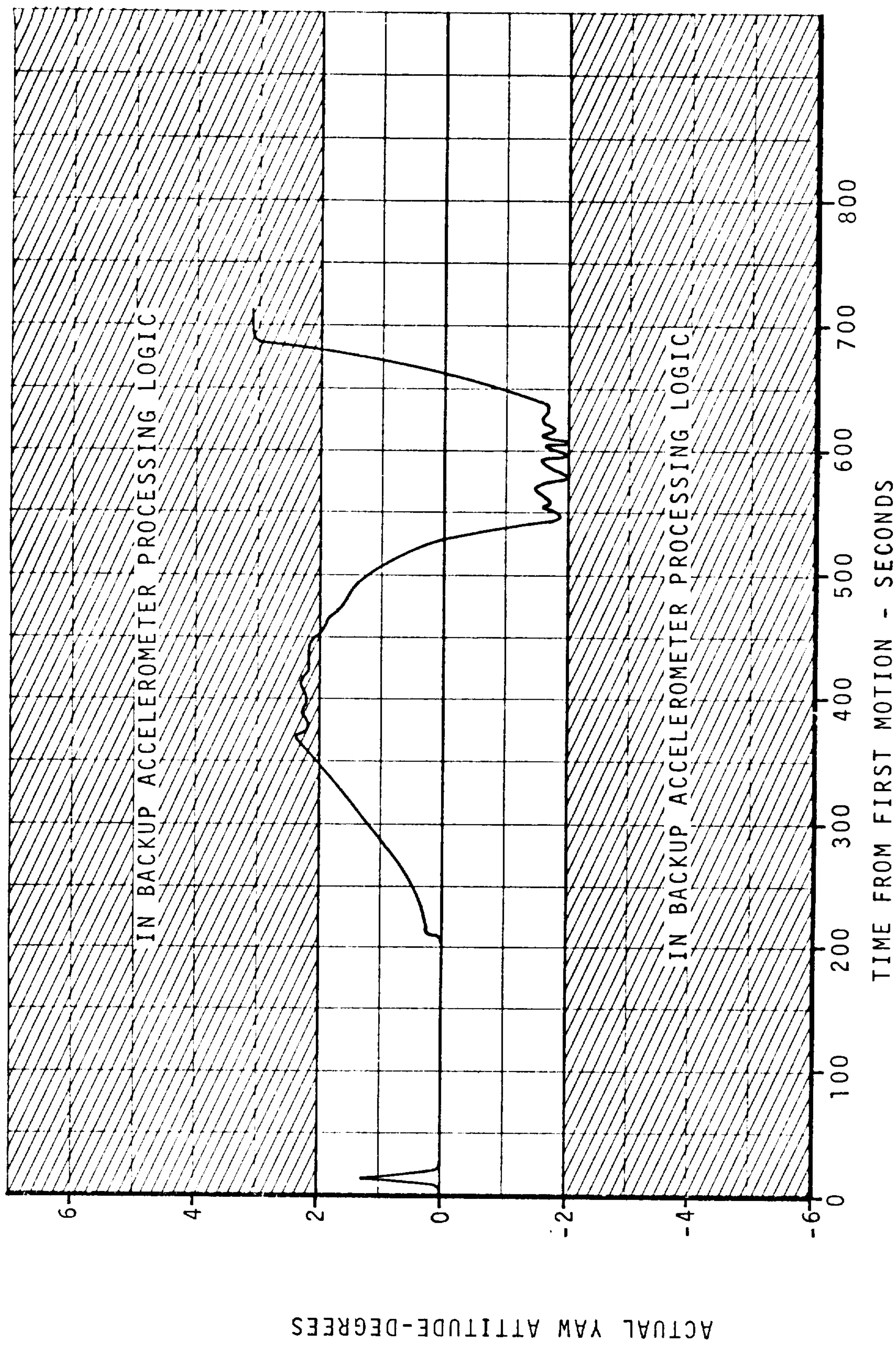


FIGURE 4-60 YAW ATTITUDE HISTORY FOR Y-AXIS ACCELEROMETER FAILURE
AT 2 SECONDS (BOOST TO INSERTION)

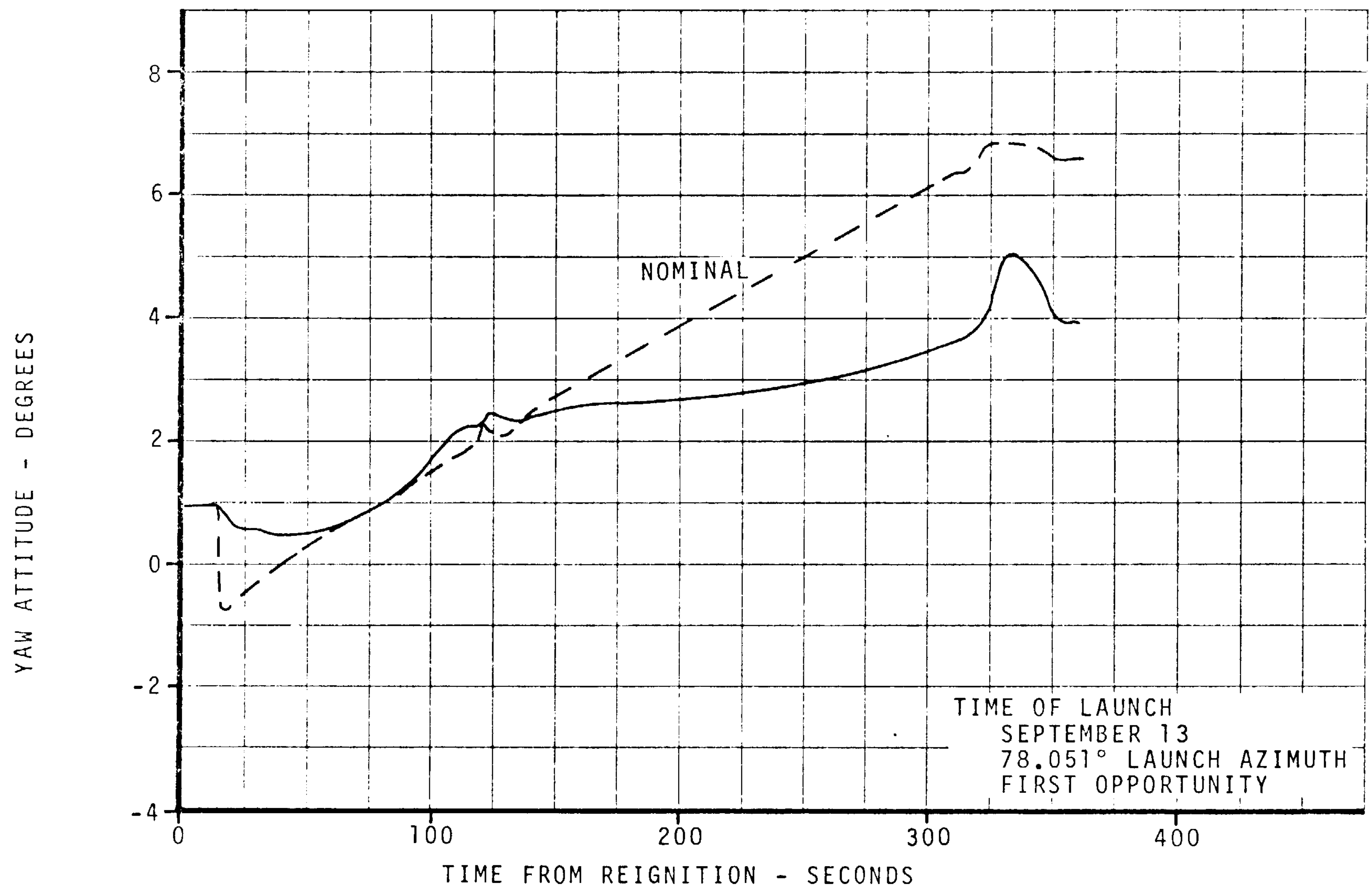


FIGURE 4-61 YAW ATTITUDE HISTORY FOR Y-AXIS ACCELEROMETER FAILURE AT 2 SECONDS FROM FIRST MOTION (BOOST TO TLI)

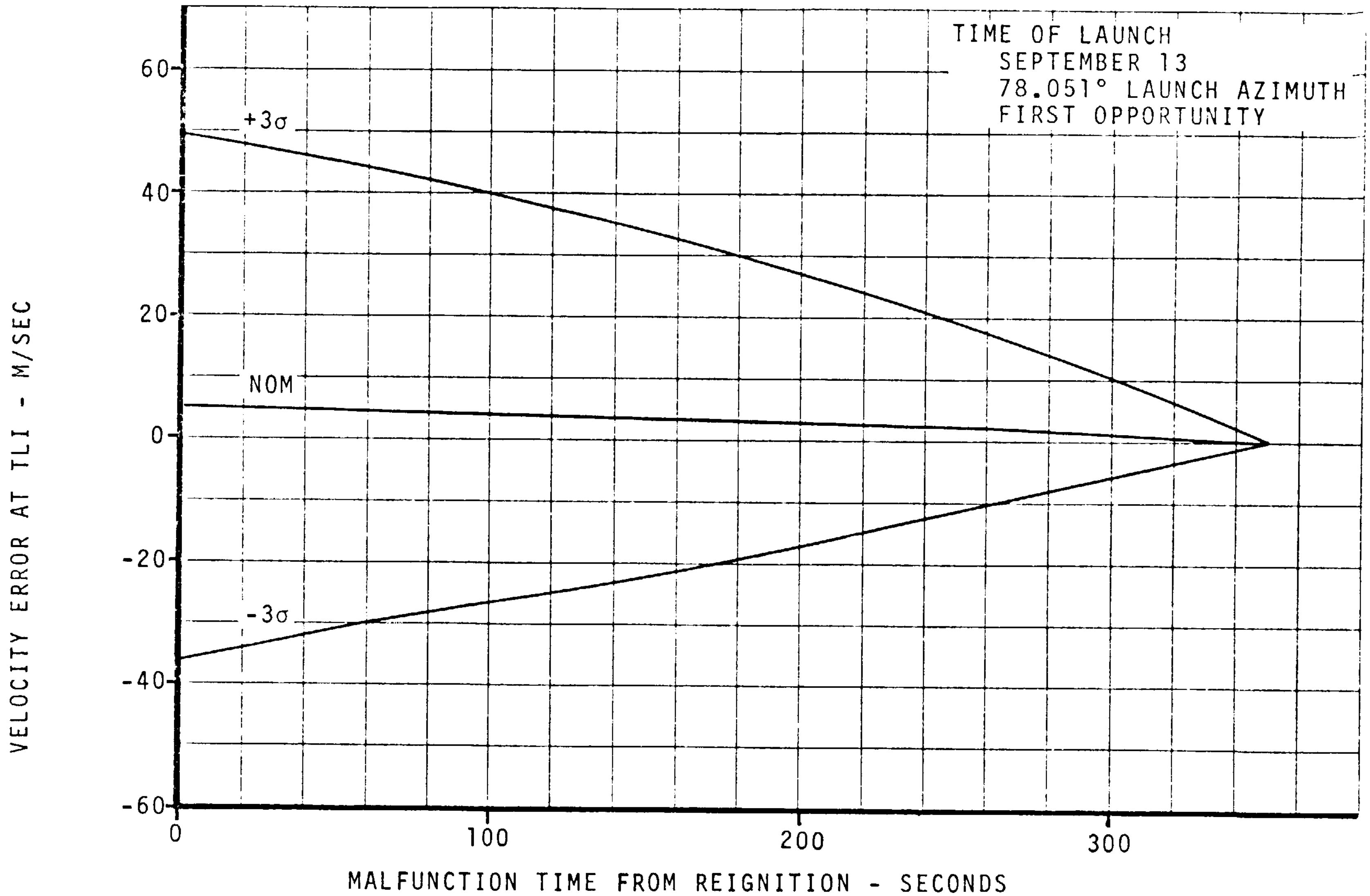


FIGURE 4-62 VELOCITY ERROR AT TLI IN FAILED AXIS FOR X-ACCELEROMETER FAILURES IN S-IVB SECOND BURN

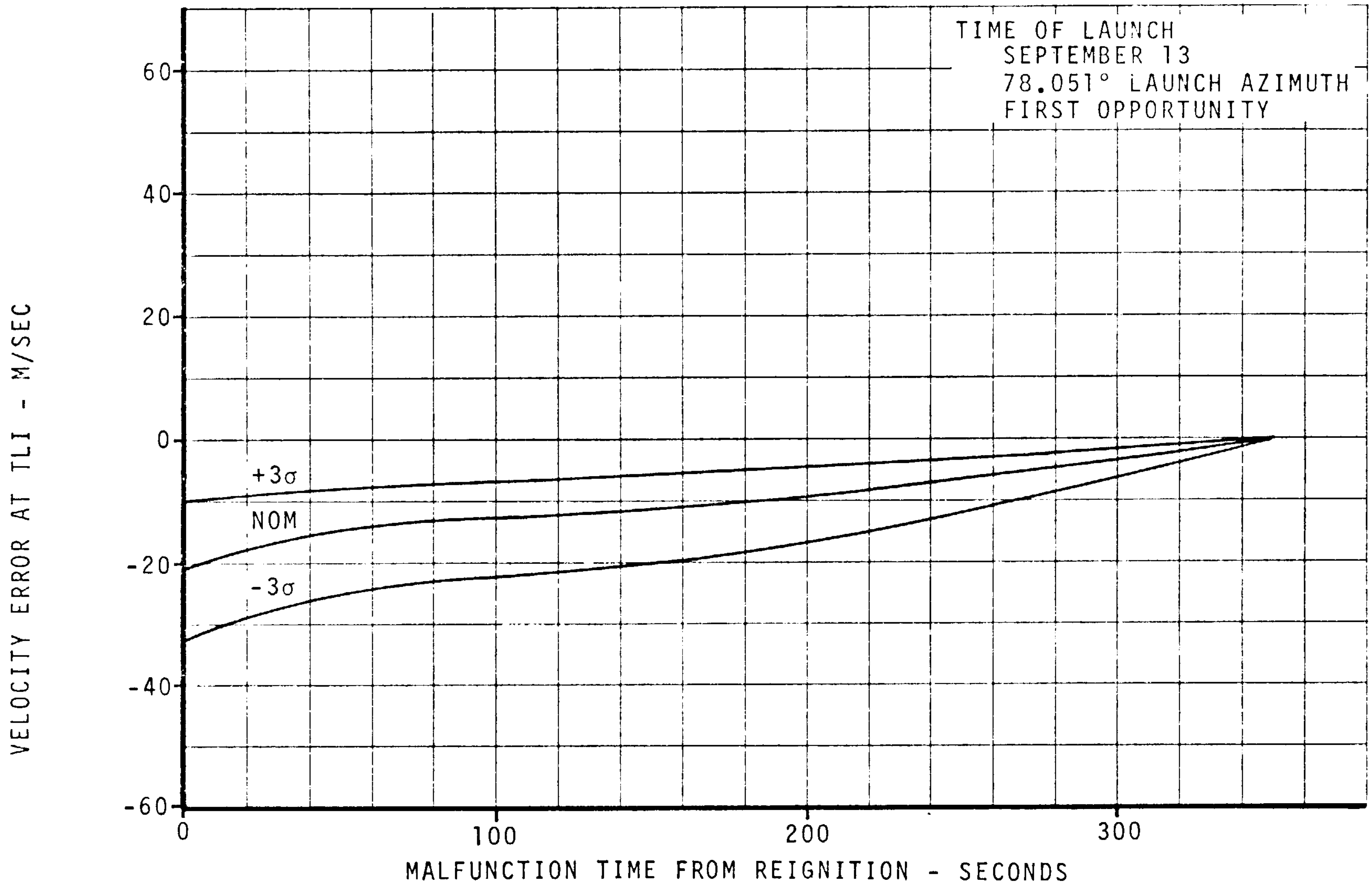


FIGURE 4-63 VELOCITY ERROR AT TLI IN FAILED AXIS FOR Y-ACCELEROMETER FAILURES IN S-IVB SECOND BURN

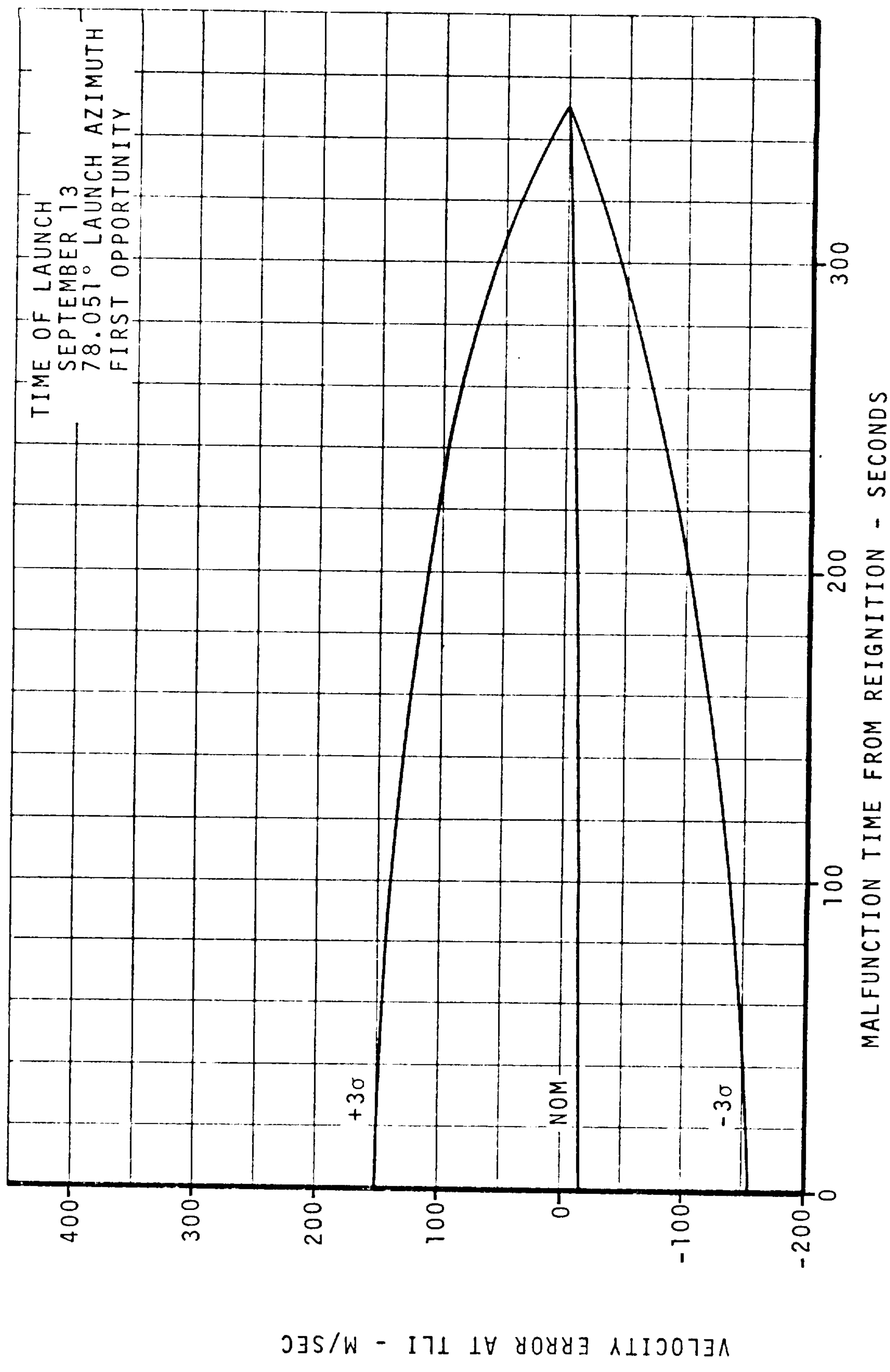


FIGURE 4-64 VELOCITY ERROR AT TLI IN FAILED AXIS FOR Z-ACCELEROMETER FAILURES IN S-IVB SECOND BURN

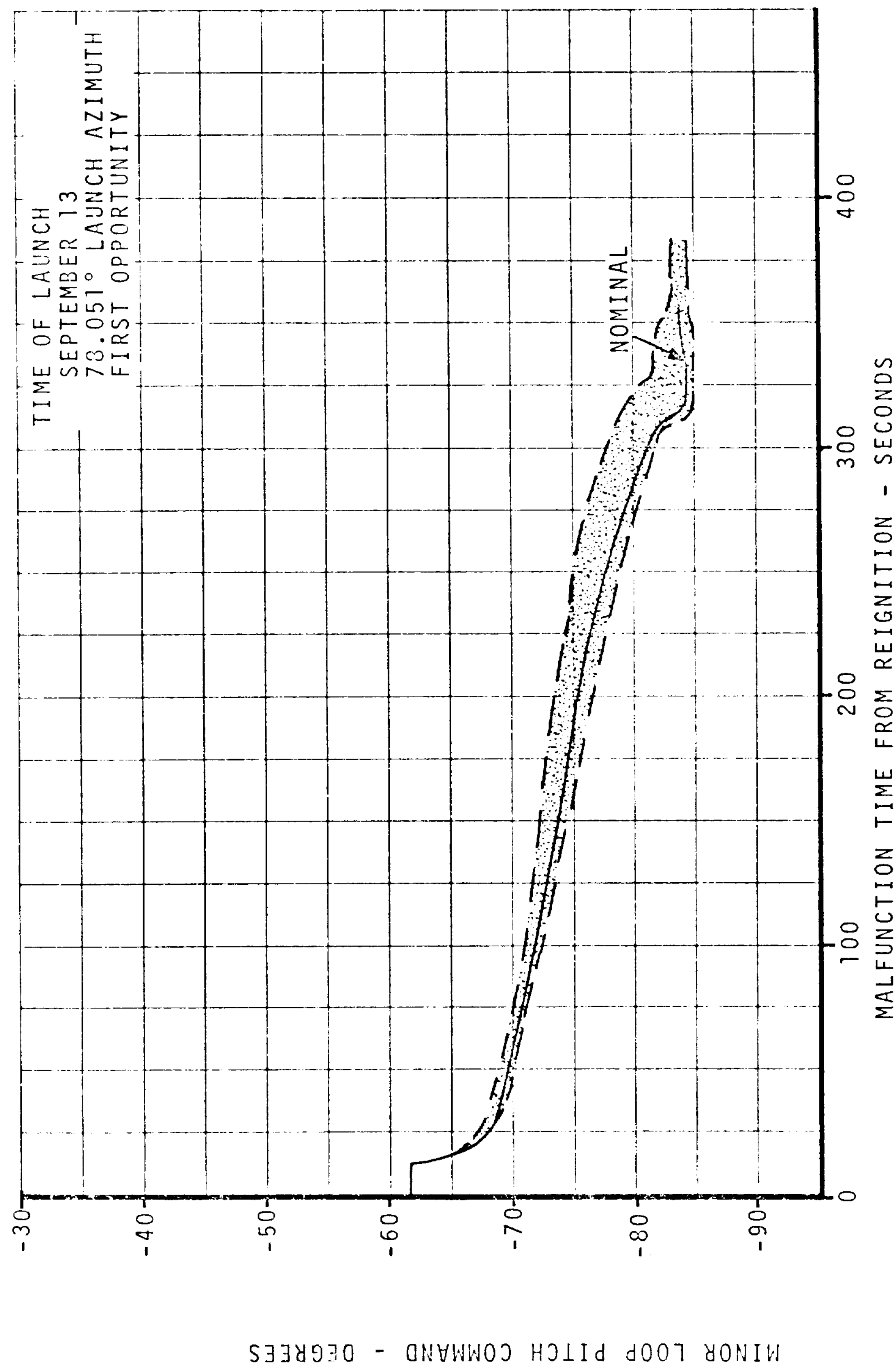


FIGURE 4-65 COMMANDED PITCH ATTITUDE ENVELOPE FOR ACCELEROMETER FAILURE DURING FIRST OPPORTUNTIY BURN

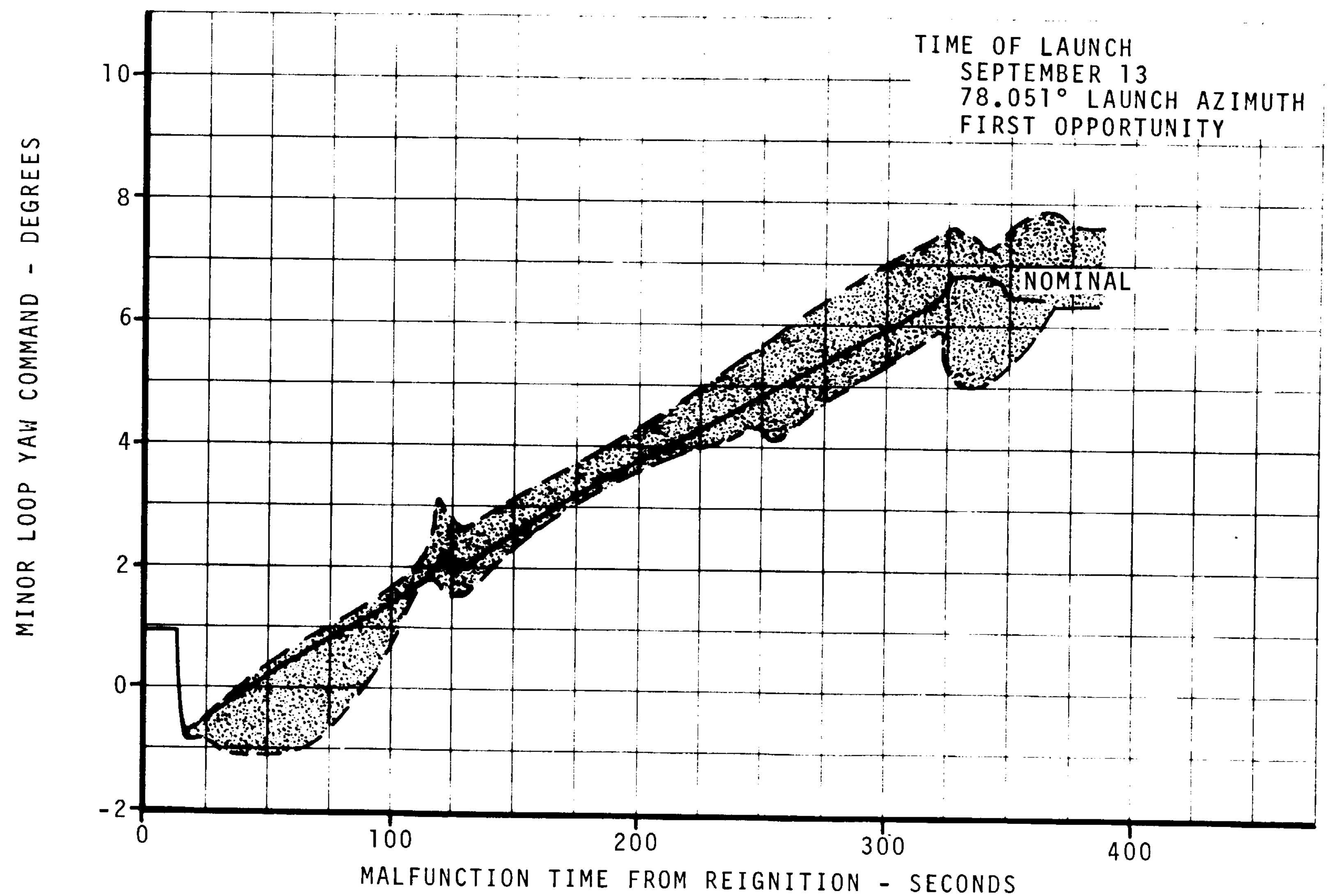


FIGURE 4-66 COMMANDED YAW ATTITUDE ENVELOPE FOR ACCELEROMETER FAILURES DURING FIRST OPPORTUNITY BURN

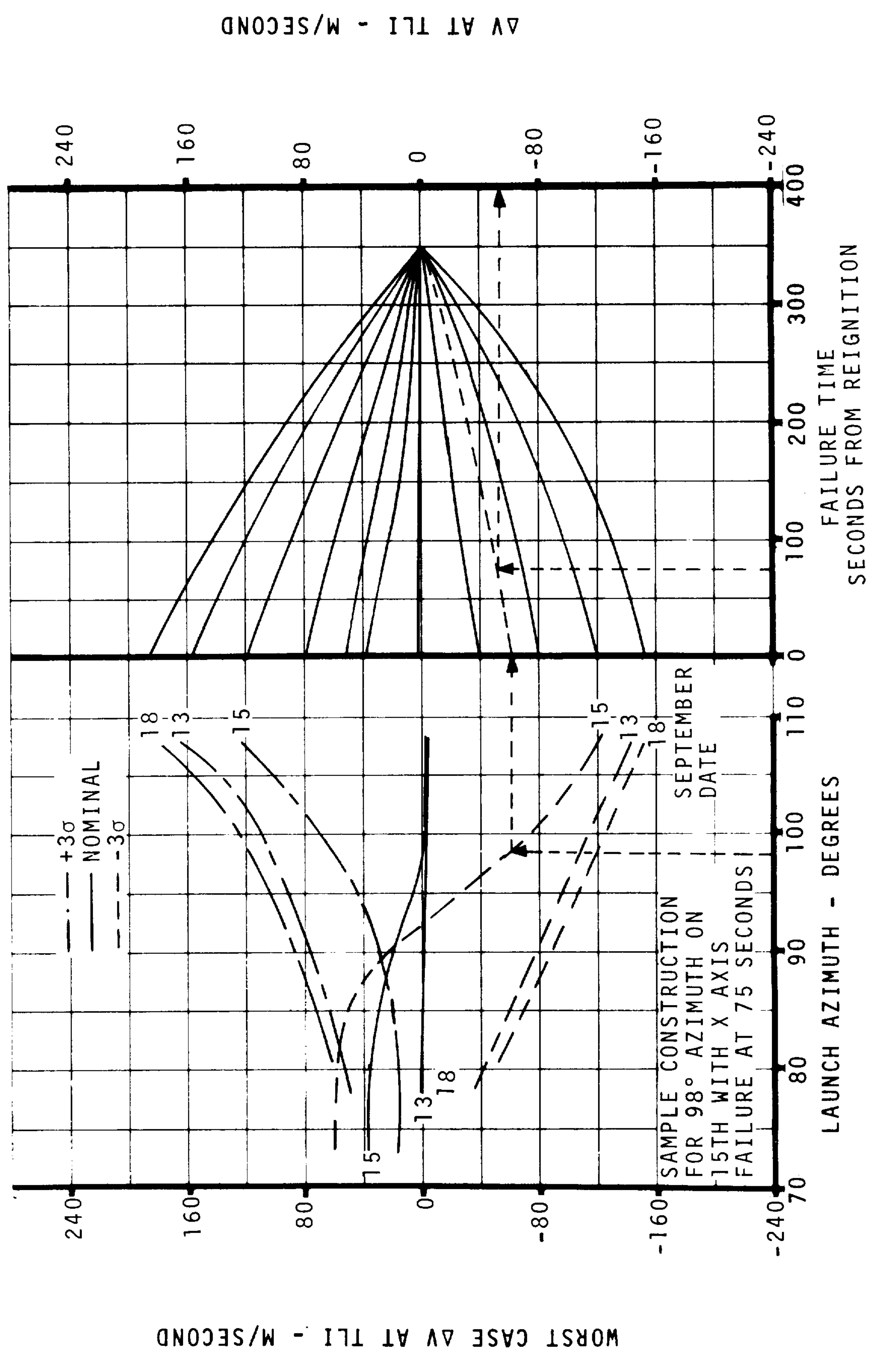


FIGURE 4-67 VELOCITY ERROR IN FAILED AXIS FOR X-ACCELEROMETER FAILURE FOR FIRST OPPORTUNITY

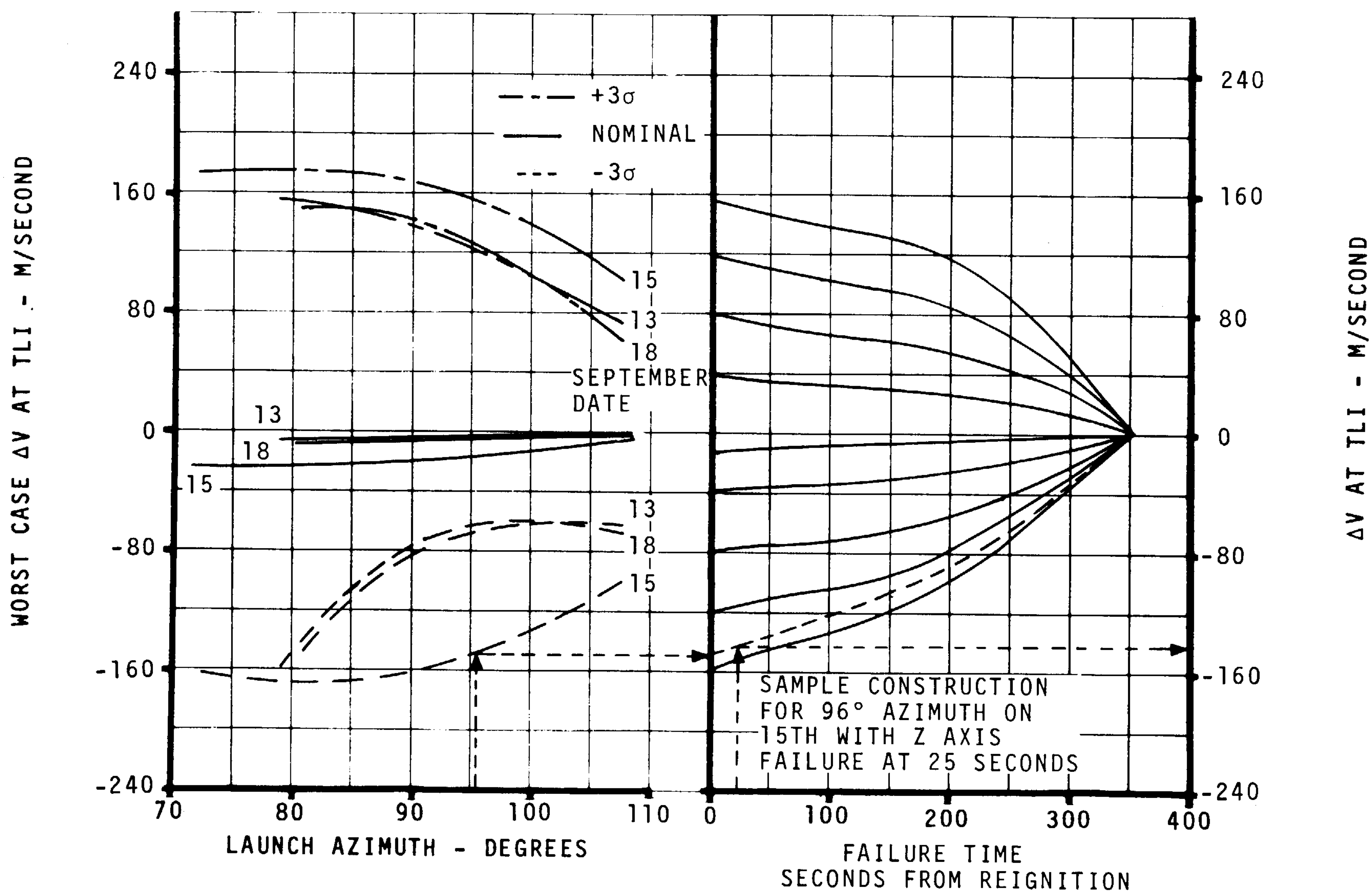


FIGURE 4-68 VELOCITY ERROR IN FAILED AXIS FOR Z-ACCELEROMETER FAILURE FOR FIRST OPPORTUNITY

4.12 PU SYSTEM MALFUNCTIONS

.12.1 Malfunction Description

The S-II and S-IVB Propellant Utilization (PU) System malfunctions are those in which the PU valve fails to follow the flight plan. No malfunctions are considered to occur before PU unlock. All valves fail simultaneously to some fixed position for remainder of flight.

4.12.2 Malfunction Dynamics

The primary effects of S-II PU malfunctions are trajectory deviations and perturbed S-IVB first burn times. Trajectory for PU failures to null and low stop exceed the nominal at S-IVB insertion into parking orbit. S-IVB PU malfunctions cause less deviation from the nominal trajectory than S-II PU malfunctions.

4.12.3 EDS Effectiveness and Crew Safety

Aborts are not required for PU malfunctions, and no false aborts are initiated.

4.12.4 Mission Completion Capability

Parking orbit insertion can be achieved for all PU malfunctions in boost-to-orbit. Translunar injection (TLI) cannot be achieved for PU malfunctions to null position or low stop occurring in the early portion of S-II flight. For a first opportunity reignition, PU malfunctions to the low stop in the later portion of the S-IVB first burn and all of the S-IVB second burn achieve TLI. For a second opportunity reignition, PU malfunctions to the low stop in S-IVB first burn and the early portion of S-IVB second burn will not achieve TLI. Failure to achieve TLI is due to propellant depletion. Table 4-I presents AS-505 preflight predictions applicable to AS-507. Failure times indicated are the earliest for which TLI may be achieved. "N/A" in the table indicates the PU valve is in its normal operating position. "All" in the table indicates vehicles with failures to the corresponding valve position have TLI capability for all failure times.

TABLE 4-I. TLI CAPABILITY FOR PU MALFUNCTIONS

STAGE	FAILURE TO NULL POSITION	FAILURE TO LOW STOP
S-II	340 Sec	475 Sec
S-IVB First Opportunity		
First Burn	N/A	370 Sec
Second Burn	All	All
S-IVB Second Opportunity		
First Burn	N/A	No TLI
Second Burn	N/A	14,610 Sec

4.13 LOSS OF ONE APS MODULE

4.13.1 Malfunction Description

Any nozzle firing failure in the APS system which causes lack of APS response to firing commands in the S-IVB stage is defined as APS System Failure. Other failures upstream of the control computer output are not considered here.

Loss of one APS module is defined as the failure of all thrusters in one APS module to fire when commanded to do so.

APS ullaging failures are defined as malfunctions resulting in failure to turn on an APS ullage thruster. Only single failures are considered.

4.13.2 Malfunction Dynamics

Loss of one APS module in S-IVB first or second burn will not result in abort conditions; pitch and yaw control are maintained by the main engine and the remaining module provides sufficient roll torque to correct for small perturbations.

Loss of one APS module during coast flight is cause for abort, since control about the pitch axis exists in only one direction. This malfunction during S-IVB coast will also cause a gradual roll/yaw attitude divergence.

Loss of one APS ullage motor does not cause loss of control.

4.13.3 EDS Effectiveness and Crew Safety

Loss of one APS module during parking orbit produces loss of control in the pitch axis. Since attitude divergence is slow, APS failure may not be detected immediately. The abort cues are ± 20 degrees attitude deviation and ground requested abort. All aborts are safe aborts.

False abort or crew loss does not occur for loss of one APS module.

4.13.4 Mission Completion Capability

While loss of a single APS module during S-IVB first burn allows a nominal POI, an abort during parking orbit is required. Abort is required if this malfunction occurs during parking orbit. Loss of one APS module during S-IVB second burn does not prevent TLI; subsequent to TLI, however, no pitch control is available. This lack of control results in uncorrected vehicle body rates.

4.14 LOSS OF BOTH APS MODULES

4.14.1 Malfunction Description

"Loss" is defined as in Paragraph 4.13.1, except both APS modules are affected.

4.14.2 Malfunction Dynamics

Loss of both APS modules in S-IVB powered flight produces loss of roll control.

Loss of both APS modules during S-IVB parking orbit coast causes total loss of attitude control and requires an abort.

Although TLI is attained for loss of both APS modules during S-IVB second burn, rates continue uncorrected, presenting difficulties for transposition, docking, and extraction (TD&E).

4.14.3 EDS Effectiveness and Crew Safety

Loss of both APS modules during S-IVB powered flight results in loss of roll control. Excessive roll attitude is used as a cue for manual abort.

For both APS modules out during coast flight, the S-IVB is uncontrollable, but divergence is slow. Since body rates are small, this malfunction may not be detected immediately. Abort should be initiated before platform gimbal limits are exceeded. Abort cues for loss of both APS modules are excessive attitude deviation and abort request light.

False abort or crew loss is not expected for loss of both APS modules.

4.14.4 Mission Completion Capability

Loss of both APS modules during S-IVB first burn or parking orbit coast results in mission loss. During S-IVB second burn, this malfunction will not prevent TLI; but the S-IVB will be uncontrollable after TLI, and capability to perform the TD&E is questionable.

4.1ʳ SEQUENCING AND STAGING MALFUNCTIONS

1.15.1 Malfunction Description

A sequencing malfunction is defined as any malfunction that causes an erroneous sequence or no sequence to occur, or a sequence that occurs at the wrong time. Sequencing failures studied are:

Early gain switching
b. Late (or no) gain switching.

A staging malfunction is defined as any malfunction that causes premature staging, lack of staging, or complications during staging. Those studied include:

a. Lack of retrorocket firing
b. Lack of (or partial) S-IC first plane separation
c. Lack of (or partial) S-II second plane separation
d. Lack of (or partial) S-II/S-IVB plane separation
e. Failure to jettison the Launch Escape Tower (LET)

4.15.2 Malfunction Dynamics

An early control system gain change (prior to 85 seconds) results in vehicle loss because the vehicle is unstable in the max-q region with low gains.

A late gain change does not require abort even if no gain change occurs during S-IC flight. The vehicle becomes unstable due to slosh, but more than 50 seconds of instability are required to cause loss of control.

Failure to achieve S-IC/S-II staging requires abort since parking orbit insertion cannot be attained by staging directly to the S-IVB.

One retrorocket out at S-IC/S-II or S-II/S-IVB staging does not require abort. Failure to issue firing commands to the S-IC or S-II retrorockets produces delayed stage separation and a possible period of local deformation where the stages remain in contact after they have been severed. Thrust tailoff forces of the engines tend to keep the stages in contact; precise structural consequences are not known.

A sequencing failure affecting time of gain change does not cause an abort during S-II flight.

4.15.2 (Continued)

Sequencing failures or separation device malfunctions which prevent jettison of the S-IC/S-II interstage cause the thermal environment limits in the S-II boattail area to be exceeded. Engine shutdown and abort are required prior to generation of excessive temperatures. Due to the short time between nominal interstage jettison and exceeding termperature limits, the mission rule presented in Reference 6 (abort prior to TB3 + 52 seconds) is recommended.

Failure to jettison the launch escape tower introduces no stability problem during S-II and S-IVB flight. The launch escape tower and boost protective cover prevent spacecraft/ lunar module transposition and docking. Reentry is not possible since the drogue and main parachutes cannot be deployed. Procedures are required to allow the crew to remove the LET and boost protective cover during earth parking orbit.

Failure to achieve S-II/S-IVB staging due to a failure to issue the sequence command requires abort.

An early gain change or absence of one during S-IVB second burn does not require abort.

4.15.3 EDS Effectiveness and Crew Safety

Table 4-IIcontains the abort logic and limits for all sequencing and staging malfunctions considered in this analysis. All abort lead times are adequate for crew safety.

False abort or crew loss does not occur for sequencing and staging malfunctions.

4.15.4 Mission Completion Capability

Mission completion capability for sequencing and staging malfunctions is given in Column 4 of Table 4-I.

TABLE 4-II ABORT LOGIC AND MISSION CAPABILITY FOR SEQUENCING AND STAGING MALFUNCTIONS

Sequencing or staging malfunctions	ABORT LOGIC		Mission completion capability
	Abort Cues	Procedure	
Lack of S-IC/S-II first plane separation	1. S-II engine out lights 2. Loss of acceleration	Abort by LET (orbit not possible)	Complete loss of mission
S-IC early gain change	1. 5° attitude error 2. $\Delta P\alpha$ of 3.2 psi	Abort by LET	Complete loss of mission
S-IC late gain change	NO ABORT REQUIRED		Full mission completion capability
Lack of S-II second plane separation	Second plane separation indicator (redundant indication)	Abort by LET prior to TB3 + 52 seconds (orbit not possible)	Complete loss of mission
Failure to jettison LET	1. Visual by crew 2. Tower off indication (ground)	Immediate abort not recommended	If removal of LET is achieved in orbit, full mission completion capability exists
S-II early or late gain change	NO ABORT REQUIRED		Full mission completion capability
Lack of S-II/S-IVB plane separation	1. S-IVB engine out light 2. Loss of acceleration	Abort by SPS (orbit not possible)	Complete loss of mission
S-IVB early or late gain change	NO ABORT REQUIRED		Full mission completion capability

4.16 S-II/S-IVB EARLY STAGING

4.16.1 Description

Manual early staging to alternate timebase 4 sequence can be initiated between thrust OK arm, 6.7 seconds after timebase 3, and propellant depletion cutoff arm, 334.2 seconds after timebase 3.

4.16.2 Limits

The largest body rate from which early staging can be initiated and not fail the fine gimbal resolver test is approximately 2 degrees/second. The rate from which successful S-IVB recovery can be anticipated after early staging should not exceed 3 degrees/second.

4.16.3 Mission Completion Capability

S-II/S-IVB early staging capability is shown in Figure 1-8. S-IVB burn duration as a function of the time of early staging is presented in Figure 4-69.

4.16.4 African Continent Impacts

Table 4-III details the intervals of S-II/S-IVB early staging times for a nominal vehicle which result in African continent impact. These intervals are projected from AS-507 data and AS-505 results.

TABLE 4-III S-II/S-IVB EARLY STAGING CONDITIONS FOR AFRICAN CONTINENT IMPACTS

LAND IMPACT AREA	STAGING TIME† SECONDS	GEOCENTRIC RADIUS METERS	INERTIAL VELOCITY M/SEC	INERTIAL FLIGHT PATH ANGLE DEGREES
72° AZIMUTH WEST COAST (CANARY ISLANDS)	330	6545526.	3995.	3.983
EAST COAST	375*	6555080.	4516.	1.936
90° AZIMUTH WEST COAST (CAPE VERDE ISLANDS)	330	6546123.	4014.	3.973
EAST COAST	375*	6555673.	4536.	1.918
108° AZIMUTH WEST COAST	365	6553990.	4392.	2.303
EAST COAST	375*	6555628.	4516.	1.922

†NOTE: These times are given to the nearest 5 seconds.
*Achieves parking orbit.

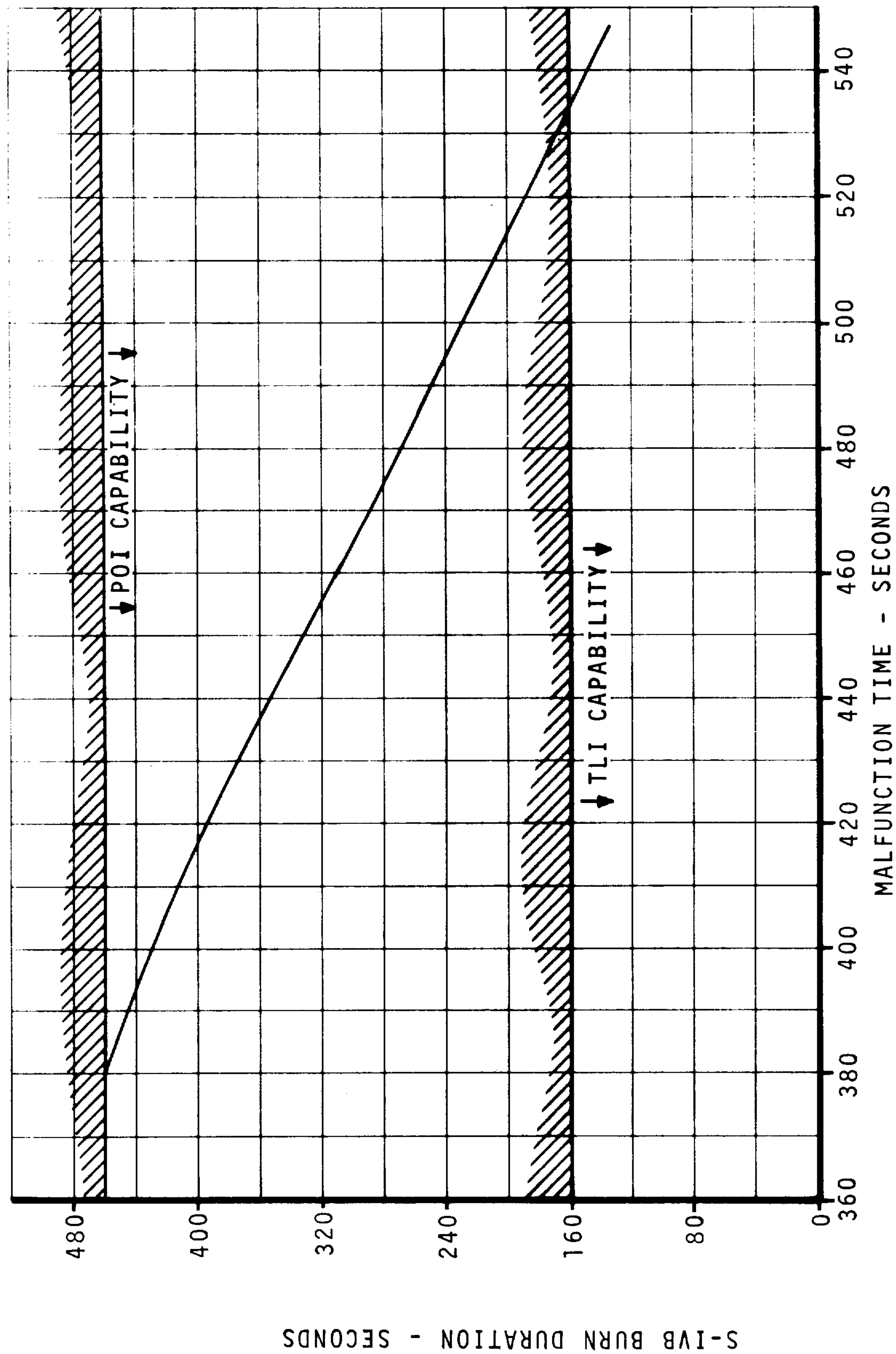


FIGURE 4-69 S-IVB BURN DURATION VS TIME OF S-II/S-IVB EARLY STAGING

SECTION 5

COMMUNICATIONS ANALYSIS FOR ABORT AND ALTERNATE MISSIONS

5.1 COMMUNICATIONS ANALYSIS BACKGROUND

The objective of this study is to provide an assessment of the tracking and communications capability, for a defined surveillance network (Table 5-I), for abort and alternate missions within the capability of the AS-507 vehicle. This study complements analyses presented in Boeing Document D5-15697-507, "Tracking and Telemetry Design Analysis for AS-507 (G Mission)," dated June 20, 1969. Abort and alternate mission trajectories, resulting from vehicle system or subsystem malfunctions, result in deviations from the nominal communications capability such that additional data are required for contingency planning.

In the referenced document, an extensive analysis of the tracking and communications capability is presented for the nominal AS-507 G Mission based upon the AS-507 Launch Vehicle Operational Flight Trajectory. A summary of the nominal 0-degree elevation angle surveillance for the Earth Parking Orbit is given in Table 5-II for the 78.051-degree launch azimuth on September 13, 1969.

The nominal tracking and communications analysis for the launch and parking-orbit flight phases is based upon the AS-506 Launch Vehicle Operational Flight Trajectory. The AS-507 Launch Vehicle Operational Trajectory during those flight phases does not show any appreciable differences, in regards to geometrical surveillance, from AS-506 trajectory data. Computer runs were made using the AS-507 Operational Trajectory to verify the applicability of the AS-506 surveillance data. The alternate mission analysis is based on the AS-507 launch vehicle.

An abort mission, for purposes of this analysis, is a mission that results when any vehicle system or subsystem failure prevents the launch vehicle from achieving insertion into a parking orbit; therefore, the communications analysis of the abort missions is restricted to the launch and boost-to-parking-orbit phase of flight.

An alternate mission is, for purposes of this analysis, any mission resulting from a system or subsystem failure that permits insertion of the launch vehicle into an earth orbit. Alternate missions are analyzed during both the boost and earth parking orbit flight phases to develop surveillance data for use in an overall assessment of the tracking and communications capability for the AS-507 G Mission.

5.1 (Continued)

A detailed tracking and communications analysis for all of the possible abort and alternate missions to the detail provided by the referenced AS-507 G Mission nominal surveillance analysis is impractical. To satisfy the need for displaying meaningful data for the various possible abort and alternate missions, analyses are conducted to identify fundamental tracking properties applicable to the various flight phases.

A special surveillance analysis is included in this section that provides surveillance for both the instance of no S-IVB stage restart ignition (for variable azimuth across the launch window) and for the instance of an early stage cutoff for September 13, 1969, second TLI opportunity and 78.051-degree launch azimuth.

No allowances are made in this study for the effects that result from a degradation in antenna look-angles. Any degradation in antenna look-angles may result in a change in the capability to communicate with the launch vehicle.

5.2 ANALYTICAL PROCEDURES AND STUDY LIMITATIONS

During the launch and boost-to-parking-orbit flight phase, continuous tracking and communications surveillance is available for all defined alternate mission trajectories. Because of the diversity of the altitude/surface range profiles for abort trajectories, each abort condition must be considered as an individual communication problem; therefore, specific abort cases are not presented in this analysis.

Any trajectory with an altitude/surface range profile above the surface of surveillance defined by the minimum altitude/ surface range contours shown in Figure 5-1 is under surveillance continuously by one or more Eastern Test Range stations. This surface of surveillance is generated by joining the points of intersection of the adjacent tracking and communications station visibility cones of the MILA, Grand Bahama, Grand Turk, Antigua, and Bermuda ground stations. Limited surveillance is available below this surface, but continuous surveillance is unavailable.

The AS-507 G Mission nominal altitude/surface range profile and the minimum altitude/surface range profile for an alternate mission, resulting from an S-IC stage dual-engine shutdown at 90 seconds after liftoff, are presented in Figure 5-1 to indicate nominal and typical worst-case malfunction flight profiles that achieve a parking orbit.

5.2 (Continued)

The data of Figure 5-1 are valid for a launch azimuth variation from 72 degrees to 108 degrees east of north. The nominal and S-IC dual-engine-out altitude/surface range profiles shown are valid across the launch azimuth spread.

Use of the surveillance contours defined in Figure 5-1 does not identify specific acquisition and loss times or parameters for each possible abort or alternate mission. Analysis has shown, however, that coverage for all alternate missions resembles predictions made for a nominal flight during the boost flight interval.

The earth parking orbits that result from alternate missions are classified into the following categories:

a. Alternate missions that achieve a near-nominal earth parking orbit.

b. Alternate missions that achieve an off-nominal elliptical parking orbit.

The alternate mission trajectories that achieve a near-nominal earth parking orbit result from propulsion system or subsystem failures that reduce the vehicle's overall performance capability. These trajectories are discussed in detail in Paragraph 5.3.

Alternate missions that achieve an elliptical earth parking orbit are the result of accelerometer malfunctions. An analysis of the alternate missions, resulting from accelerometer malfunctions, is presented in Paragraph 5.4.

The surveillance analysis applicable to the particular instance where no S-IVB second burn occurs consists of extending the nominal earth parking orbits for the AS-507 G Mission. Additional analyses are provided to define surveillance for September 13, 1969, 78.051-degree launch azimuth, second opportunity for the special instance of an S-IVB stage early cutoff. This analysis is discussed in Paragraph 5.5.

A study of station acquisition azimuths is included to augment the station acquisition aid systems that are limited to a 20-degree-wide antenna pattern. This study includes a presentation of all acquisition azimuths that deviate more than 10 degrees from the corresponding nominal azimuth during the critical first orbit after insertion.

5.3 SURVEILLANCE FOR NONACCELEROMETER MALFUNCTIONS

Alternate missions resulting from nonaccelerometer malfunctions are defined in Table 5-III and are classified into the following types:

a. Single and dual S-IC and S-II stage engine malfunctions.

b. Malfunctions resulting in early staging from the S-II to the S-IVB stage.

The earth parking orbits resulting from the defined failures lie in the plane of, and closely approximate, the nominal earth parking orbit as defined by the AS-507 G Mission operational trajectory. The groundtracks of these orbits are given in Figures 5-2, 5-3, and 5-4. Each of the nonaccelerometer failures, however, achieves earth parking orbit insertion at a later flight time than does the nominal.

The difference in station acquisition and loss times, obtained by comparing a delayed-nominal orbit with the corresponding nominal parking orbit, is referred to as delta time. The delta time that is related to the time that a nonaccelerometer malfunction occurs is approximately constant from station to station for any orbit or for any launch azimuth.

The delta time for a particular time of nonaccelerometer malfunction is computed by recording the time and longitude after the malfunctioning vehicle has achieved orbital insertion and by differencing this time with the time at which the corresponding nominal vehicle reaches the same longitude in the nominal parking orbit.

The delta times computed for nonaccelerometer failures that occur between liftoff and nominal S-II cutoff are provided in Figure 5-5. The acquisition and loss times for a delayed parking orbit are determined by adding the delta time from Figure 5-5 to the nominal acquisition and loss times of Figures 5-6, 5-7, and 5-8.

The endpoints (designated by solid lines) of the delta time curves for the nonaccelerometer failures shown in Figure 5-5 represent the extent of the trajectory data available for analysis. The analysis of the engine-out and early staging malfunctions begins with the earliest malfunctioning vehicle that achieves parking orbit insertion.

The dashed lines on the delta time curves in Figure 5-5 represent an extrapolation from the last actual trajectory data available to the known terminal condition.

5.3 (Continued)

In this analysis, the delta time is computed for the first orbit and is a good approximation of the delta times for the second and third orbits.

No significant acquisition azimuth deviations from the nominal occur for the nonaccelerometer malfunctions.

5.4 SURVEILLANCE FOR ALTERNATE MISSIONS RESULTING FROM ACCELEROMETER FAILURES

The specific measurements normally supplied to the guidance and control system by an accelerometer are replaced by guidance-computed estimates when an accelerometer malfunction occurs. The errors in these guidance estimates, in general, cause inplane overspeed and flight-path-angle errors at orbital insertion. The resultant orbits are characterized by apogees varying between 185 and 1815 kilometers and by perigees varying between 99 and 187 kilometers.

The apogee and perigee altitudes resulting from X- and Z-accelerometer malfunctions are described in Figures 4-58 and 4-59. These altitudes are a function of accelerometer malfunction time during the boost phase for both a nominal and $\pm 3\sigma$ performance vehicle. The property of most importance to the tracking and communications analysis is that malfunctions during boost result in earth parking orbits that have related orbital properties.

The Z-accelerometer malfunctions have related parking orbits since the resultant perigees are approximately a single-valued function of apogee. The X-accelerometer malfunctions do not have this property, and the surveillance data are presented as a function of X-accelerometer time of malfunction

Station acquisition and loss flight time histories for X-accelerometer failures occurring on a nominally performing vehicle are presented in Figures 5-9 through 5-12. These data are presented as a function of failure time after liftoff. The data of these figures approximate the acquisition and loss times for the $\pm 3\sigma$ performance vehicle for malfunctions occurring after 60 seconds. An early X-accelerometer malfunction occurring on a -3σ performance vehicle results in an abort situation.

5.4 (Continued)

A 10-degree deviation in acquisition azimuth from the nominal surveillance occurs only for Honeysuckle. The acquisition azimuth as a function of time of X-accelerometer malfunction is described in Figure 5-13 for this surveillance station.

Station acquisition and loss flight time histories are presented for Z-accelerometer (both $\pm 3\sigma$ and nominally performing vehicles) failures as a function of orbital apogee altitude. (See Figures 5-14 through 5-17.) These data are related to an accelerometer malfunction as follows:

a. The apogee altitude is determined for a malfunction at a specific time during launch by using the data of Figure 4-59.

b. The station acquisition and loss time is determined as a function of the apogee altitude by using the data of Figures 5-14 through 5-17.

A 10-degree deviation in acquisition azimuth from the nominal surveillance occurs for the Tananarive, Carnarvon, Honeysuckle, and Goldstone tracking stations. The acquisition azimuths for these stations is provided as a function of apogee in Figure 5-18.

5.5 S-IVB SECOND-BURN EARLY CUTOFF SURVEILLANCE ANALYSIS

The analysis for the S-IVB second-burn early cutoff is divided into two separate studies. The first study concerns the instance of no S-IVB stage restart ignition and is the most likely malfunction of this kind. The surveillance analysis for the no-restart stage ignition is provided across the launch window for the launch-azimuth variation from 72 degrees to 108 degrees. The data provided in Figures 5-7, 5-8, and in Figures 5-19 through 5-21 define the surveillance for the instance of no S-IVB stage reignition and is based upon the continuation of the earth parking orbits for the AS-507 G Mission.

The second analysis describes the type of tracking and communications surveillance expected for a typical launch day, azimuth, and TLI opportunity for an S-IVB early cutoff occurring at any instant during the second burn. This particular study was completed for the September 13, 1969, second TLI opportunity and for a 78.051-degree launch azimuth. (See Figure 5-22.) These data are shown only to 19,000 seconds because of the difference in the orbital period for orbits resulting from the different failure times. The acquisition azimuths for an early S-IVB second burn cutoff for the Canary Island and Madrid stations deviate by more than 10 degrees from the nominal second-burn and post-TLI

5.5 (Continued)

surveillance. The acquisition azimuths for these two stations as a function of time of malfunction during the second burn are given in Figure 5-23. Acquisition azimuths for the special instance where no S-IVB stage second burn occurs is provided in Table 5-II.

The Y-accelerometer malfunctions result in out-of-plane errors at insertion. A limited analysis of these malfunctions is represented by the station acquisition and loss history for a typical worst case. (See Table 5-IV.) No acquisition azimuth deviations exceed 10 degrees from the nominal surveillance.

TABLE 5-I. SURVEILLANCE NETWORK USED FOR AS-507 G MISSION COMMUNICATIONS ANALYSIS FOR ABORT AND ALTERNATE MISSIONS

STATION	SYSTEM	SYMBOL	GEODETIC LATITUDE (DEGREES)	LONGITUDE EAST (DEGREES)	HEIGHT ABOVE ELLIPSOID (METERS)
MERRITT ISLAND	CCS, TM	MIL	28.508272	- 80.693417	10.00
GRAND BAHAMA	CCS, TM	GBM	26.632857	- 78.237664	5.00
BERMUDA	CCS, TM	BDA	32.351286	- 64.658181	21.00
ANTIGUA	CCS, TM	ANT	17.016916	- 61.752849	43.00
INSERTION SHIP	CCS, TM	INS	25.00000	- 49.000000	0.00
CANARY ISLAND	CCS, TM	CYI	27.764536	- 15.634814	173.00
ASCENSION	CCS, TM	ASC	- 7.955056	- 14.327578	562.00
MADRID	CCS	MAD	40.455358	- 4.167394	825.00
TANANARIVE	TM	TAN	-19.000797	47.315053	1322.30
CARNARVON	CCS, TM	CRO	-24.907592	113.724247	58.00
GUAM	CCS, TM	GWM	13.309244	144.734414	127.00
HONEYSUCKLE	CCS	HSK	-35.597222	148.979167	1097.00
HAWAII	CCS, TM	HAW	22.124897	-159.664989	1150.00
GOLDSTONE	CCS	GDS	35.341694	-116.873289	965.00
GUAYMAS	CCS, TM	GYM	27.963206	-110.720850	19.00
CORPUS CHRISTI	CCS, TM	TEX	27.653750	- 97.378469	10.00

NOTE: ALL GEODETIC DATA ARE REFERENCED TO THE FISCHER ELLIPSOID.

CCS — COMMAND AND COMMUNICATIONS SYSTEM

TM — TELEMETRY

TABLE 5-II. SUMMARY OF THE 0-DEGREE SURVEILLANCE FOR THE NOMINAL VEHICLE DURING EPO (NO S-IVB SECOND BURN)

LAUNCH AZIMUTH 78.051°	ORBIT											
	BOOST & FIRST			SECOND			THIRD			FOURTH		
STATION SYMBOL	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)
MIL	28.3	0	533	278.2	5702	6137	280.4	11296	11736	264.3	16912	17292
GBM	314.1	66	562	287.0	5742	6166	286.8	11336	11776	268.7	16948	17341
BDA	257.4	242	760	266.5	5915	6340	254.9	11527	11878	X	X	X
ANT	344.0	514	639	334.3	6081	6345	311.5	11623	12036	279.4	17221	17629
INS	296.0	556	967	289.6	6144	6579	265.4	11761	12128	X	X	X
CYI	269.5	1006	1396	X	X	X	X	X	X	X	X	X
ASC	X	X	X	X	X	X	315.1	12449	12867	263.1	18077	18434
TAN	318.0	2182	2579	271.8	7766	8170	233.3	13435	13736	214.4	19089	19358
CRO	244.9	3123	3519	246.7	8711	9139	265.6	14312	14719	X	X	X
GWM	X	X	X	X	X	X	171.9	15132	15317	235.3	20593	21035
HSK	304.3	3626	3865	X	X	X	X	X	X	X	X	X
HAW	X	X	X	225.9	10215	10633	267.3	15804	16234	297.1	21451	21828
GDS	209.4	5306	5628	241.0	10832	11240	254.0	16411	16821	247.9	22025	22362
GYM	250.8	5293	5730	275.7	10889	11326	283.7	16491	16934	274.1	22096	22521
TEX	268.7	5472	5907	283.9	11074	11510	281.4	16672	17113	259.7	22303	22655

NOTE: X - THIS STATION DOES NOT TRACK.
AZI- STATION ACQUISITION AZIMUTH.

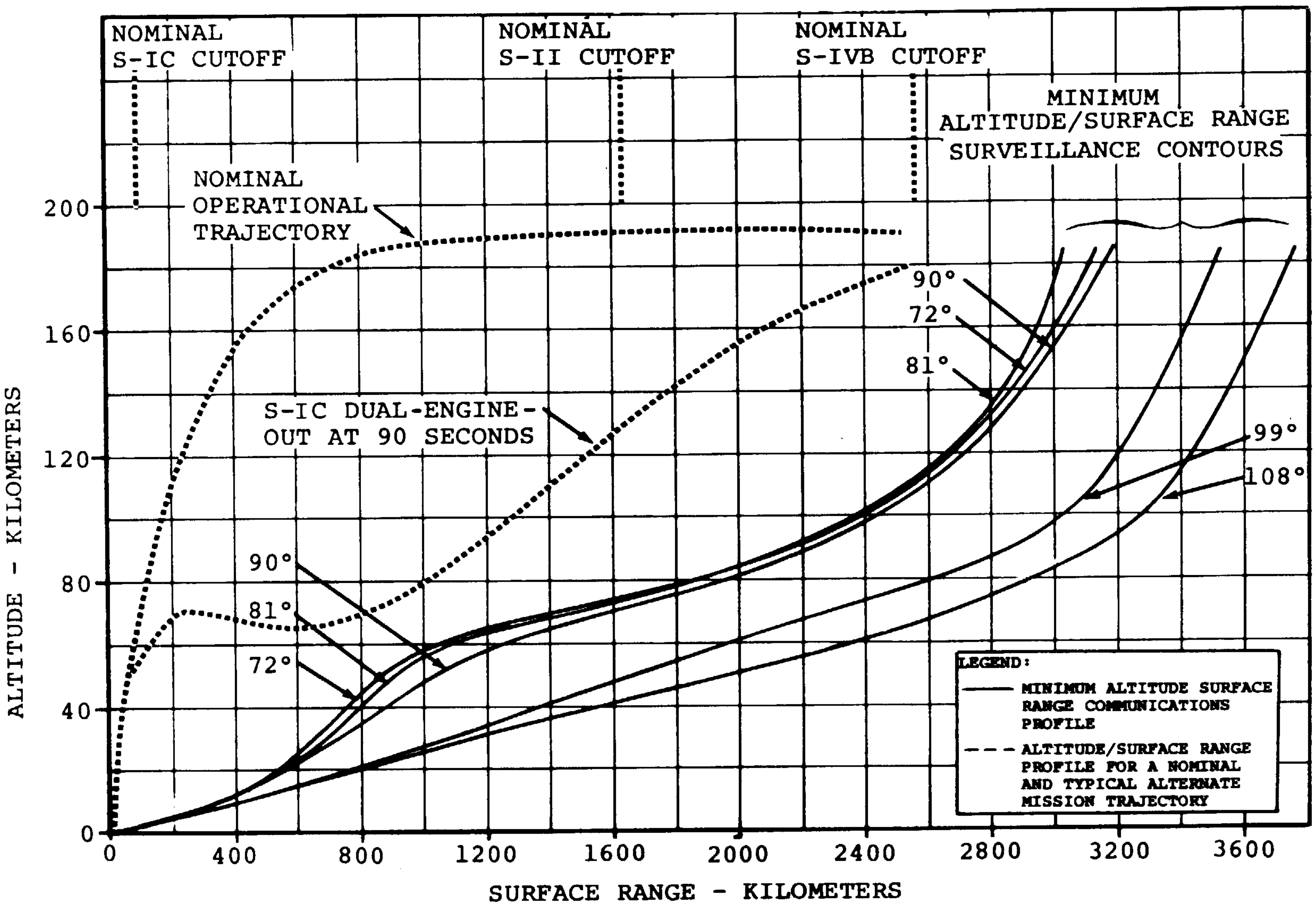
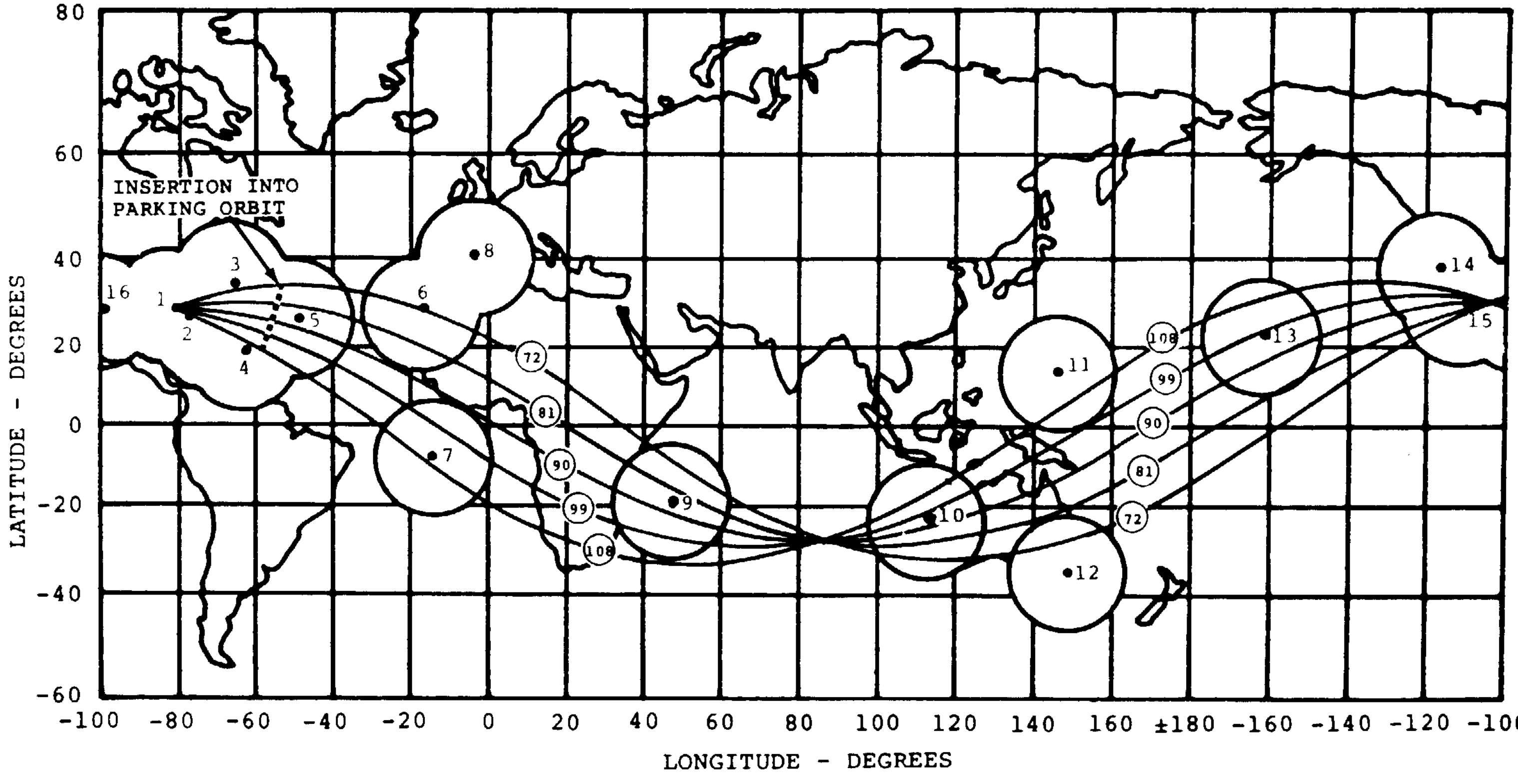


FIGURE 5-1 EASTERN TEST RANGE MINIMUM ALTITUDE/SURFACE RANGE 0-DEGREE ELEVATION ANGLE PROFILE FOR CONTINUOUS TRACKING COVERAGE BASED ON MILA, GRAND TURK, GRAND BAHAMA, ANTIGUA, AND BERMUDA GROUND STATIONS

Table 5-III. ALTERNATE MISSIONS ACHIEVING A NOMINAL PARKING ORBIT

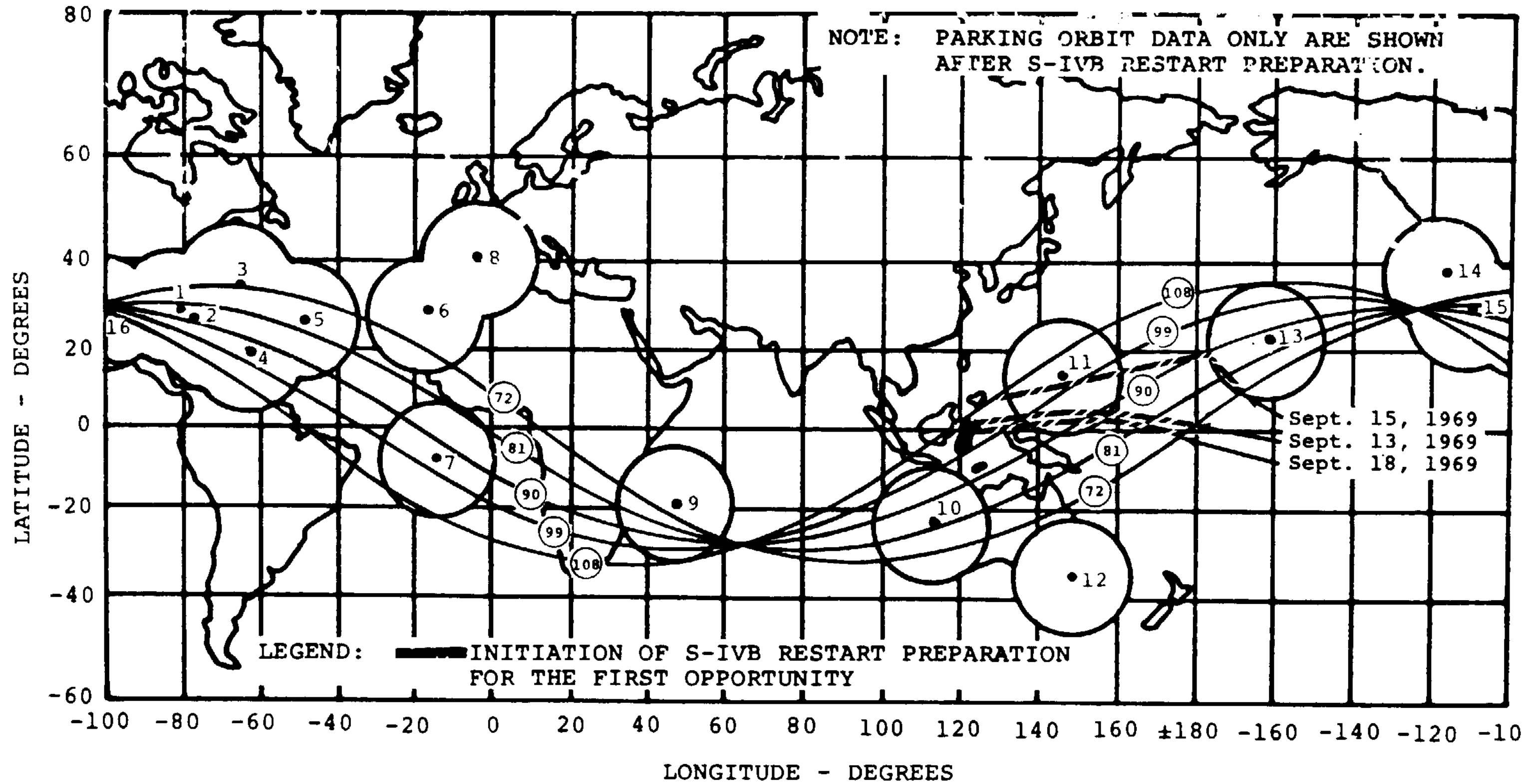
SYMBOLS USED ON ALL DATA PLOTS	TYPE OF FAILURE
S-IC 1 EO	Single-engine-out (Engine No. 1) failure in the S-IC stage.
S-IC 2 EO	Single-engine-out (Engine No. 2) failure in the S-IC stage.
S-IC 1+5 EO	Dual-engine-out (Engines No. 1 and No. 5) failure in the S-IC stage.
S-IC 2+4 EO	Dual-engine-out (Engines No. 2 and No. 4) failure in the S-IC stage.
S-II 1 EO	Single-engine-out (Engine No. 1) failure in the S-II stage.
S-II 1+5 EO	Dual-engine-out (Engines No. 1 and No. 5) failure in the S-II stage.
S-II 1+4 EO	Dual-engine-out (Engines No. 1 and No. 4) failure in the S-II stage.
S-II 2+4 EO	Dual engines out (Engines No. 2 and No. 4) on the S-II stage.
S-II/S-IVB ES	Early staging of the S-II to the S-IVB stage.



STATION LEGEND:

1. CAPE AREA: MILA	5. INSERTION SHIP	9. TANANARIVE	13. HAWAII
2. GRAND BAHAMA	6. CANARY ISLAND	10. CARNARVON	14. GOLDSTONE
3. BERMUDA	7. ASCENSION ISLAND	11. GUAM	15. GUAYMAS
4. ANTIGUA	8. MADRID	12. HONEYSUCKLE	16. CORPUS CHRISTI

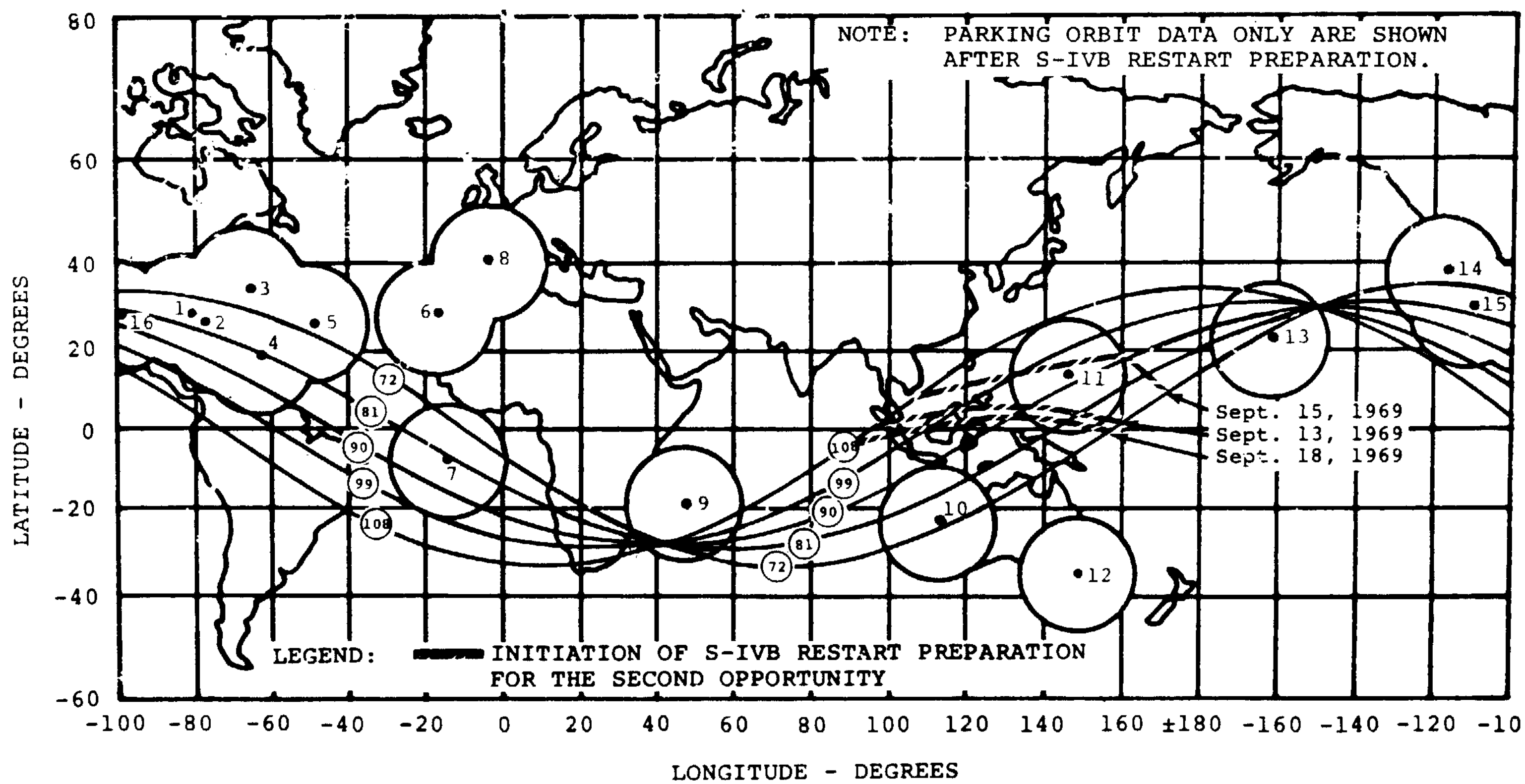
FIGURE 5-2 NOMINAL FIRST PARKING ORBIT GROUNDTRACK WITH 0-DEGREE ELEVATION ANGLE VISIBILITY ENVELOPES



STATION LEGEND:

1. CAPE AREA: MILA	5. INSERTION SHIP	9. TANANARIVE	13. HAWAII
2. GRAND BAHAMA	6. CANARY ISLAND	10. CARNARVON	14. GOLDSTONE
3. BERMUDA	7. ASCENSION ISLAND	11. GUAM	15. GUAYMAS
4. ANTIGUA	8. MADRID	12. HONEYSUCKLE	16. CORPUS CHRISTI

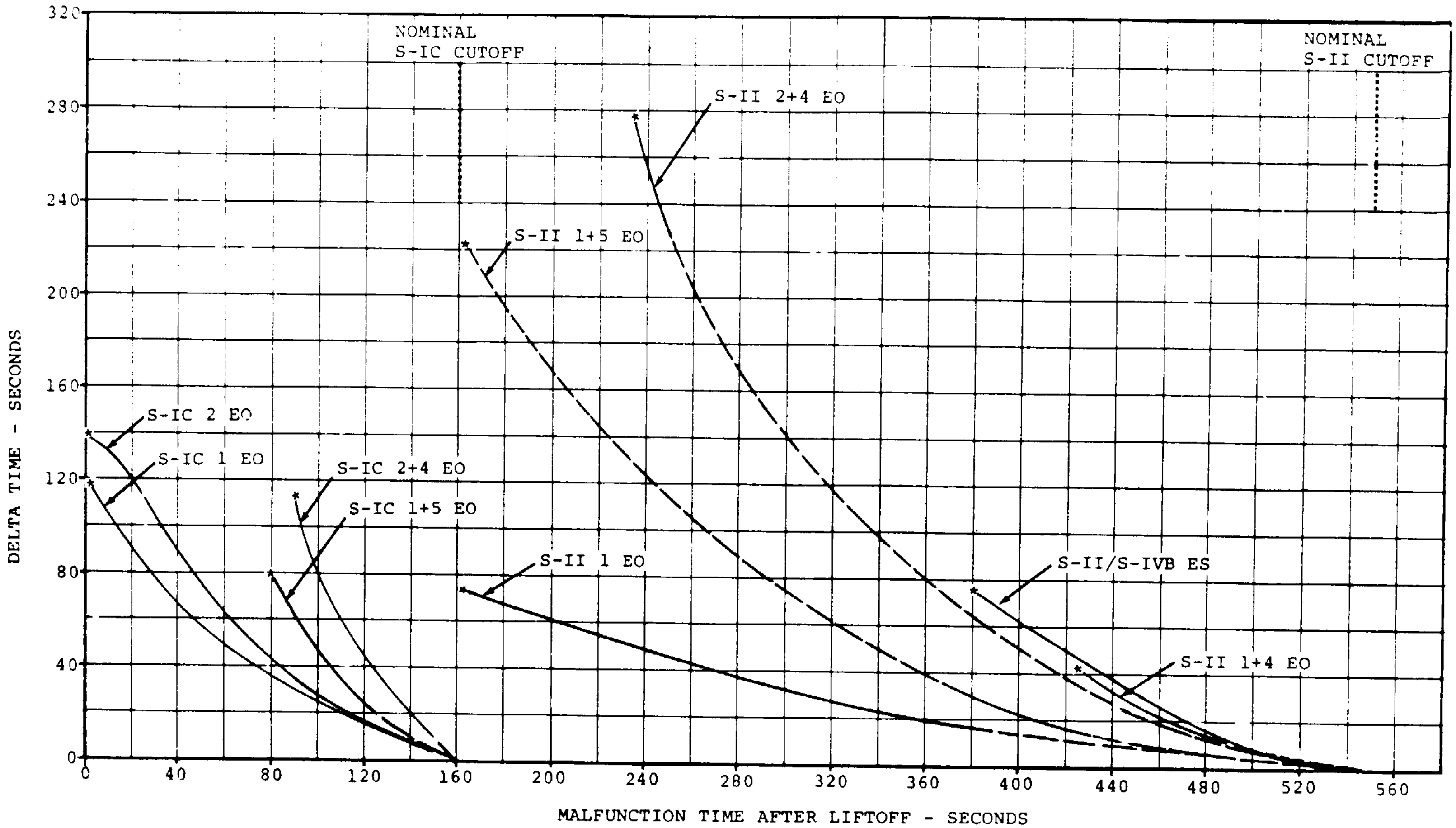
FIGURE 5-3 NOMINAL SECOND PARKING ORBIT GROUNDTRACK WITH 0-DEGREE ELEVATION ANGLE VISIBILITY ENVELOPES



STATION LEGEND:

1. CAPE AREA: MILA	5. INSERTION SHIP	9. TANANARIVE	13. HAWAII
2. GRAND BAHAMA	6. CANARY ISLAND	10. CARNARVON	14. GOLDSTONE
3. BERMUDA	7. ASCENSION ISLAND	11. GUAM	15. GUAYMAS
4. ANTIGUA	8. MADRID	12. HONEYSUCKLE	16. CORPUS CHRISTI

FIGURE 5-4 NOMINAL THIRD PARKING ORBIT GROUNDTRACK WITH 0-DEGREE ELEVATION ANGLE VISIBILITY ENVELOPES



LEGEND:

——— TRAJECTORY DATA AVAILABLE

– – – EXTRAPOLATED DATA

NOTE: * EARLIEST MALFUNCTION OCCURRENCE WITH VEHICLE ACHIEVING PARKING ORBIT INSERTION

FIGURE 5-5 TRACKING DELTA TIMES FOR NONACCELEROMETER MALFUNCTIONS

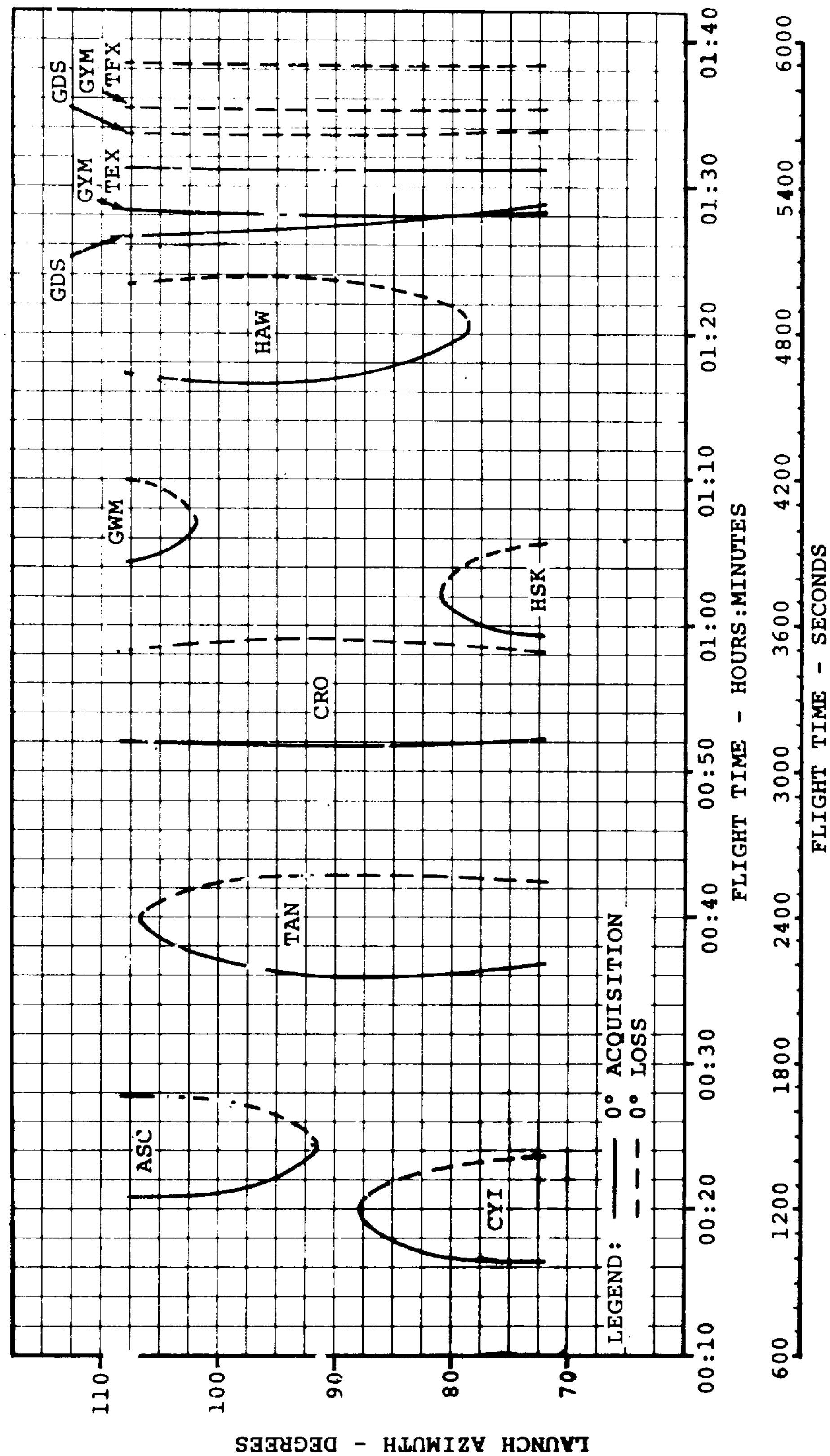


FIGURE 5-6 NOMINAL SURVEILLANCE ABOVE 0 DEGREE FOR FIRST ORBIT

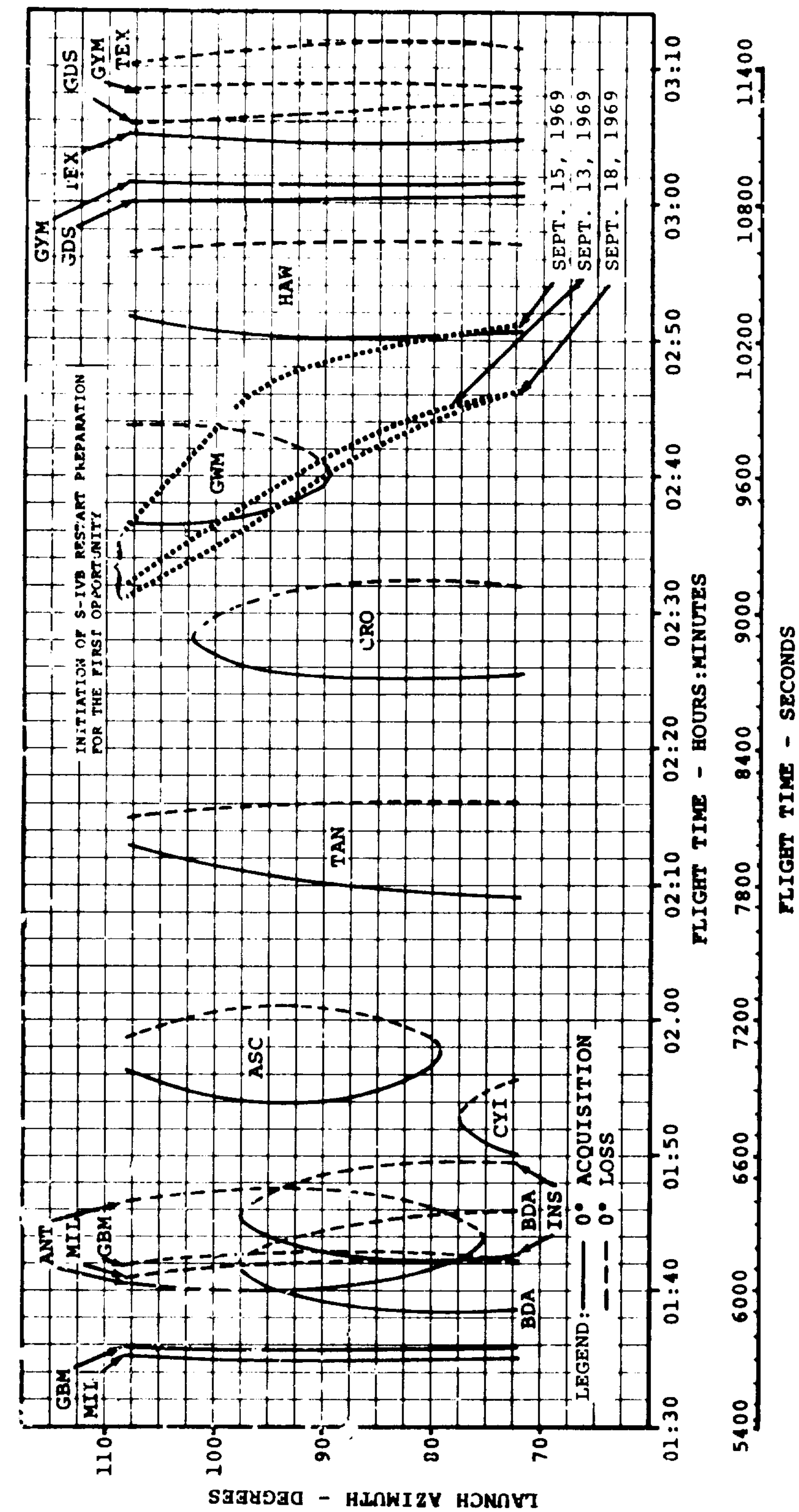


FIGURE 5-7 NOMINAL SURVEILLANCE ABOVE 0 DEGREE FOR SECOND ORBIT

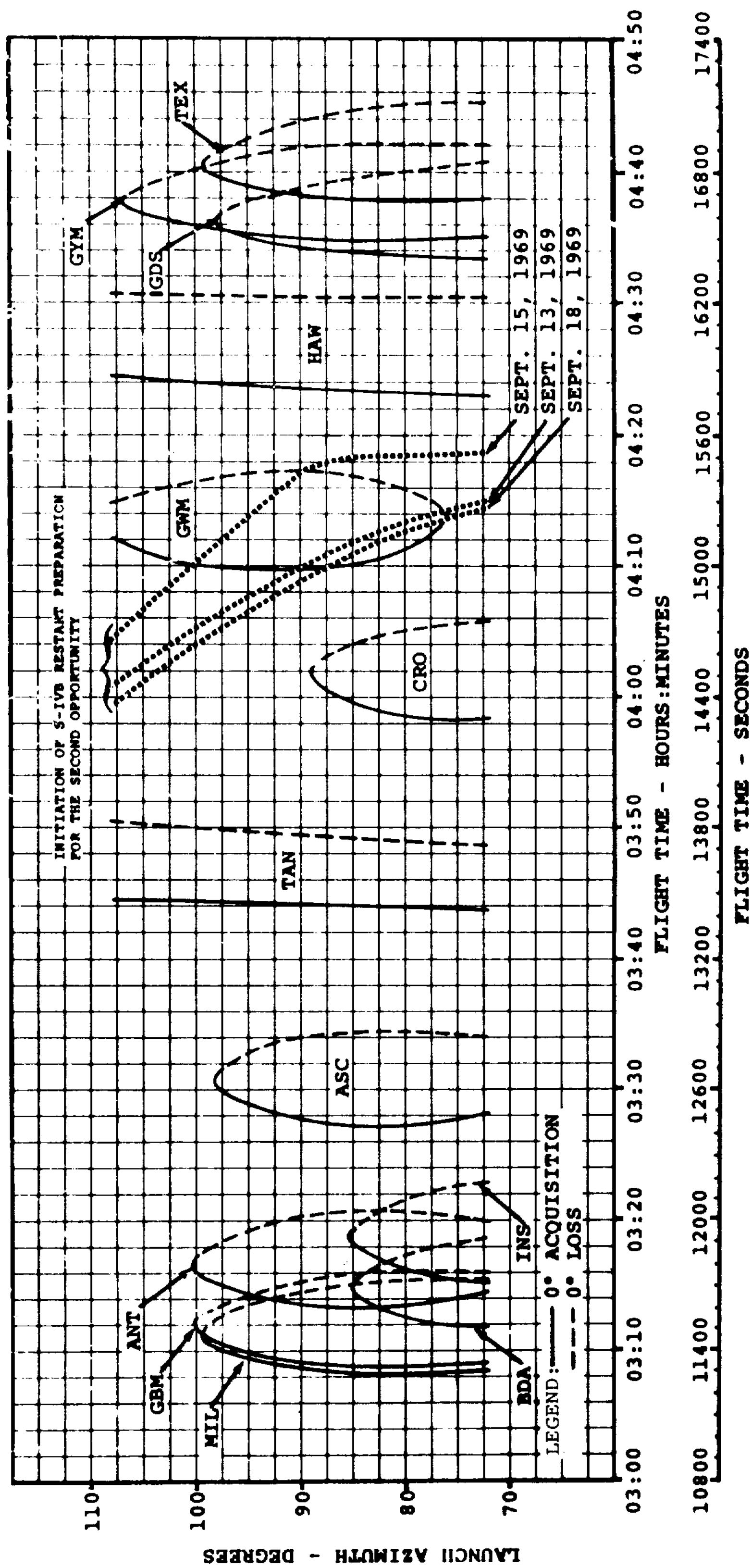
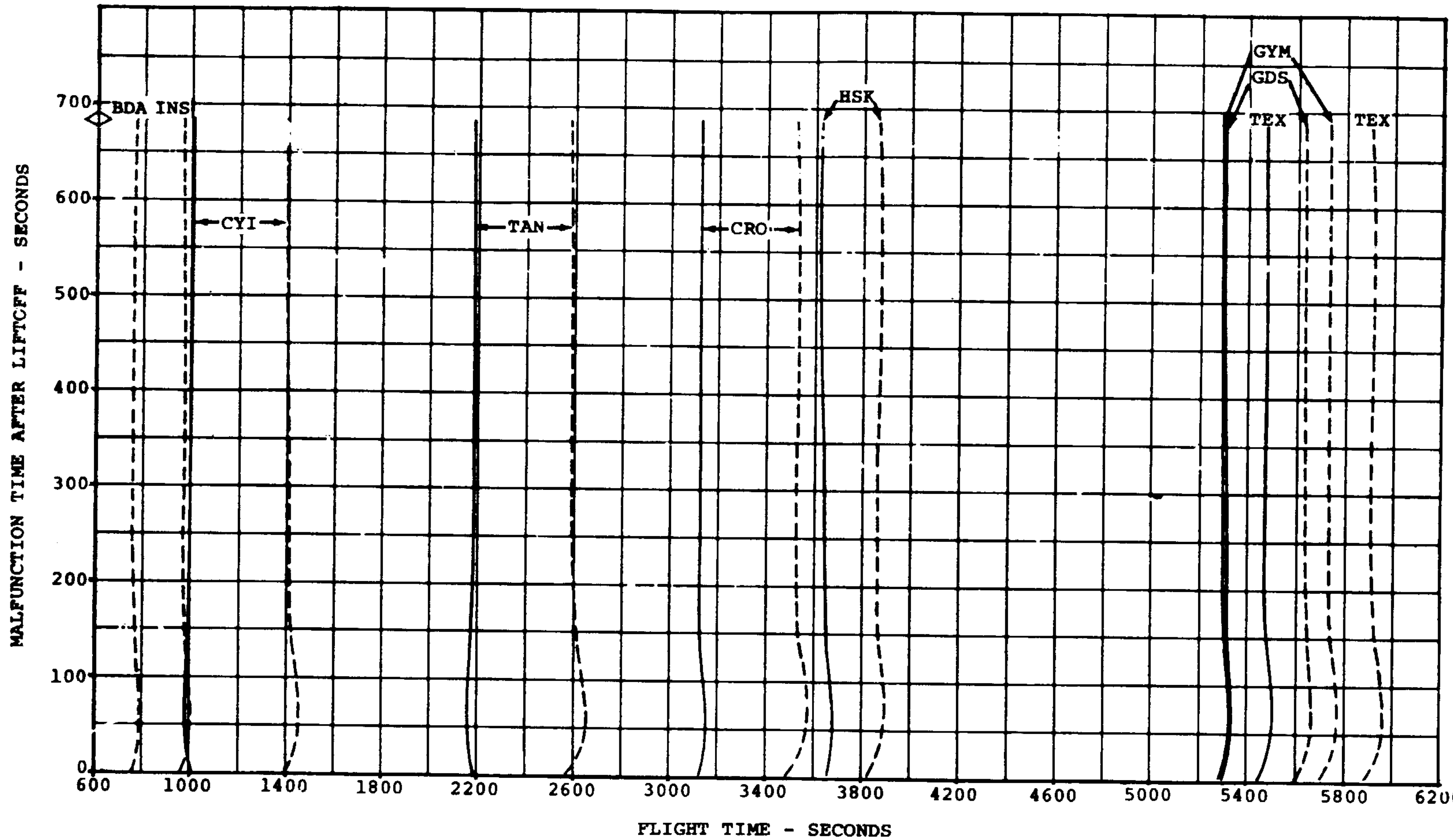


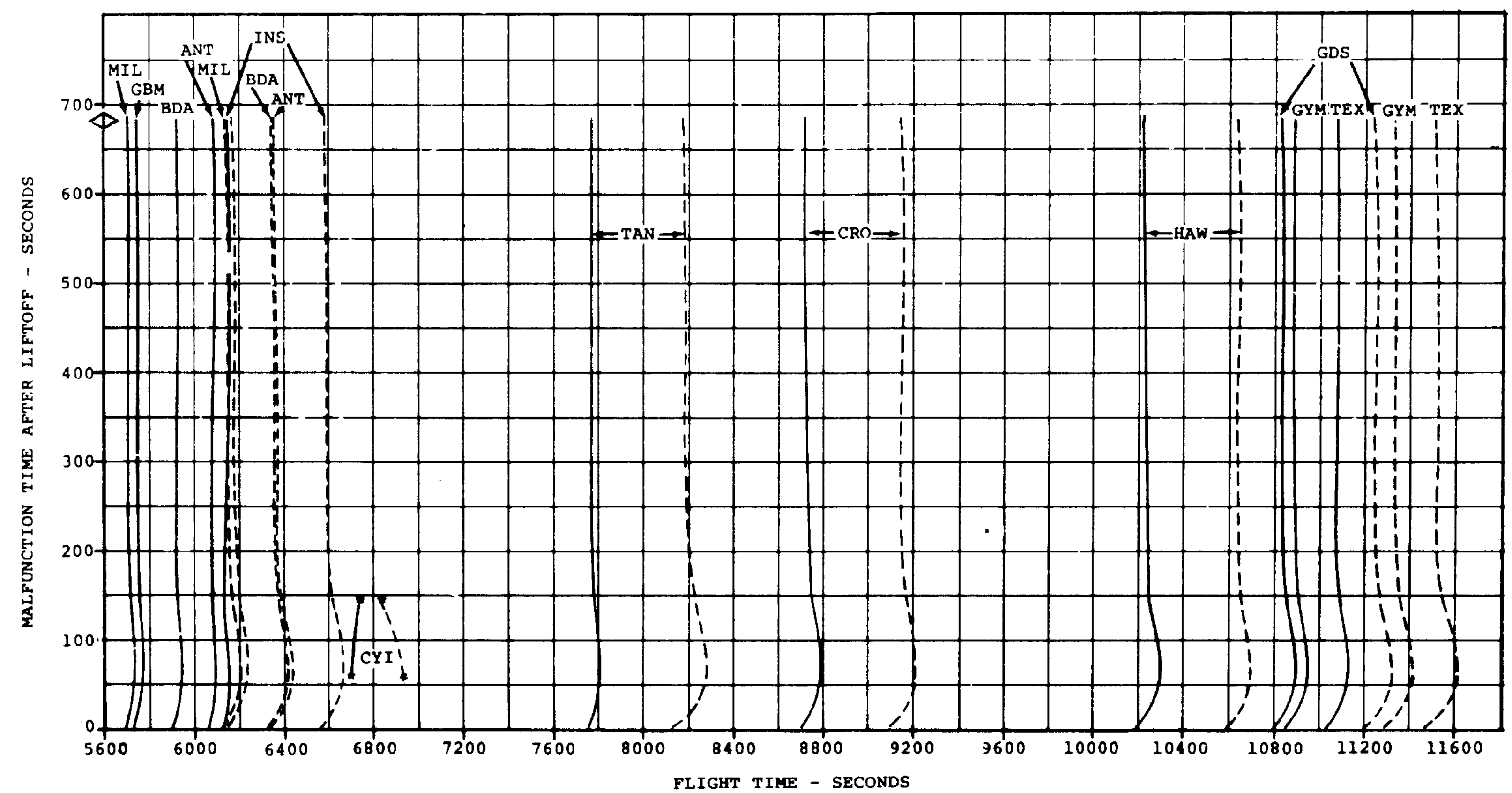
FIGURE 5-8 NOMINAL SURVEILLANCE ABOVE 0 DEGREE FOR THIRD ORBIT



LEGEND:
—— ACQUISITION
--- LOSS

NOTE: ◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE

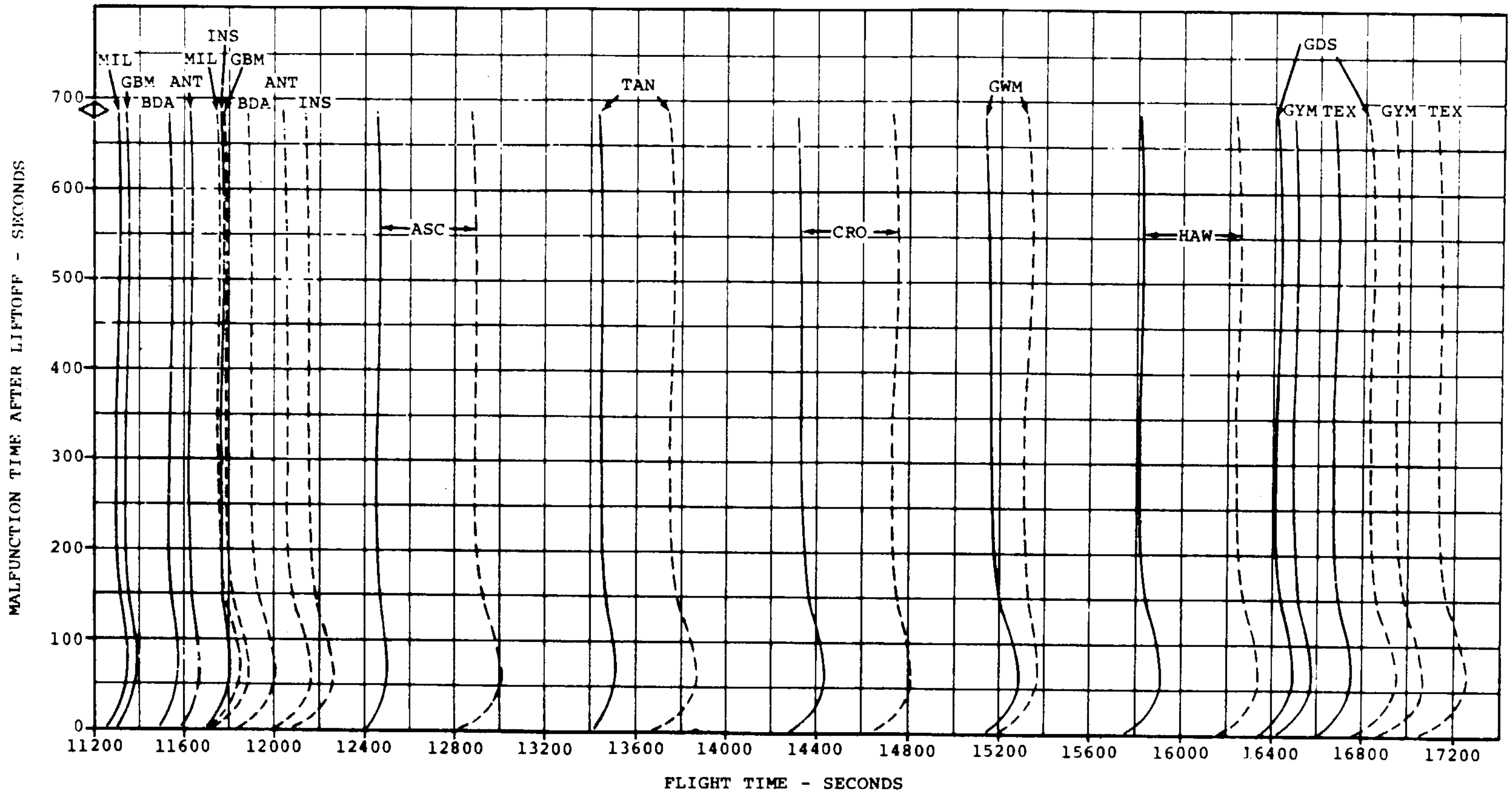
FIGURE 5-9 0-DEGREE ACQUISITION AND LOSS HISTORY FOR FIRST ORBIT FOR X-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
– – – LOSS

NOTE: * DATA NOT AVAILABLE BEYOND THIS POINT.
◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE

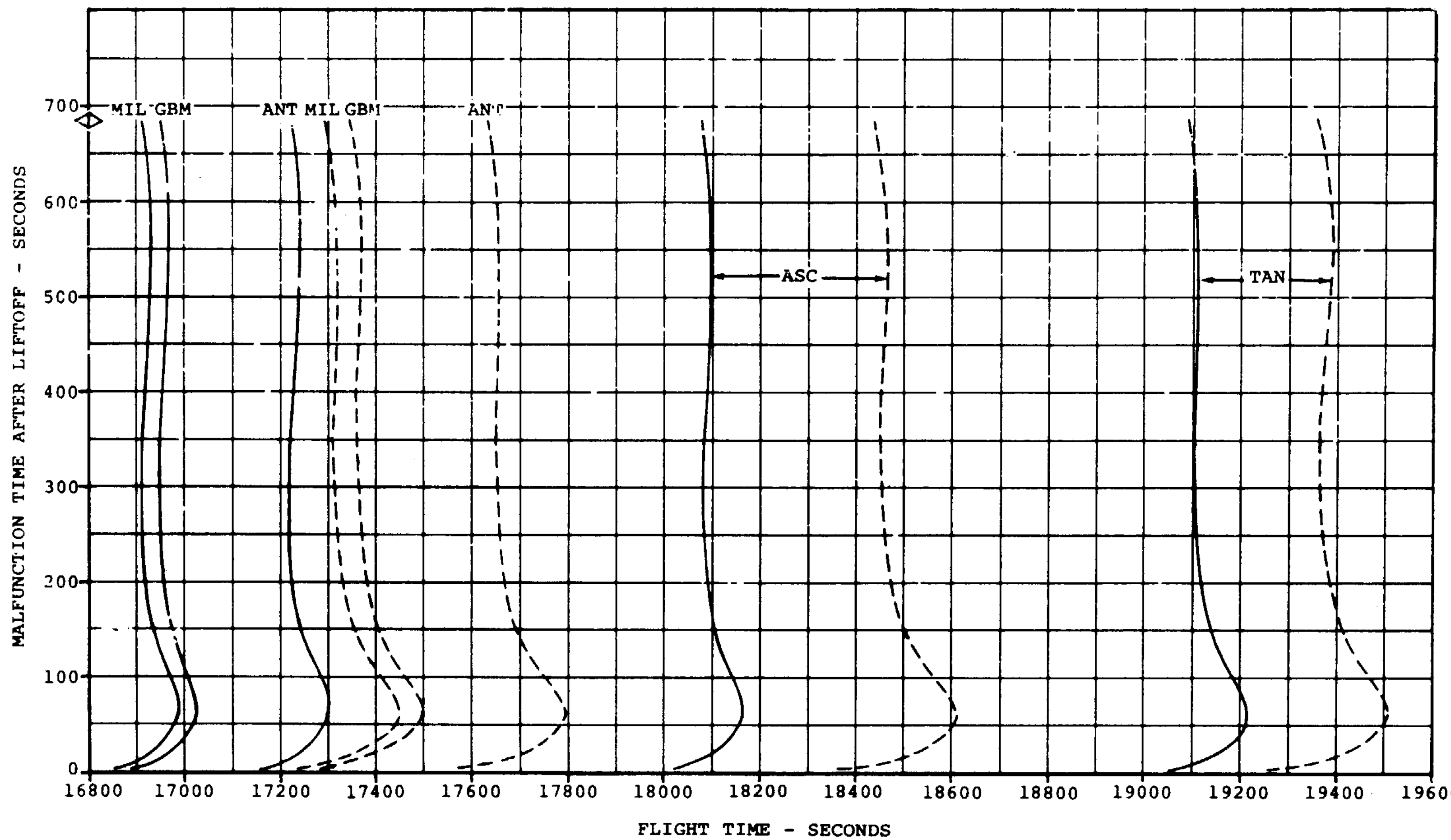
FIGURE 5-10 0-DEGREE ACQUISITION AND LOSS HISTORY FOR SECOND ORBIT FOR X-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
– – – LOSS

NOTE: ◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE (NO S-IVB SECOND BURN)

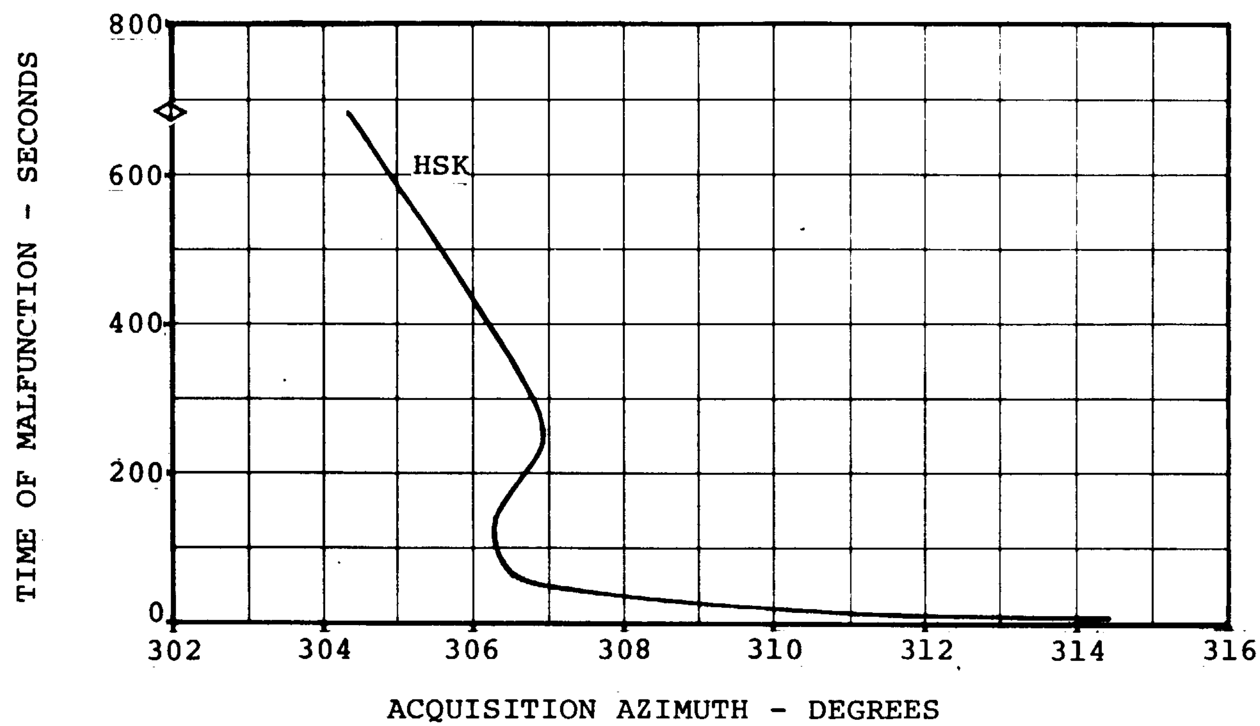
FIGURE 5-11 0-DEGREE ACQUISITION AND LOSS HISTORY FOR THIRD ORBIT FOR X-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
– – – LOSS

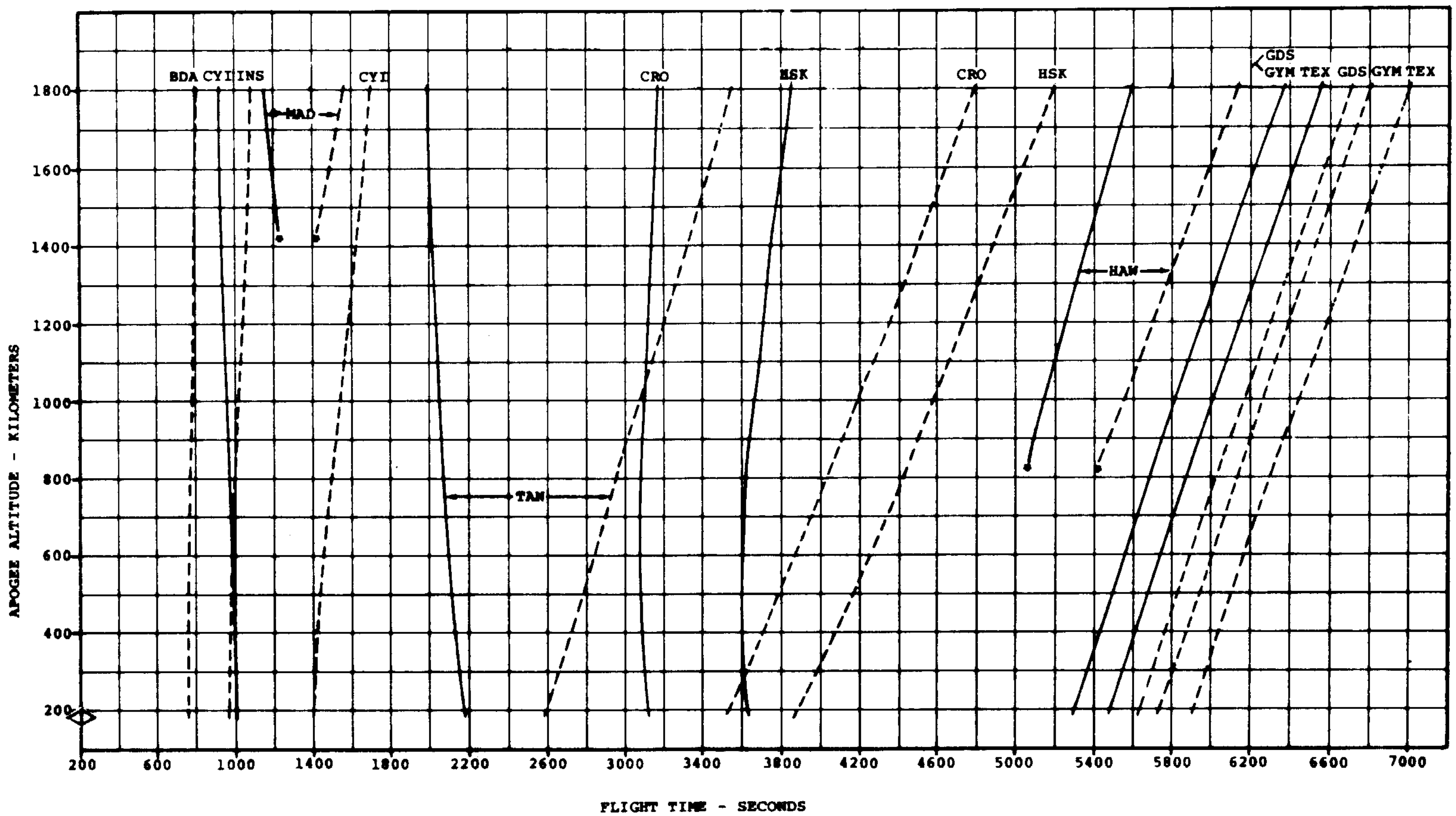
NOTE: ◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE
(NO S-IVB SECOND BURN)

FIGURE 5-12 0-DEGREE ACQUISITION AND LOSS HISTORY FOR FOURTH ORBIT FOR X-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



NOTE: ◇ NOMINAL ACQUISITION AZIMUTH ABOVE 0 DEGREE (NO S-IVB SECOND BURN)

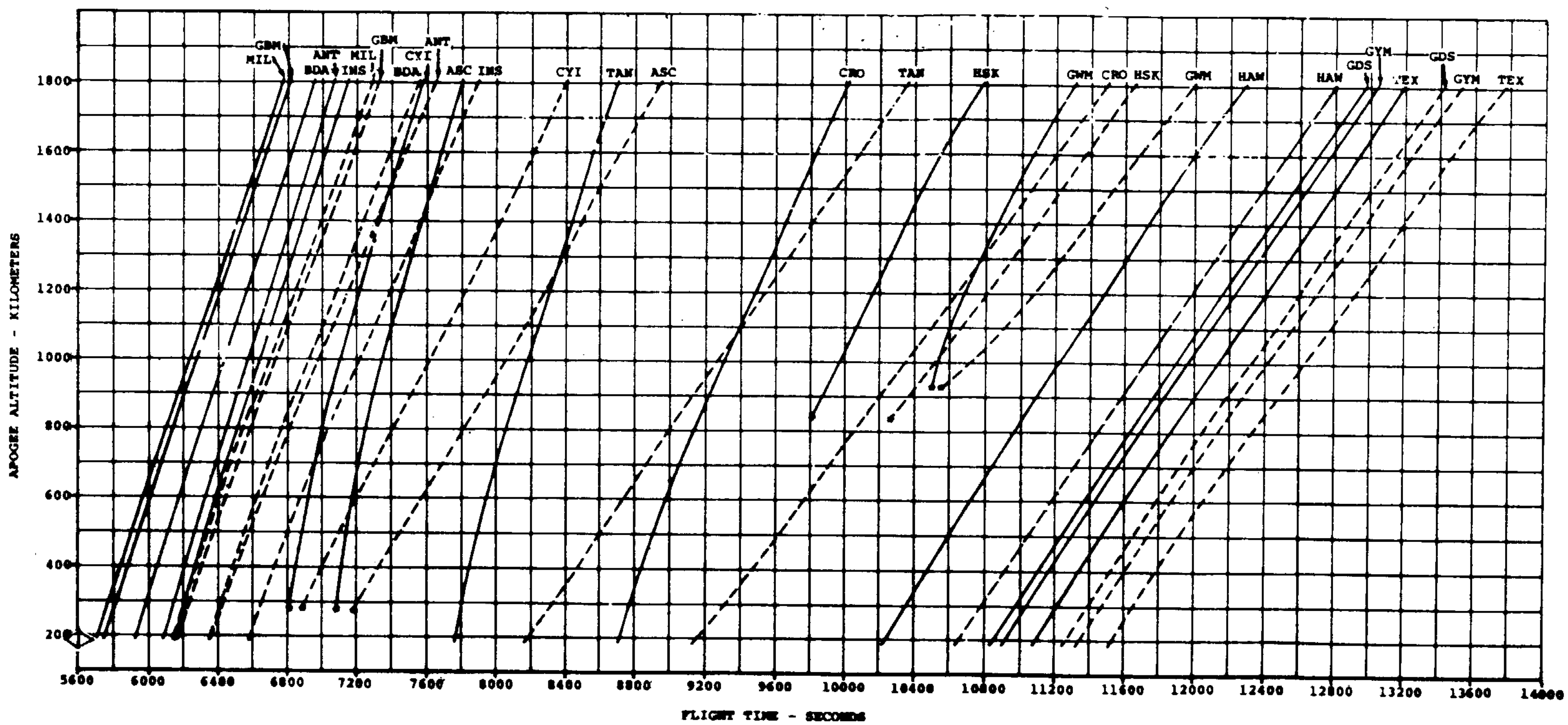
FIGURE 5-13 ACQUISITION AZIMUTH FOR X-ACCELEROMETER MALFUNCTIONS ABOVE 0-DEGREE ELEVATION FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
- - - LOSS

NOTE: * DATA NOT AVAILABLE BEYOND THIS POINT.
◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE.

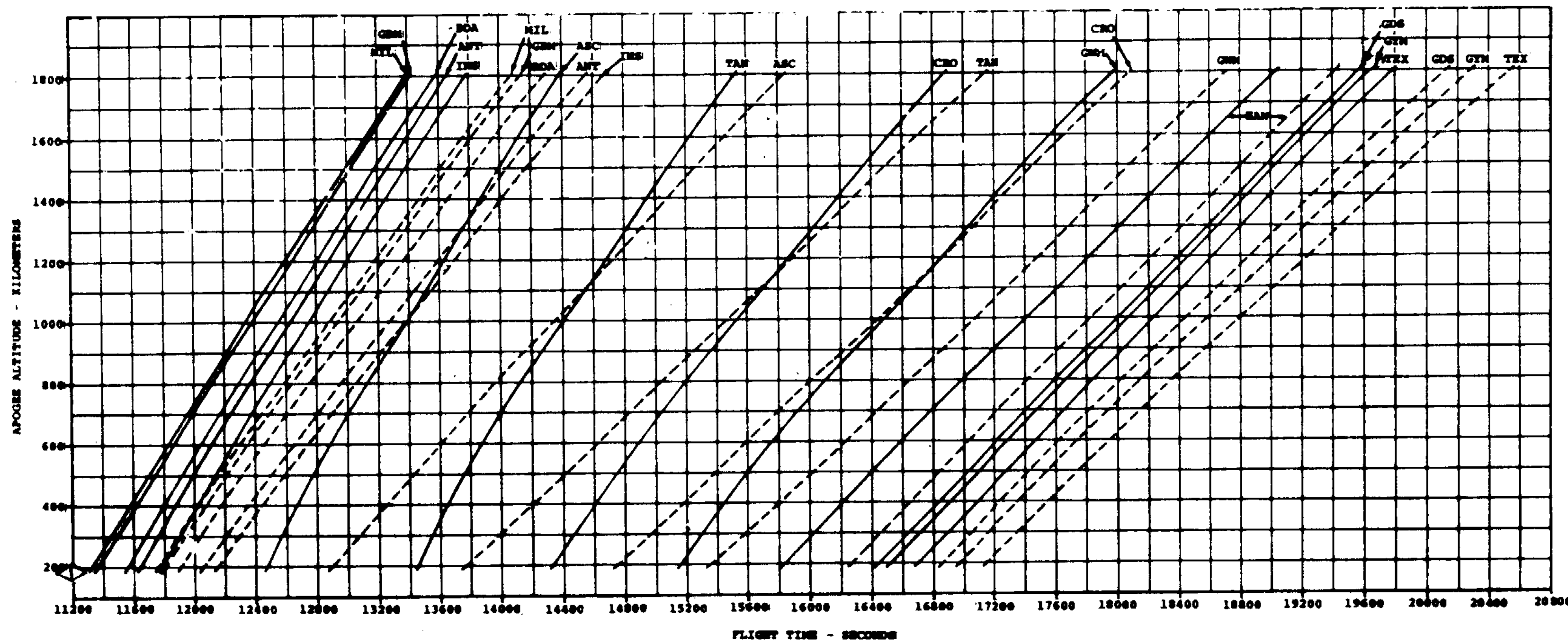
FIGURE 5-14 0-DEGREE ACQUISITION AND LOSS HISTORY FOR FIRST ORBIT FOR Z-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
– – – LOSS

NOTE: * DATA NOT AVAILABLE BEYOND THIS POINT.
◊ NOMINAL SURVEILLANCE ABOVE 0 DEGREE

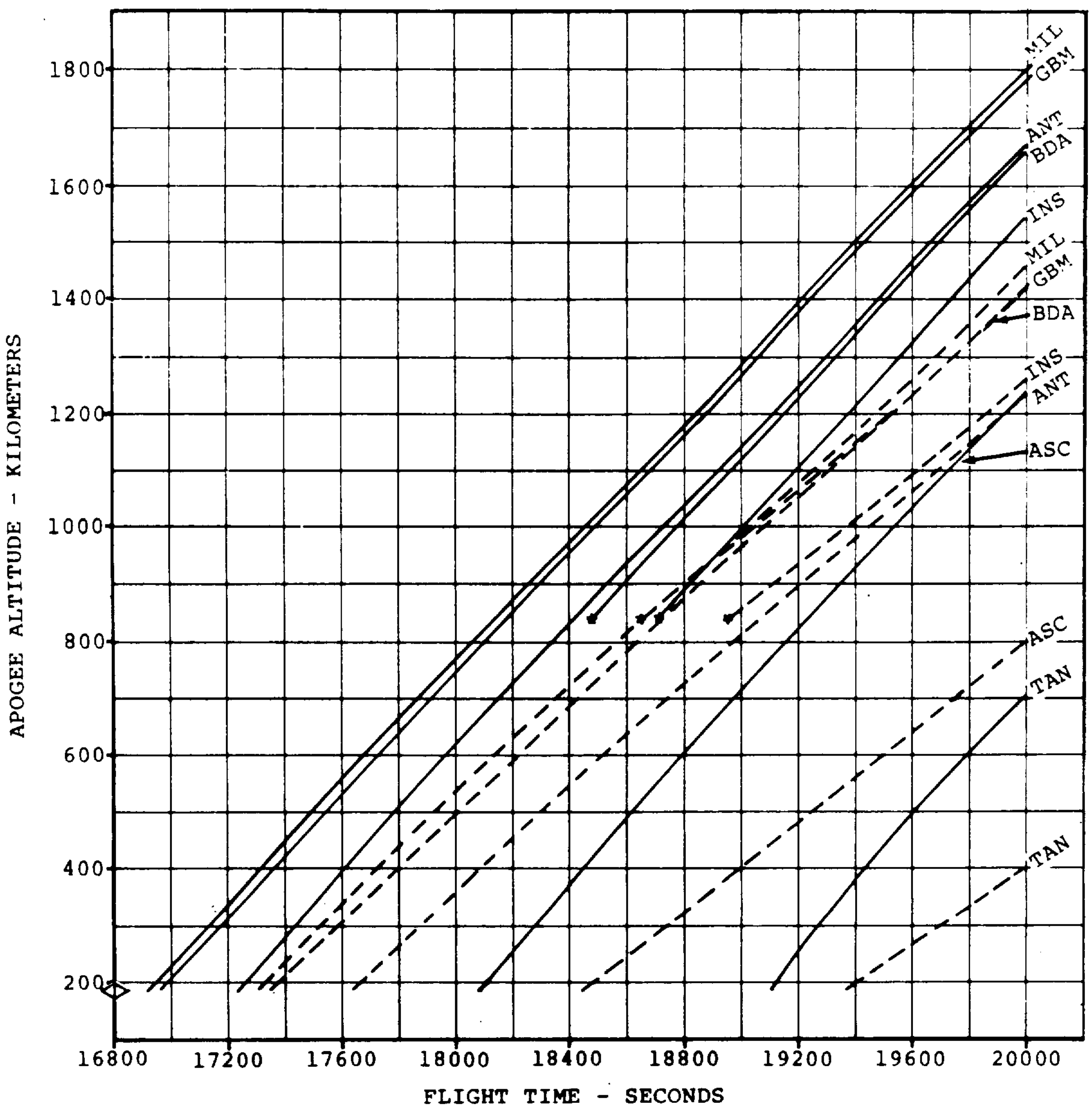
FIGURE 5-15 0-DEGREE ACQUISITION AND LOSS HISTORY FOR SECOND ORBIT FOR Z-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
—— ACQUISITION
– – – LOSS

NOTE: ◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE (NO S-IVB SECOND BURN)

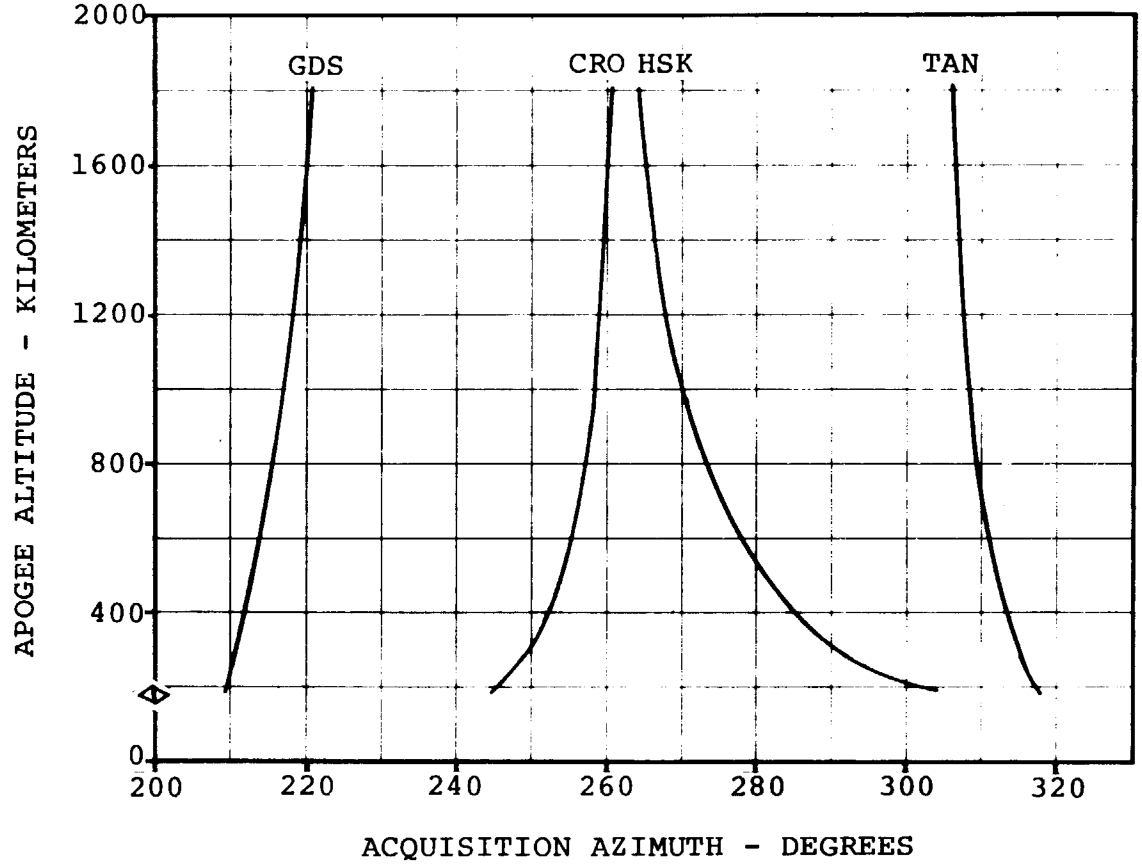
FIGURE 5-16 0-DEGREE ACQUISITION AND LOSS HISTORY FOR THIRD ORBIT FOR Z-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



LEGEND:
——— ACQUISITION
– – – LOSS

NOTE: * DATA NOT AVAILABLE BEYOND THIS POINT.
◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE. (NO S-IVB SECOND BURN)

FIGURE 5-17 0-DEGREE ACQUISITION AND LOSS HISTORY FOR FOURTH ORBIT FOR Z-ACCELEROMETER MALFUNCTIONS FOR 78.051-DEGREE LAUNCH AZIMUTH



NOTE: ◇ NOMINAL ACQUISITION AZIMUTH ABOVE 0 DEGREE (NO S-IVB SECOND BURN)

FIGURE 5-18 ACQUISITION AZIMUTH FOR Z-ACCELEROMETER MALFUNCTIONS ABOVE 0-DEGREE ELEVATION FOR 78.051-DEGREE LAUNCH AZIMUTH

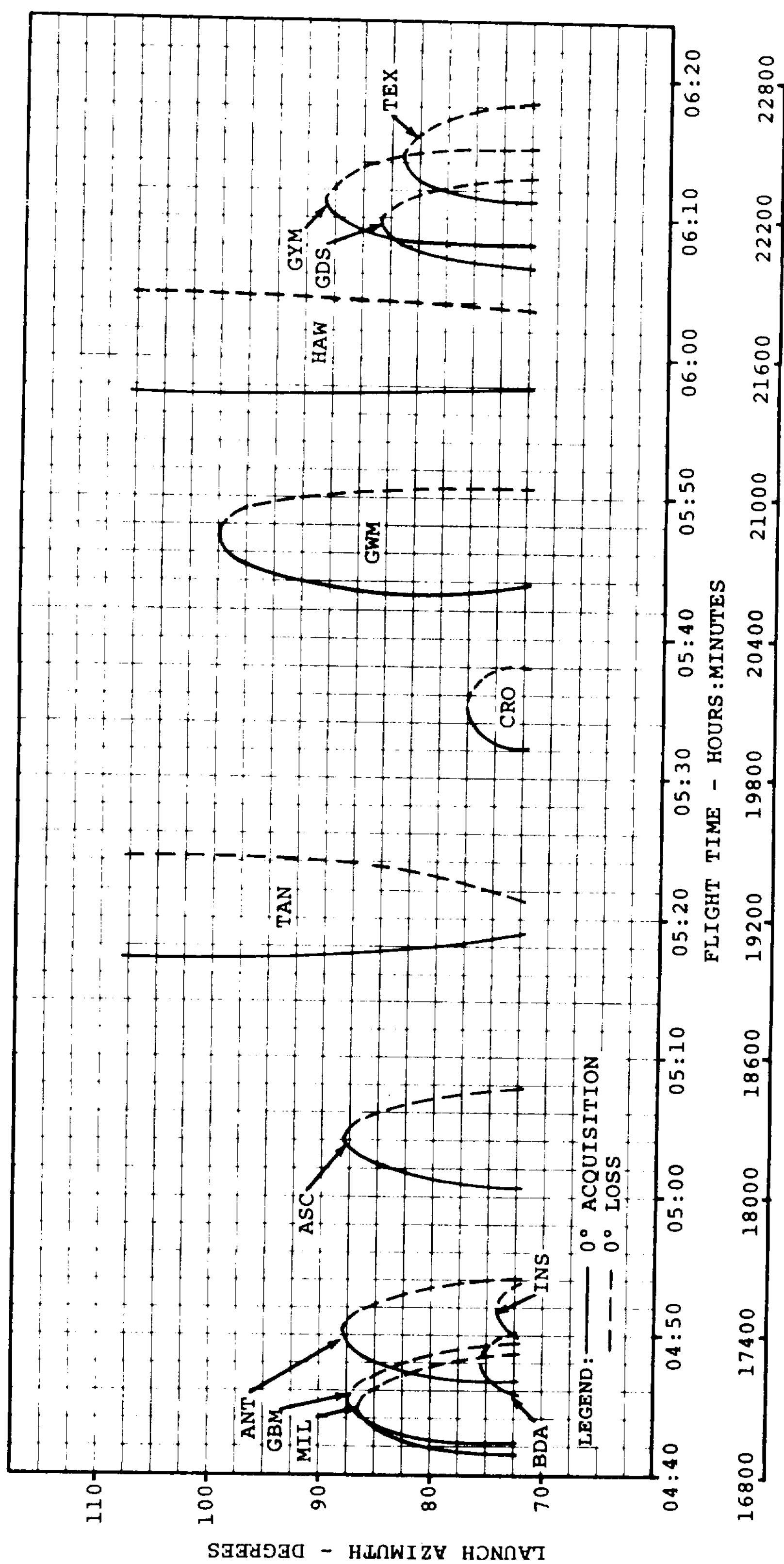


FIGURE 5-19 NOMINAL SURVEILLANCE ABOVE A 0-DEGREE ELEVATION ANGLE FOR FOURTH PARKING ORBIT (NO S-IVB SECOND BURN)

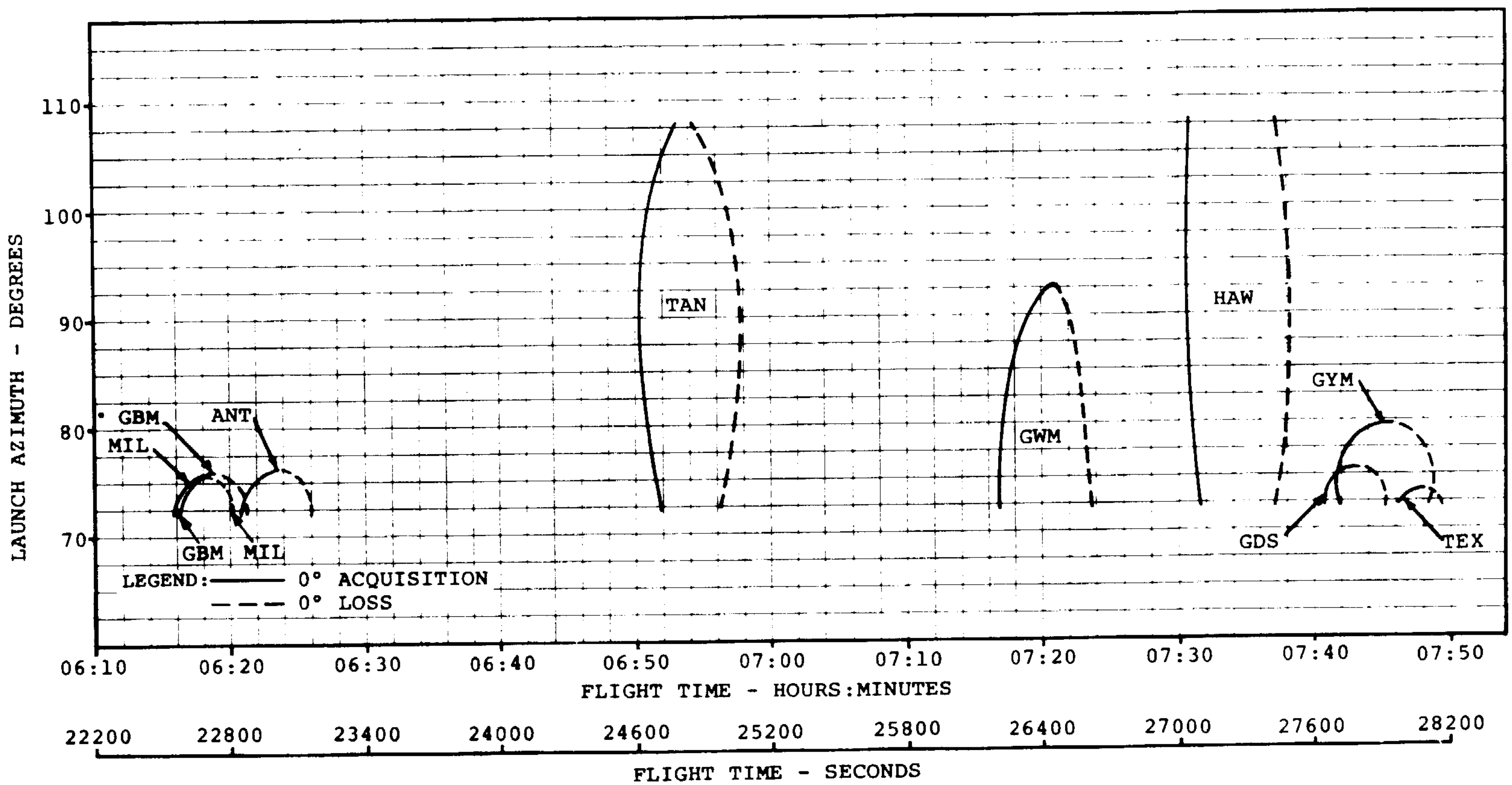


FIGURE 5-20 NOMINAL SURVEILLANCE ABOVE A 0-DEGREE ELEVATION ANGLE FOR FIFTH PARKING ORBIT (NO S-IVB SECOND BURN)

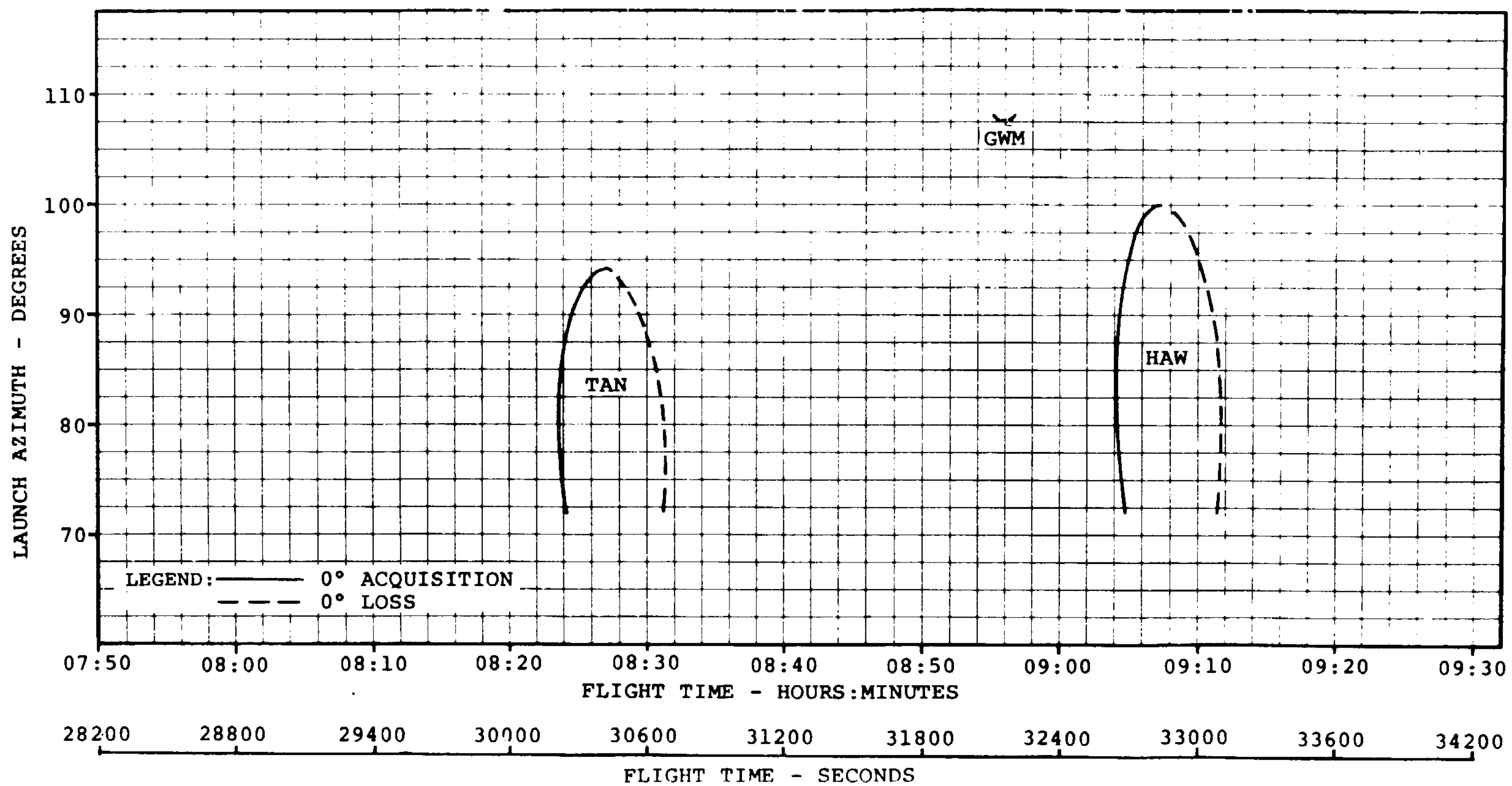
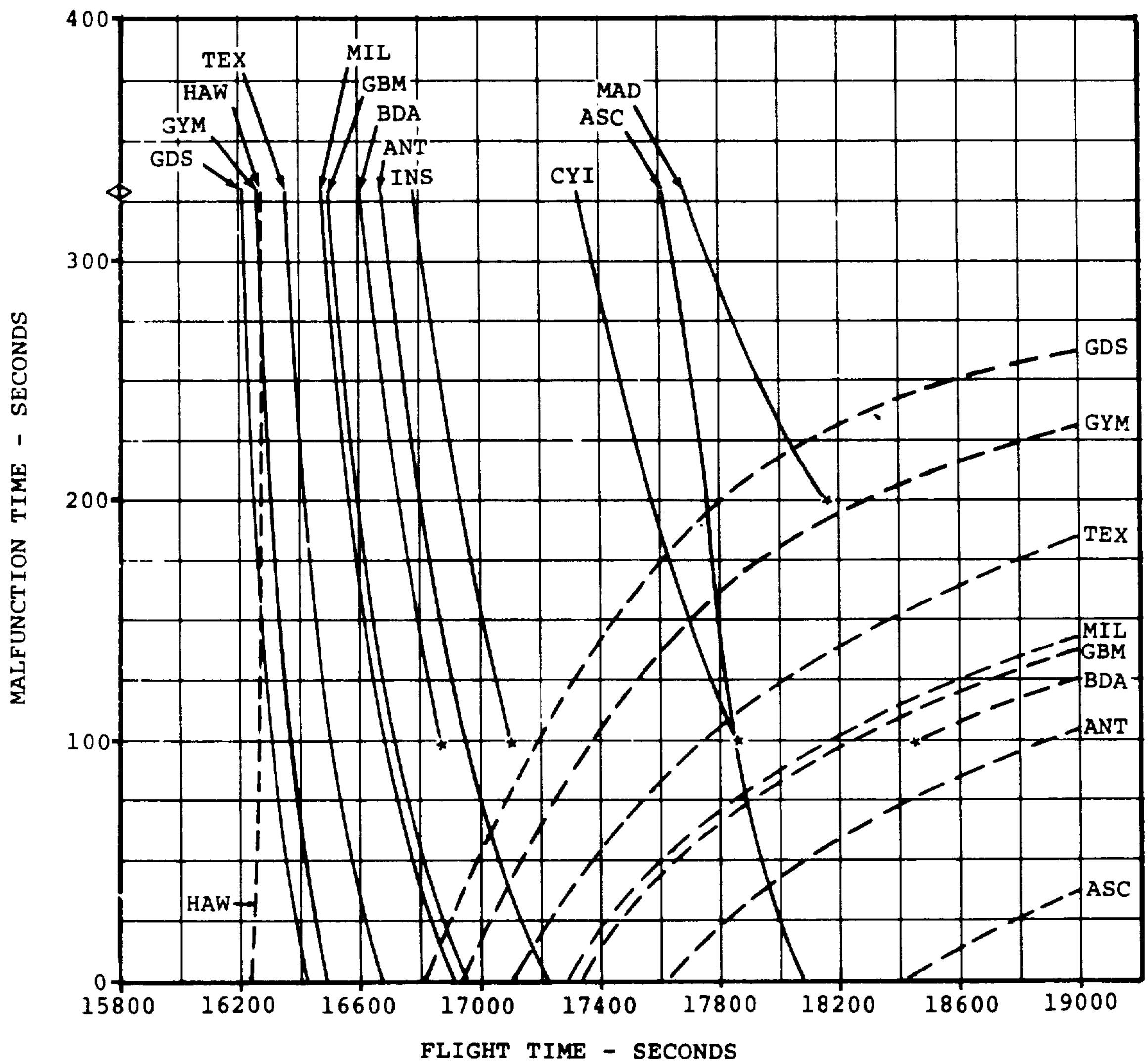


FIGURE 5-21 NOMINAL SURVEILLANCE ABOVE A 0-DEGREE ELEVATION ANGLE FOR SIXTH PARKING ORBIT (NO S-IVB SECOND BURN)



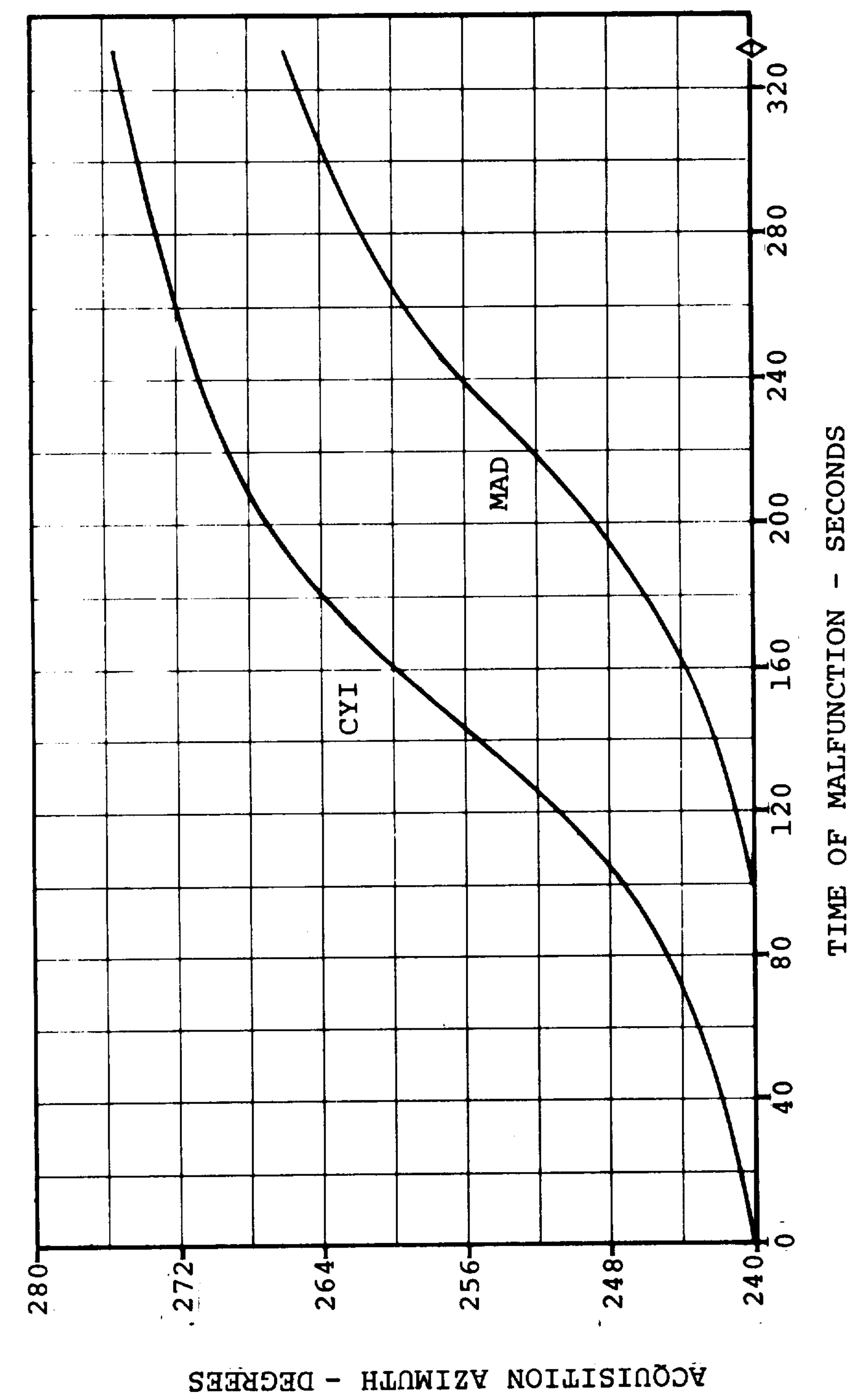
LEGEND:

——— ACQUISITION

– – – LOSS

NOTE: * DATA NOT AVAILABLE BEYOND THIS POINT.
◇ NOMINAL SURVEILLANCE ABOVE 0 DEGREE

FIGURE 5-22 S-IVB STAGE EARLY SECOND CUTOFF SURVEILLANCE ABOVE 0-DEGREE ELEVATION FOR 78.051-DEGREE LAUNCH AZIMUTH



NOTE: ◊ NOMINAL ACQUISITION AZIMUTH ABOVE 0 DEGREE

FIGURE 5-23 S-IVB STAGE EARLY SECOND CUTOFF ACQUISITION AZIMUTH ABOVE 0-DEGREE ELEVATION FOR 78.051-DEGREE LAUNCH AZIMUTH

TABLE 5-IV. SUMMARY OF THE 0-DEGREE SURVEILLANCE FOR A Y-ACCELEROMETER MALFUNCTION AT 2 SECONDS

LAUNCH AZIMUTH 78.051°	ORBIT											
	BOOST & FIRST			SECOND			THIRD			FOURTH		
STATION SYMBOL	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)	ACQUISITION (AZI DEG)	ACQUISITION (SEC)	LOSS (SEC)
MIL	28.3	0	533	278.1	5705	6143	279.7	11303	11746	262.9	16925	17303
GBM	314.1	66	562	286.7	5745	6173	286.0	11343	11786	267.3	16961	17353
BDA	257.4	242	760	265.9	5918	6344	253.6	11537	11884	X	X	X
ANT	344.0	514	639	332.6	6079	6358	310.0	11627	12050	277.8	17233	17642
INS	296.0	556	967	288.6	6146	6586	263.8	11771	12135	X	X	X
CYI	268.2	1006	1396	X	X	X	X	X	X	X	X	X
ASC	X	X	X	X	X	X	313.0	12453	12882	261.8	18091	18450
TAN	316.1	2176	2585	271.0	7769	8181	233.8	13443	13753	216.0	19098	19383
CRO	245.5	3122	3527	247.6	8715	9150	267.0	14324	14730	*	*	*
GWM	X	X	X	X	X	X	176.0	15130	15342	*	*	*
HSK	305.7	3633	3862	X	X	X	X	X	X	*	*	*
HAW	X	X	X	227.3	10221	10642	268.3	15818	16246	*	*	*
GDS	210.3	5307	5632	241.2	10840	11247	253.6	16423	16833	*	*	*
GYM	251.5	5296	5734	275.8	10896	11336	283.2	16502	16948	*	*	*
TEX	269.0	5476	5912	283.6	11081	11520	280.5	16683	17127	*	*	*

NOTE: X - THIS STATION DOES NOT TRACK.
AZI - STATION ACQUISITION AZIMUTH.
* - DATA NOT AVAILABLE