# FLIGHT MECHANICS PANEL DOCUMENT NO. 72 - FMP-2

# APOLLO LUNAR LANDING MISSION

# EMERGENCY DETECTION SYSTEM (EDS) AND FLIGHT DYNAMICS LIMITS



APOLLO 17, MISSION J-3 DECEMBER 6, 1972

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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MANNED SPACECRAFT CENTER HOUSTON, TEXAS EMERGENCY DETECTION SYSTEM (EDS)

AND FLIGHT MECHANICS LIMITS

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#### ABSTRACT

This document contains the MSFC/MSC agreements relative to the Emergency Detection System (EDS) and flight limits for the Apollo Lunar Landing Mission. The recommended EDS and Flight Dynamics Officer (FDO) flight limits are designed to provide, where possible, acceptable time for flight monitoring, emergency warning, abort decision and crew escape following an inflight malfunction from liftoff to separation of the Spacecraft from the S-IVB/IU.

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LIST (	OF ABBRI	EVIATIONS AND DEFINITIONS	
ΔP		Pressure differential	
Δt	=	Time differential	
$\Delta V$	=	Change in velocity	
α	=	Angle of attack	
β		Beta (The probability that a specific foil we int	
		cause mission and/or vehicle loss)	
γ	=	Inertial Flight Path Angle	
AOA	==	Angle of attack	
APL	=	Accelerometer Processing Loop	
APS	=	Auxiliary Propulsion System (S-TUP Change)	
BTO	=	Boost-to Orbit	
CECO	= '	Center engine cutoff	
CM	••••••	Command Module	
CN	=	Criticality number (the mission and/on mali 1 1	
		rate resulting from the failure of a second distance loss	
		expressed as the number of logges per will in )	
Cntrl	=	Loss of control	
COI	=	Contingency orbit insertion	
CSM		Command and Service Modules	
deg	=	degree	
DSKY	=	Display and keyboard	
EDS	-	Emergency Detection System	
ESS	-	Early S-TVB Staging	
FCC	-	Flight Control Comments	
FDAI	=	Flight Director Attitude Tell	
FDO	=	Flight Dynamics Office	
F/M	=	Thrust to many metric	
FMR	=	Flight Mission Pulsa	
FPS	=	Feet per second (ft/com)	
g	=	32.2 ft/sec <sup>2</sup>	
GET	=	Ground elapsed time	
IGM	=	Iterative guidance mode	
IGN	=	Ignition	
IMU	=	Inertial Measurement Unit	
IU		Instrument Unit	
LES	=	Launch Escape System	
LET	=	Launch Escape Tower	
LEV	=	Launch Escape Vehicle	
LH <sub>2</sub>	=	Liquid hydrogen	
lox	=	Liquid Oxygen	
L/V		Launch Vehicle	
LVDA	=	Launch Vehicle Data Adapter	
LVDC	=	Launch Vehicle Digital Computer	
MAX	=	Maximum	
MCC-H	=	Mission Control Center - Houston	
MCC	=	Mission completion capability (28,000 p mi apogoo	
-		with boost vehicle - (Minimum lunar orbit canability)	
Minimun	n Warnin	ng Time = (The minimum time available between	
		reaching an abort limit and vehicle break-up)	
MR	=	Mixture ratio	
MSFC	=	Marshall Space Flight Center	1
MS FN	=	Manned Spaceflight Network	1
NAR		North American Rockwell	

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# LIST OF ABBREVIATIONS AND DEFINITIONS (Concluded)

N.Mi.	=	Nautical miles
OECO	=	Outboard engine cutoff
Р	=	Probability of failure occurring in one flight times $10^6$ flights
POI	=	Parking orbit insertion
PSID	=	Pounds per square inch differential
PU	=	Propellant utilization
q	=	Dynamic pressure
RCS	=	Reaction Control System
RT	=	Reasonableness test
RTC	=	Reasonableness test constant
s/c	=	Spacecraft
Sec	=	Seconds
Sep	=	Loss of control after separation
SPS	=	Service Propulsion System
TB	=	Time base
TC	=	Tower collision
TD&E	=	Transposition, docking and extraction
<sup>t</sup> fail	=	Time of failure
<sup>T</sup> ff	=	Time of free fall
TLI	=	Translunar injection

1.1 10



#### I. SUMMARY

This document is prepared in compliance with the Flight Limits Sub-Panel agreements of the 15th Flight Mechanics Panel minutes for January 24, 1966. The scope of the document encompasses the crew safety ground rules and objectives, capability of the space vehicle, selected abort modes, rationale for the various flight limits, and the recommended limits.

The recommended flight limits include those for the EDS on board the space vehicle and the FDO trajectory plotboard displays in Mission Control Center - Houston. The limits are chosen so that acceptable time (if possible) is available for flight monitoring, emergency warning, abort decision, and crew escape following inflight malfunctions.

For the TLI phase, the EDS function is limited to lighting the overrate light. All aborts (S-IVB thrust termination) during this phase are initiated by the flight crew. Procedures and limits for their actions are defined herein. The procedure for performing the actual abort after it has been initiated is discussed only in a general manner, since it is not within the scope of this report or that of the Flight Limits Sub-Panel.

Changes between Apollo 16 (AS-511) and Apollo 17 (AS-512) will be indicated by a vertical line in the margin.

A list of references is provided which contains those documents in direct support of the analyses used to obtain and verify the various flight limits.

#### **II. OBJECTIVES**

The objectives to be met in establishing emergency detection system and flight dynamics limits lead to conflicting requirements in that the limits must maximize crew safety while minimizing false aborts. The objectives used in performing the necessary studies to establish abort limit settings were:

- Determine whether safe abort could be achieved from a malfunctioning vehicle for all significant overall failure modes.
- 2. Identify the sensors required and the limit settings to be used for each failure mode.
- 3. Determine improvements in software and/or hardware that would enhance crew safety.
- 4. Determine the malfunctions that require automatic abort and those for which manual abort may be used.
- 5. Determine whether the space vehicle structural integrity is maintained up to the time an abort is initiated.
- 6. Identify failures which may result in a trajectory from which the spacecraft cannot execute a safe return to earth in the event of an abort.

#### III. GROUND RULES

The ground rules used in performing the studies leading to recommended abort limit settings have resulted from numerous Intercenter Panel and Subpanel meetings over the past several years. Many ground rules are dictated by vehicle design constraints. While it is not the purpose of this report to list all the ground rules that might be pertinent to crew safety consideration and studies, the more important ground rules considered in establishing abort limits are listed below:

- 1. Crew abort action must be based on two valid and related abort cues.
- The onboard EDS senses, primarily, the effects of a malfunction; however, some displays provide system status such as engine out, platform failure, and S-II separation.
- 3. Abort will be manual rather than automatic where possible.
- 4. Ground monitoring will provide abort information for slow divergent failures.
- 5. A safe separation distance of the spacecraft from the launch vehicle will be provided before launch vehicle breakup and explosion, when possible. Explosion and launch vehicle breakup are assumed to occur simultaneously; however, in all probability a few seconds will exist.
- Spacecraft land impact is to be avoided, as far as is practical, for aborts in any flight phase.
- The launch escape vehicle capabilities are not to be exceeded for aborts in any flight phase.
- 8. When possible under malfunction conditions, flight will be continued in order to gain altitude and reduce the possibility of land impact.

The S/C displays were assumed to be unchangeable. The rationale and limit settings were developed to use combinations of the same onboard sensors to make the decision for abort. Terrestrial environmental conditions used were those specified for vehicle design (Reference 1). The launch escape vehicle constraints were obtained from Reference 2. The launch vehicle structural limits used in establishing emergency detection limit settings are found in References 3, 4, 5, and 6. The spacecraft structural limits are found in Reference 7.

Sensor accuracies also were considered in establishing recommended limit settings, since the accuracies are a function of the setting. The accuracies for the established settings are listed below.

Launch Vehicle Rate Light	Setting (Full Scale)	Accuracy
Pitch and Yaw Roll	4 deg/sec 20 deg/sec	+ 0.49 deg/sec + 1.9 deg/sec
Pitch and Yaw (after auto- matic abort deactivation)	9.2 deg/sec	- 2.0 deg/sec + 0.80 deg/sec

Angle of Attack Meter(read in %)

+ 0.06	PSID
	+ 0.06

Flight Director Attitude Indicator (FDAI)

Attitude Error	Setting (Full Scale)	Accuracy
Pitch and Yaw (fine)	<u>+</u> 5 deg	<u>+</u> 0.09 deg
Roll (Fine)	<u>+</u> 20 deg	<u>+</u> 0.36 deg
Pitch and Yaw (coarse)	<u>+</u> 15 deg	<u>+</u> 0.27 deg
Roll (coarse)	<u>+</u> 50 deg	+ 0.90 deg

Angular Rate

Pitch, Yaw, & Roll (fine)	+ 5 deg/sec	<u>+</u> 0.09 deg/sec
Pitch and Yaw (coarse)	<u>+</u> 10 deg/sec	<u>+</u> 0.18 deg/sec
Roll (coarse)	+ 50 deg/sec	+ 0.90 deg/sec

The above accuracies are found in Reference 8 for launch vehicle rate light, References 9 and 10 for the angle of attack meter, and Reference 11 for the FDAI.

The manual abort displays in the S/C are listed below. Sensors numbered 1 through 7 are used as indicators for abort action for the failure modes considered in this report. Sensors 8 through 13 are information cues.

- 1. <u>Abort request light</u> a red light, commanded "on" or "off" from the ground.
- Engine status lights a yellow light; "on" when engine is not operating (one per engine).
- 3. <u>Flight director attitude indicator</u> vehicle attitude, attitude error and attitude rates.
- 4. Launch vehicle overrate light a red light; "on" when rates are exceeded\*.
- 5. <u>Guidance failure light</u> a red light; "on" when attitude reference is lost\*.
- 6. <u>S-II separation light</u> a red light; "on" if S-II first plane separation occurs; "off" if second plane separation occurs\*\*.
- 7. <u>Angle-of-attack indicator</u> a meter which indicates the combination of angle-of-attack and dynamic pressure as measured by the Q-ball (read in %).
- 8. <u>G meter</u> longitudinal accelerometer.
- 9. Altimeter.
- 10. Digital event timer.
- 11. Caution and warning master alarm.
- 12. Liftoff/no automatic abort lights.
- 13. <u>S-IVB Propellant and Tank Pressures</u> ullage pressure in S-II and S-IVB fuel tanks and in S-IVB oxygen tank.

\* During period of automatic abort (0-120 seconds), loss of attitude reference will light both the guidance failure light and the launch vehicle overrate light.

\*\* This display is also used to indicate  $O_2H_2$  burner ignition for S-IVB restart preparation and again to indicate translunar injection commit.

## IV. DISCUSSION OF LIMIT SETTING CRITERIA

The limit setting criteria must maximize crew safety while minimizing false aborts for malfunctions.

#### A. On-Board Limit Settings

The occurrence of a subsystem malfunction during the flight of a Saturn V launch vehicle does not necessarily lead to a mission abort. Vehicle loss following a malfunction is contingent upon (1) the particular subsystem which malfunctions, (2) the manner in which the system malfunctions, (3) the time of flight at which the malfunction occurs, and (4) the external disturbances such as wind loads which the vehicle encounters. The requirement for crew safety dictates a relatively narrow range for the flight parameter abort limits in order to assure crew escape prior to tumbling and vehicle structural failure. Conversely, a wide range for the limits is desirable to prevent unnecessary abort from a vehicle which, even though malfunctioning, may survive and continue to achieve part or possibly all of the mission objectives. A major task in defining the flight parameter abort limits for the Saturn V was the determination of values which best satisfy the conflicting requirements mentioned above. However, since even the optimum limiting values were obtained by means of compromise, they must be accompanied by qualifying statements and logic to be effectively utilized as abort criteria.

The first task undertaken was to estimate, for various flight parameters, the extreme values which are associated with the trajectory of a non-malfunctioning vehicle. The envelopes of extreme values for normal flight established the minimum values for limits on each parameter utilized in abort criteria. If values for the limit levels are adopted which fall within the extreme values for a non-malfunctioning vehicle, then there is a measurable risk of aborting a good flight. The second major task was the generation of trajectories which include simulation of a subsystem malfunction. These trajectories were then examined to determine if and when unsafe aborts can occur, vehicle structural limits are exceeded, or uncontrolled vehicle will result. Also considered were the subsidiary effects such as violation of range safety corridors, unsafe conditions, and excessive performance degradation.

For the malfunctioning vehicle flights in which immediate or eventual abort is a requirement, the time histories of many flight parameters were examined to determine which provide the earliest and most definite indication of the associated vehicle malfunction. The flight parameter indications of a malfunction are obtained when the parameter value exceeds an extreme value boundary for non-malfunctioning vehicle flight. A limit level for the parameter is then selected at some value which lies between the extreme boundary value and the value at the time of hazardous conditions.

In selecting limits for the on-board displays, consideration was given to both the automatic abort and manual abort capability. Automatic abort is primarily for protection against those launch vehicle malfunctions for which there is insufficient crew reaction time, whereas the manual abort, using crew judgment, provides protection against malfunctions for which there is sufficient crew reaction time. For manual abort, the action must be based upon two (2) valid and related abort cues; therefore, the task was to attempt to select limits for at least two (2) parameters which the crew can use in an abort decision. The discussion and results of the flight mechanics simulation of the failure modes are given in Section VI.

All trajectories calculated for this study were generated using digital or hybrid computer routines which simulate rigid and flexible vehicle motion in three-dimensional space and in six degrees of freedom, with the operational trajectory data (Ref. 14) as baseline. The trajectory model includes features which are particularly important for malfunctioning vehicle studies, i.e., the simulation of the aerodynamic force and moment characteristics and the simulation of the vehicle attitude control system. The methods used for computing vehicle structural loads and the assumptions concerning the incorporation of winds in the trajectory simulations are also

The aerodynamic center of pressure location and the normal and axial aerodynamic force coefficients were computed as bivariant functions of both local angle of attack and local Mach number. The non-linearity of these aerodynamic parameters with angle of attack often becomes significantly large with a subsystem malfunctioning. The logic, equations, and numerical data which were used in this study to simulate the overall control system are representative, within the limitations of digital simulation, of the actual control system onboard the launch vehicle. Filter networks, internal limits, and engine actuator dynamics were included in the mathematical model to assure accurate simulation of the vehicle.

## B. Mission Control Center - Houston (MCC-H) Limit Settings

The following MCC-H limits were established to enhance crew safety during the launch powered flight phase:

- (1) Spacecraft structural limit
- (2) Spacecraft aerodynamic heating limit
- (3) Crew acceleration tolerance limit (16 g)
- (4) Free fall time limit

The first limit applies to the first phase of powered flight. The second limit is applicable to the first and second stages of powered flight. The last two limits apply during the second and third stages of powered flight. All of these limits are included in the mission rules.

As pointed out elsewhere in this report, the launch vehicle can reach flight conditions which will cause structural failure. An abort limit line for launch vehicle structural failures that can be monitored by the FDO has been defined and reported by MSC in Reference 15. This limit was determined by integrating simulated platform malfunctions that result in a relatively slow change in vehicle attitude, velocity, attitude rate, angle of attack, etc. The limit is constructed such that a vehicle which exceeds the limit will have a structural failure. However, it is entirely possible to have a structural failure prior to exceeding the limit. The limit is expressed in terms of inertial velocity and flight path angle and is biased to account for wind effects and data system delay times.

The spacecraft aerodynamic heating limit is based on both structural and systems performance considerations. Any launch vehicle malfunction which results in a depressed trajectory may subject the spacecraft to aerodynamic heating beyond that for which it was designed. In this case the spacecraft will either fail structurally or system performance will be degraded below that which is acceptable for orbital operation. This boundary is shown in Figure 1 (page 10) and is biased for reaction and data system delays. A complete description of this limit and its limitations are given in Reference 17. The major qualification of the limit is that it is only valid for slow trajectory deviations resulting from platform drifts, and the limit is not valid for particular L/V malfunctions such as engine out or low thrust, where the crew and the ground are both aware of a specific anomaly. The heating limit, shown in Figure 1, will be violated prior to exceeding the structural limit. The crew acceleration tolerance limit is a limit beyond which the crew and possibly the spacecraft would be subjected to unacceptable deceleration forces during an entry following an abort. This boundary is far removed from the predicted launch vehicle 3-sigma dispersion and from trajectories for predictable launch vehicle failures. Thus, this boundary is used to decide when to terminate the launch vehicle thrust for those cases where the vehicle is diverging from the nominal and will not achieve acceptable cutoff conditions, even if allowed to continue. The limit, in terms of inertial flight path and velocity, is based on nominal launch vehicle altitude, geographic location and azimuth, nominal atmosphere and predicted spacecraft launch abort aerodynamics for full lift entry. The crew acceleration tolerance limit has been set at 16 g. Altitude, aerodynamic and atmosphere effects result in entry deceleration dispersions of about  $\pm 0.5$  g.

The free fall time limit is a boundary beyond which the crew would have insufficient time to perform the required separation sequencing and spacecraft reorientation prior to atmospheric entry. This boundary is far removed from the predictable launch vehicle 3-sigma dispersion and from the trajectories for predictable launch vehicle failures. Thus, this boundary is used to terminate the launch vehicle thrust for those cases where the vehicle is diverging from the nominal and will not achieve acceptable cutoff conditions, even if allowed to continue. The limit in terms of inertial flight path angle and velocity is based on nominal launch vehicle altitude. Because of the sensitivity of this limit to altitude, in actual practice the abort decision is made from a display of instantaneously computed free fall time. The limit is set at 100

The use of early S-IVB staging, although not strictly an abort mode, is mentioned here to point out the ground flight control limits for this event. Early S-IVB staging is used to attempt an alternate mission rather than an abort. The MCC-H limits for early staging are based on the performance capability of the S-IVB to achieve an acceptable orbit. Early staging is accomplished through the implementation of mission rules and is not shown as an abort limit on the FDO plot board. These limits, expressed in terms of inertial flight path angle and velocity, are shown in Figure 1. This figure presents two different capability envelopes. One shows the orbit capability boundaries for a malfunctioning launch vehicle and the other gives the mission capability limits for the L/V, S/C and crew. The boundaries are the maximum envelopes of all malfunctioned flights which make a minimum 70 N.Mi. perigee orbit; they serve as a warning cue that abort may be required. The limit lines serve as the actual abort cue. Figure 2 (page 11) summarizes S/C near-insertion abort capability for suborbital aborts. S/C contingency orbit insertion (COI) capability is summarized in Figure 3 (page 12).



INERTIAL VELOCITY, FPS

Figure 1. LAUNCH ABORT AND CAPABILITY LIMITS



1

Figure 2. NEAR-INSERTION ABORT CAPABILITY-SUBORBITAL ABORT

11

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x



Figure 3. CONTINGENCY ORBIT INSERTION

## V. FAILURE MODES IDENTIFICATION

The failure modes considered and the interpretation of these failure modes as used in the simulations supporting this document are presented in Table 1 (pages 14 and 15). The probability of occurrence and the criticality numbers are given for each failure mode in Table 2 (pages 16 through 18; references 25 and 26). A schematic of Saturn V guidance and control failure points is shown in Figure 4 (page 19).

Particular emphasis was placed upon failure during periods of flight when the occurrence of a malfunction could lead immediately to a hazardous condition for the crew. The identified failures are believed to include all possible adverse effects on flight conditions due to single-point failures which affect vehicle dynamics. The analysis does not include structural failures which could occur within the abort limits, e.g., stress corrosions, over pressurizations, defective workmanship, etc.

#### TABLE 1

### FAILURE MODE DEFINITIONS

- 1. Loss of thrust, one engine Self explanatory.
- 2. Loss of thrust, two engines Self explanatory.
- 3. <u>One actuator hardover</u> A control system failure resulting in a fully extended or fully retracted engine actuator.
- 4. <u>One actuator to null</u> Actuator servo system failure resulting in a null actuator position.
- 5. <u>Saturated error/rate signal</u> Failure in the launch vehicle control channel resulting in all control engines being commanded to the fully deflected position.
- 6. Loss of inertial attitude Platform gimbal angle failure of Launch Vehicle Digital Computer (LVDC) reasonableness test resulting in use of previous good value for inertial attitude. The reasonableness test consists of the launch vehicle platform being interrogated every 40 milliseconds for the correct attitude. If an excessive attitude rate discrepancy (equivalent to 10 deg/sec) is found during consecutive checks on the fine resolvers, the test is switched to the coarse resolver. If excessive attitude rate discrepancy (equivalent to 32.5 deg/sec) is present for 12 times in 20 checks, the reasonableness test has failed. If the reasonableness test fails, the L/V guidance light in the CM is activated\* and spacecraft takeover is initiated by the crew.
- 7. <u>Erroneous attitude error signal</u> Any failure that causes a false attitude error signal between zero and saturated to the flight control computer.
- 8. Loss of attitude rate signal Any failure that causes a false vehicle rate indication of zero degrees per second to the flight control computer.
- 9. Loss of inertial velocity Failure of a platform accelerometer to pass reasonableness or zero test resulting in use of preloaded acceleration data. If, at some time later, the reasonableness or zero test is passed, the platform accelerometers will be used.

<sup>\*</sup> Prior to deactivation of the automatic abort modes at 120 seconds, the L/V overrate light will also be lighted.

### TABLE I (Concluded)

#### FAILURE MODE DEFINITIONS

- 10. Loss of Flight Control Computer Switch Points Any malfunction that causes early, late or no-gain switching
- 11. Propellant utilization (PU) failure The S-II and S-IVB propellant utilization (PU) system malfunctions are those in which the PU valve fails to follow the flight plan.
- 12. Loss of one APS module inability to fire any of the three engines on one of the APS modules.
- 13. Loss of both APS modules Inability to fire any of the three engines on both APS modules.
- 14. <u>Staging and sequencing malfunction</u> Failures which cause premature, lack of, late, or complications during, staging or sequencing.

## TABLE 2

# PROBABILITY AND CRITICALITY OF SATURN V VEHICLE FAILURE MODES

Failure Mode	S-IC Burn		S-II Burn				
	Р	β	CN	P	β	CN	
1. Loss of thrust, one engine	643	0.21	135	10,757	<0.00001	<1	
2. Loss of thrust, two engines	<1	0.90	<1	33	0.61	20	
3. One actuator hardover	466	0.15	70	307*	0.97	299	
4. One actuator to null	159	<0.001	<1	154	<0.001	<1	
5. Saturated error signal	37	1.00	37	35	1.00	35	
6. Saturated rate signal	18	1.00	18	18	1.00	18	
7. Loss of inertial attitude	1,650	0.08**	132	1,028	0.08**	82	
8. Erroneous attitude error signal	135	1.00	135	168	1.00	168	
9. Loss of attitude rate signal	22	1.00	22	60	1.00	60	
10. Loss of inertial velocity	1,030	0.01	10	980	0.01	10	
ll. Loss of FCC switch points	48	0.2	10	75	<0.001	<1	
12. Propellant utili- zation failure		DNA		934	0.23	215	
13. Loss of one APS module		DNA			DNA		
14. Loss of both APS modules		DNA			DNA		
15. Staging and sequenc- ing malfunction	565	1.00	565	128	1.00	128	

P - probability of the failure occurring in one flight (times  $10^6$ )

 $\beta\,$  - probability of mission loss if failure occurs

CN - criticality number, product of P and  $\beta$ 

DNA - does not apply to that portion of flight

\* Hardover inboard only

\*\* does not include failure of the backup S/C system

# TABLE 2 (Continued)

# PROBABILITY AND CRITICALITY OF SATURN V VEHICLE FAILURE MODES

		1						
	Failure Mode	S-IVB First Burn			S-IVB First Coast			
ļ		Р	β	CN	Р	β	CN	
1. 2.	Loss of thrust, one engine Loss of thrust	8,891	1.00	8,891		DNA		
3.	two engines One actuator		DNA			DNA		
4.	hardover One actuator to	5	1.00	5		DNA		
5.	null Saturated error	244	1.00	244		DNA		
6.	signal Saturated rate	<1	1.00	<1	< 1	1.00	< 1	
7.	signal Loss of inertial	4	1.00	<1	<1	1.00	< 1	
8.	attitude Erroneous attitude	602	0.08	48	356	0.08*	28	
9.	error signal Loss of attitude	9	1.00	9	1,705	1.00	1,705	
10.	rate signal Loss of inertial	< 1	0.67	< 1	< 1	0.67	< 1	
11.	velocity Loss of FCC	377	0.01	4	267	0.001	< 1	
12.	switch points Propellant utili-		DNA			DNA		
13.	zation failure Loss of one APS	616	<0.001	<1		DNA		
14.	module Loss of both APS	N/A			2995	**		
15.	modules Staging/seq. malf.	<1	** DNA		446	** DNA		

P - probability of failure occurring in one flight (times 10<sup>6</sup>)

 $\beta\,$  - probability of mission loss if failure occurs

CN - criticality number, product of  ${\bf P}$  and  $\beta$ 

DNA - does not apply to that portion of flight

N/A - not available

\*Does not include failure of the backup S/C system

\*\*Dependent on Manual RCS takeover

## TABLE 2 (Concluded)

# PROBABILITY AND CRITICALITY OF SATURN V VEHICLE FAILURE MODES

Failure Mode	S-IVB Second Burn			S-IVB Second Coa		nd Coast	
	Р	β	CN	P	ß	CN	-
1. Loss of thrust, one engine	6,199	0.90	5,579		DNA		North Constants
2. Loss of thrust, two engines		DNA			DNA		
3. One actuator hardover	3	0.9	3		DNA		
4. One actuator to null	171	0.87	149		DNA		
5. Saturated error signal	<1	0.82	<1	< 1	*		
6. Saturated rate signal	<1	0.95	<1	<1	*		
7. Loss of inertial attitude	1,420	0.08**	114	269	*		
<ol> <li>Erroneous atti- tude error signal</li> </ol>	106	0.58	61	62	0.001	<1	
9. Loss of attitude rate signal	<1	0.58	<1	<1	0.001	<1	
<ol> <li>Loss of inertial velocity</li> </ol>	1,100	0.0001	<1	203	0.001	<1	
ll. Loss of FCC switch points	<1	0.001	<1		DNA		
12. Propellant utili- zation failure	1,108	0.0001	<1		DNA		
13. Loss of one APS module	N/A	stastasta		1,016	***		
14. Loss of both APS modules		410 410 475		1,601	***		
15. Staging/seq. malf.		DNA		53	1.0	53	

P - probability of failure occurring in one flight (times  $10^6$ )

 $\boldsymbol{\beta}$  - probability of mission loss if failure occurs

CN - criticality number, product of P and  $\beta$ 

DNA - does not apply to that portion of flight

N/A - not available

\* Dependent on crew takeover

\*\* Does not include failure of the backup S/C system

\*\*\* Dependent on Manual RCS takeover



Figure 4. SATURN V GUIDANCE AND CONTROL FAILURE POINTS

#### VI. RESULTS AND DISCUSSION

#### A. Malfunction Summary

In the analysis, mission completion criteria for AS-512 were considered satisfied if the orbit achieved at S-IVB second burn cutoff has an apogee of at least 28,000 ni. mi.; from this orbit, the capability exists for achieving a minimum lunar orbit. A summary of malfunction capability for all types of malfunctions in all stages is shown in Figures 5 through 11 (pages 21 through 32; reference 26). Not all malfunctions occurring in the indicated time span will cause vehicle loss due to the capability of the vehicle to sustain safe flight despite most malfunctions.

The following vehicle loss cases were considered:

- (a) Tower collision or pad fallback because of early failures. Included is interference with ground support equipment at liftoff.
- (b) Loss of control followed by consequent tumbling and possible structural failure.
- (c) Structural failure because vehicle tension or compression load is increased by engine shutdown at abort.
- (d) Structural failure resulting directly from malfunction dynamics.
- (e) Unsuccessful staging. This results when the malfunction causes the vehicle to exceed staging limits.
- (f) Trajectory deviations due to guidance logic interactions with malfunction modes.

Malfunction conditions leading to possible crew loss are summarized in Figures 5 and 6 (pages 21 through 24). Crew losses may occur if the launch vehicle explodes before the launch escape vehicle (LEV) achieves a safe separation distance by abort action, or if the LEV angle-of-attack (LEV- $\alpha$ ) limit is exceeded prior to abort, making safe abort improbable.



\*THROUGHOUT S-IC FLIGHT

FIGURE 5. SUMMARY OF MALFUNCTIONS REQUIRING NEAR-PAD ABORT (SHEET 1 OF 2)

: 2]



\*THROUGHOUT S-IC FLIGHT

. 22

FIGURE 5. SUMMARY OF MALFUNCTIONS REQUIRING NEAR-PAD ABORT (SHEET 2 OF 2)

		TIME - SECONDS
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	LIFTOFF MAX Q S-IC/S-II 50 83 100 163
LOSS OF THRUST ONE ENGINE* (643)	CREW (0.03, 19) VEHICLE (0.03, 19) MISSION (0.21, 135)	8     988     1       2     986     1       3888     3888     1
ONE ACTUATOR HARDOVER (466)	CREW (0.06, 28) VEHICLE (0.15, 70) MISSION (0.15, 70)	۲۰۰۵ ۲۰۰۵ ۲۰۰۵ ۲۰۰۵ ۲۰۰۵ ۲۰۰۵ ۲۰۰۵ ۲۰۰۵
SATURATED ERROR SIGNAL (37)	CREW (0.07, 3) VEHICLE (1.0, 37) MISSION (1.0, 37)	X 2000000000000000000000000000000000000
SATURATED RATE SIGNAL (18)	CREW (0.01, <1) VEHICLE (1.0, 18) MISSION (1.0, 18)	
LOSS OF INERTIAL ATTITUDE (1650)	CREW (0.02, 33) VEHICLE (0.08, 132) MISSION (0.08, 132)	
LOSS OF THRUST TWO ENGINES (<1)	CREW (0.01, <1) VEHICLE (0.90, <1) MISSION (0.90, <1)	8888 8888888888 8888888888 8888888888

**BRANCE** POSSIBLE LOSS

NO LOSS

\*Does not include catastrophic failures (criticality = 61).

FIGURE 6. S-IC MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	LIFTOFF MAX Q S-IC/S-II 50 83 100 163
ONE ACTUATOR INOPERATIVE (159)	VEHICLE (0, <1) MISSION (0, <1)	
LOSS OF FCC SWITCH POINTS AND NULL OFFSET (48)	VEHICLE (0.2, 10) MISSION (0.2, 10)	XXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXX
ERRONEOUS ATTITUDE ERROR (135)	VEHICLE (0.67, 90) MISSION (1.0, 135)	xxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxxx
LOSS OF ATTITUDE RATE (22)	VEHICLE (1.0, 22) MISSION (1.0, 22)	
ACCELEROMETER FAILURE (1030)	VEHICLE (0, <1) MISSION (0.01, 10)	××××××××××××××××××××××××××××××××××××××
STAGING MALFUNCTIONS (565)	VEHICLE (1.0, 565) MISSION (1.0, 565)	

FIGURE 6. S-IC MALFUNCTION SUMMARY (SHEET 2 OF 2)

LOSS

**POSSIBLE LOSS** 

24

H NO LOSS

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	S-IC/S-II 100 200 300 S-II/S-IVB 398
ONE ACTUATOR HARDOVER INBOARD (307)	CREW (0.97, 299) VEHICLE (0.97, 299) MISSION (0.97, 299)	
LOSS OF THRUST ONE ENGINE (10757)	VEHICLE (0, <1) MISSION (0, <1)	
LOSS OF THRUST TWO ENGINES (33)	VEHICLE (0.31, 10) MISSION (0.61, 20)	
ONE ACTUATOR INOPERATIVE (154)	VEHICLE (0, <1) MISSION (0, <1)	   
SATURATED ERROR SIGNAL (35)	VEHICLE (1.0, 35) MISSION (1.0, 35)	
SATURATED RATE SIGNAL (18)	VEHICLE (1.0, 18) MISSION (1.0, 18)	
LOSS OF FCC SWITCH POINTS AND NULL OFFSET (75)	VEHICLE (0, <1) MISSION (0, <1)	
ERRONEOUS ATTITUDE ERROR (168)	VEHICLE (0.67, 113) MISSION (1.0, 168)	***************************************
LOSS OF ATTITUDE RATE (60)	VEHICLE (1.0, 60) MISSION (1.0, 60)	

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4

**POSSIBLE LOSS** 

NO LOSS

.

١

FIGURE 7. S-II MALFUNCTION SUMMARY (SHEET 1 OF 2)

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	S-IC/S-II 100 200 300 398
ACCELEROMETER FAILURE (980)	VEHICLE (0, <1) MISSION (0.01, 10)	××××××××××××××××××××××××××××××××××××××
LOSS OF INERTIAL ATTITUDE (1028)	VEHICLE (0.08, 82) MISSION (0.08, 82)	20000000000000000000000000000000000000
PU SYSTEM MALFUNC- TIONS (934)	VEHICLE (0, <1) MISSION (0.23, 215)	H
SEQUENCING/STAGING MALFUNCTIONS (128)	VEHICLE (1.0, 128) MISSION (1.0, 128)	

**XXXX** POSSIBLE LOSS

H NO LOSS

# FIGURE 7. S-II MALFUNCTION SUMMARY (SHEET 2 OF 2)

× • • •

		TIME - SECONDS
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	S-II/S-IVB TB5 50 100 150
LOSS OF THRUST ONE ENGINE (8891)	VEHICLE (1.0, 8891) MISSION (1.0, 8891)	
ONE ACTUATOR HARDOVER (5)	VEHICLE (1.0, 5) MISSION (1.0, 5)	
ONE ACTUATOR INOPERATIVE (244)	VEHICLE (0.79, 193) MISSION (1.0, 244)	
SATURATED ERROR SIGNAL (<1)	VEHICLE (1.0, <1) MISSION (1.0, <1)	
SATURATED RATE SIGNAL (<1)	VEHICLE (1.0, <1) MISSION (1.0, <1)	
ERRONEOUS ATTITUDE ERROR (9)	VEHICLE (0.28, 3) MISSION (1.0, 9)	20222222222
LOSS OF ATTITUDE RATE (<1)	VEHICLE (0.37, <1) MISSION (0.67, <1)	
LOSS OF INERTIAL ATTITUDE (602)	VEHICLE (0.08, 48) MISSION (0.08, 48)	10000000000000000000000000000000000000

Reserved POSSIBLE LOSS

HINO LOSS

FIGURE 8. S-IVB 1ST BURN MALFUNCTION SUMMARY (SHEET 1 OF 2)
		TIME - SECONDS
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	S-II/S-IVB TB5 50 100 150
ACCELEROMETER FAILURE (377)	VEHICLE (0, <1) MISSION (0.01, 4)	255255555555555555555555555555555555555
PU SYSTEM MALFUNC- TIONS (616)	VEHICLE (0, <1) MISSION (0, <1)	
LOSS OF ONE APS MODULE (NOT AVAILABLE)	VEHICLE (0, <1) MISSION*	
LOSS OF BOTH APS MODULES (<1)	VEHICLE (0, <1) MISSION*	
LOSS	XXXXXX POSSI	BLE LOSS H- NO LOSS

\*Dependent on manual RCS takeover.

FIGURE 8. S-IVB 1ST BURN MALFUNCTION SUMMARY (SHEET 2 OF 2)

			TIME -	SECONDS	
FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	TB5	3,000	6,000	REIGNITION 11369
SATURATED ERROR SIGNAL (<1)	MISSION (1.0, <1)				
SATURATED RATE SIGNAL (<1)	MISSION (1.0, <1)				
ERRONEOUS ATTITUDE ERROR (1705)	MISSION (1.0, 1705)			$\mathbf{t}_{\mathbf{t}_{i}}^{+} := (1, \dots, 2^{n}) : \mathbf{d}_{i}^{+} = (\mathbf{g})^{n}$	
LOSS OF ATTITUDE RATE (<1)	MISSION (0.67, <1)	8888888	100000000000000000000000000000000000000	2022222222222222	989985
LOSS OF INERTIAL ATTITUDE (356)	MISSION (0.08, 28)	8888888	202222222222222222222222222222222222222	2003299933999	88888
ACCELEROMETER FAILURE (267)	MISSION (0, <1)	F			
LOSS OF ONE APS MODULE (2995)	MISSION*	5888886	888888888888888888888888888888888888888		88885
LOSS OF BOTH APS MODULES (446)	MISSION*	5333335	899899999999999999	888888888888888888888888888888888888888	22222

LOSS

SOCOS POSSIBLE LOSS

NO LOSS

\*Dependent on manual RCS takeover.

FIGURE 9. S-IVB PARKING ORBIT COAST MALFUNCTION SUMMARY

FAILURE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	TIME - SECONDS         *MC 309           REIGNITION         100         200         300         345
LOSS OF THRUST ONE ENGINE (6199)	MISSION (0.90, 5579)	
ONE ACTUATOR HARDOVER (3)	MISSION (0.9, 3)	
ONE ACTUATOR INOPERATIVE (171)	MISSION (0.87, 149)	€
SATURATED ERROR SIGNAL (<1)	MISSION (0.82. <1)	
SATURATED RATE SIGNAL (<1)	MISSION (0.95, <1)	
LOSS OF FCC SWITCH POINTS AND NULL OFFSET (<1)	MISSION (0, <1)	1
ERRONEOUS ATTITUDE ERROR (106)	MISSION (0.58, 61)	\$
LOSS OF ATTITUDE RATE (<1)	MISSION (0.58, <1)	10000000000000000000000000000000000000
LOSS OF INERTIAL ATTITUDE (1420)	MISSION (0.08, 114)	\$24000000000000000000000000000000000000
ACCELEROMETER FAILURE (1100)	MISSION (0, <1)	

\*MISSION COMPLETION CAPABILITY (MC)

LOSS

**POSSIBLE LOSS** 

H NO LOSS

FIGURE 10. S-IVB 2ND BURN MALFUNCTION SUMMARY (SHEET 1 OF 2)

\*MC TIME - SECONDS 309 FAILURE MODE TYPE LOSS (LOSS REIGNITION TLI (UNRELIABILITY) FACTOR, CRITICALITY) 100 200 300 345 . 1 PU SYSTEM MALFUNC-MISSION (0, <1)TIONS (1108) LOSS OF ONE APS MISSION\*\* MODULE (NOT AVAILABLE) LOSS OF BOTH APS MISSION\*\* MODULES (<1) \*MISSION COMPLETION CAPABILITY (MC)

# FIGURE 10. S-IVB 2ND BURN MALFUNCTION SUMMARY (SHEET 2 OF 2)

31

LOSS \*\* Dependent on manual RCS takeover.

**RECONS** POSSIBLE LOSS

NO LOSS

			TIME -	SECONDS	
FAILORE MODE (UNRELIABILITY)	TYPE LOSS (LOSS FACTOR, CRITICALITY)	TL I	500	1000	S-IVB/CSM
SATURATED ERROR SIGNAL (<1)	MISSION*	\$	*****	\$\$\$\$\$\$\$\$\$\$	\$8888
SATURATED RATE SIGNAL (<1)	MISSION*	8	****	200000000000000000000000000000000000000	\$\$\$\$\$
ERRONEOUS ATTITUDE ERROR (62)	MISSION (0, <1)	F			
LOSS OF ATTITUDE RATE (<1)	MISSION (0, <1)	⊢		-	
LOSS OF INERTIAL ATTITUDE (269)	MISSION*	88	\$\$\$\$\$\$\$\$\$\$\$\$\$	**************	88885
ACCELEROMETER FAILURE (203)	MISSION (0, <1)	F	-		
LOSS OF ONE APS MODULE (1016)	MISSION*	585		*****	88888
LOSS OF BOTH APS MODULES (1601)	MISSION*	8	*****	******	22222
SEQUENCING/STAGING MALFUNCTIONS (53)	MISSION (1.0, 53)				

\*Dependent on crew takeover.

LOSS

CONSTITUTE AND A POSSIBLE LOSS

NO LOSS

## FIGURE 11. S-IVB TRANSLUNAR COAST MALFUNCTION SUMMARY

## B. Contingency Procedures

The early liftoff phase and the maximum dynamic pressure region have been recognized as posing special problems. Satisfactory cues to enable the crew to abort prior to tower collision do not exist. Satisfactory separation distance and time is not available for all failure conditions, <u>assuming</u> that explosion occurs with launch vehicle structural failure. Certain individual launch escape vehicle limits (see Reference 24) may be exceeded during early flight phase for some launch vehicle malfunctions; however, flight will be continued as long as possible in order to gain altitude and reduce the possibility of land impact (if no structural limit was exceeded, and the launch vehicle was still con-

Launch Phase - A series of abort modes is required to assure crew safety throughout the launch phase. These abort modes are designed to utilize available spacecraft systems. The abort modes and associated crew procedures are defined in detail in References 12, 13 and 16. Briefly, the modes are as follows:

- MODE I (z-42:00 <GET < 3:18.8 min:sec) Utilizes the Launch Escape Vehicle (LEV) to provide abort capability during the most critical portions of powered flight, i.e., pad, near-pad emergency and high dynamic pressure region.
- MODE II (3:18.8<GET<10:24 min:sec) Characterized by S/C separation from L/V, no SPS maneuver and a full-lift entry with landing in the continuous recovery area.
- MODE III (10:24<GET<11:54 min:sec) Characterized by S/C separation from the S-IVB and an SPS maneuver performed to achieve landing in the discrete recovery area in the Atlantic Ocean. Used only if a contingency orbital insertion is undesirable.
- MODE IV (9:17.4 GET < Insertion) Contingency orbital insertion is characterized by S/C separation from the L/V and SPS maneuver to raise perigee altitude to at least 70 n.mi. A possible variation of Mode IV is the "apogee kick" in which the SPS maneuver is delayed until near apogee of the trajectory.

The areas of applicability in terms of inertial velocity and flight path angle are presented in Figures 1 through 3 (pages 10 through 12). These displays assume a nominal altitude profile. Similar charts are used for off-nominal altitudes. <u>Translunar Injection</u> - During translunar injection, the primary objective after a malfunction develops (next to crew safety) is to perform an alternate mission. Therefore, the extent of allowable deviated flight conditions must be determined in advance to ensure that the desired alternate mission capability will exist. Also, due consideration must be given to the provision of reasonable initial conditions for performance of an abort maneuver. These requirements were fulfilled by the development of a crew monitoring procedure which includes appropriate S-IVB shutdown limits.

The crew must be able to monitor and evaluate TLI without ground support becuase the S-IVB second burn can occur off the MSFN tracking range. TLI will occur over the Atlantic Ocean. A schematic of the basic crew monitoring technique (see Figure 12, page 37), shows that an abort can be performed for attitude rate problems, for attitude deviations problems, and for S/C system problems. Because S-IVB failures normally would result in an alternate mission, only a critical S/C system problem is likely to require an abort.

Several significant items can be noted about the TLI monitoring techniques:

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

2. The S-IVB engine will be cut off by the crew for pitch or yaw rates of 10 deg/sec or greater and 20 deg/sec or greater in roll body rate.

3. The S-IVB engine will be cut off by the crew with the abort handle for attitude deviations of 45° from the nominal attitude.

4. A backup to the S-IVB guidance cutoff signal will be performed by the crew two (2) seconds after the S/C displayed inertial velocity has reached the MSFC predicted cutoff velocity. As further verification that the predicted cutoff velocity has been achieved, a pre-burn up-date of the S/C platform and computer is made and the crew has additional information available such as the EMS  $\Delta V$  counter, pad burntime, and crew charts (Reference 31).

The crew is provided with preflight tables of attitude and DSKY parameter values at discrete times during the TLI burn. These tables cover both the nominal and manual takeover cases, first and second injection opportunities, for the prime day and launch azimuth. Samples of these tables are shown on Pages 35 and 36. The crew will normally monitor the progress of TLI by comparing the actual spacecraft attitudes and DSKY parameter values to the values listed on the chart. By mentally adding  $\pm$  45° to the attitude profile, the crew can establish the attitude excursion limits.



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	TABLE	3				
TYPICAL CREW	CHARTS FOR MAI	NUAL CONTROL	OF	THE	TLI	BURN

\*

.

S-IVB TLI - NOMINAL								S-IVB T	LI - MAN	UAL	
FIRST OPPORTUNITY								FIRST O	PPORTU	NITY	
θ	Ψ	DET	VI	Ĥ	Н	θ	Ψ	DET	VI	Ĥ	н
70.6	0	0:00	25554	9	107	66.6	4.5	0:00	25554	9	107
65	0	0:30	26100	-4	107	65	4.5	0:30	26100	-4	107
64	1.5	1:00	26700	19	107	64	4.5	1:00	26700	19	107
63	2.0	1:30	27380	90	107	63	4.5	1:30	27380	90	107
62	3.0	2:00	28100	210	108	62	4.5	2:00	28100	210	108
61	4.0	2:30	28880	425	110	61	4.5	2:30	28880	425	110
60	4.5	3:00	29700	725	112	<sup>59</sup> <b>८</b>	4.5	3:00	29700	725	112
59	5.5	3:30	30600	1100	117	Part	4.5	3:30	30600	1100	117
57	6.5	4:00	31600	1580	104	57	4.5	4:00	31600	1580	124
56	7.0	4:30	32600	2200	133	56	4.5	4:30	32600	2200	133
54	8.0	5:00	33650	2910	145	54	4.5	5:00	33650	2910	145
50	9.0	5:30	34800	3750	162	53	4.5	5:30	34800	3750	162
50	9.0	5:48	35563	4285	174	52	4.5	5:48	35563	4285	174

S-IVB TLI - NOMINAL								S-IVB T	LI - MAN	UAL	
SECOND OPPORTUNITY							S	SECOND (	OPPORTU	JNITY	
θ	Ψ	DET	v <sub>I</sub>	н	Н	θ	Ψ	DET	v <sub>I</sub>	н́	н
70	1.0	0:00	25546	6	109	69	1.0	0:00	25546	6	109
64	2.0	0:32	26228	-5	109	63	2.0	0:32	26228	-5	109
62	2.0	1:02	26945	+16	109	61	2.0	1:02	26945	+16	109
61	2.0	1:32	27698	101	109 6	60	2.0	1:32	27698	101	109
60	2.0	2:02	28491	260	MRO	59	2.0	2:02	28491	260	110
59	2.0	2:32	29325	4070	112	58	2.0	2:32	29325	497	112
58	1.5	3:02	30206	825	115	57	1.5	3:02	30206	825	115
56	1.5	3:32	31140	1260	120	55	1.5	3:32	31140	1260	120
55	1.5	4:02	32133	1785	127	54	1.5	4:02	32133	1785	127
53	1.0	4:32	33194	2431	138	52	1.0	4:32	33194	2431	138
51	1.0	5:02	34335	3195	152	50	1.0	5:02	34335	3195	152
49	1.0	5:32	35575	4054	170	48	1.0	5:32	35575	4054	170

## TABLE 3 (Concluded) TYPICAL CREW CHARTS FOR MANUAL CONTROL OF THE TLI BURN



Figure 12. TLI BASIC CREW MANEUVER MONITORING TECHNIQUE

If a L/V platform failure occurs before or during TLI, the crew may assume manual control of the burn with the hand controller. In order to perform a manual takeover, the guidance failure light must be lighted, indicating removal of manual takeover inhibit. In this case, the IMU would be used to determine the required attitudes.

If the launch does not occur at the nominal time, the TLI table could be updated to the launch or in real time. As a matter of course, and especially for disturbed earth parking orbit and perturbed S-IVB burns, the TLI table is updated in real time. The rationale for the monitoring procedures and for the determination of the monitoring limits is documented in References 20, 21, 22, and 23.

## C. Discussion of Malfunction Effects

A brief description of the nature of each malfunction considered, the consequences upon vehicle flight, and the abort criteria are presented in the following paragraphs:

#### 1. Loss of Thrust, One Engine

<u>Malfunction Description</u>: The description is self explanatory. The malfunction can occur in one of five F-1 engines during S-IC flight, one of five J-2 engines during S-II flight or in the single J-2 engine during S-IVB flight.

Malfunction Effects: An S-IC engine out prior to 0.2 second after liftoff will result in pad fallback. A control engine out prior to 1.4 second results in collision of an engine bell with a holddown post. Failure of engine No. 1 or 2 prior to 5.5 seconds results in tower collision. Number 4 engine out before 3.6 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 140 seconds for these early failures.

The AS-509 engine-out logic (reference 29) modified the terminal tilt arrest initiation time following any S-IC single engine-out malfunction. This modification was structured to loft the launch vehicle trajectory subsequent to any S-IC stage engine-out malfunction. While this logic alleviated launch vehicle and S/C thermal/structural problems during S-II stage flight resulting from S-IC stage single lower engineout, Mission Completion Capability was adversely affected following a center engine-out malfunction. The AS-510 engine out logic (reference 30) provided for separate inboard/outboard S-IC engine-out tilt arrest schedules. This prevented thermal/structural problems and gave maximum center engine-out mission completion capability. In order to reduce the height of the upper engine-out trajectory and consequently the abort "g's" and meet the aerodynamic heating constraints, it was decided to implement detection logic to distinguish among upper, lower, and center engine-out malfunctions (reference 31).

The engine-out X-freeze schedule applicable to S-IC stage engine failure prior to 66 seconds flight time remains unchanged from previous vehicles. Because of the high winds used in the AS-512 analysis, it is possible for an engine out between 72 and 82 seconds to cause vehicle loss of control.

Parking orbit insertion (POI) is possible for all S-IC engine out times, except for the first few seconds of flight as stated above. Mission completion capability (MCC) exists for center, upper and lower engine failures after 20, 40, and 30 sec flight time, respectively. No S-IC/S-II staging problems result from one engine out in S-IC flight. An S-IC engine-out between 158 and 161 seconds causes the S-II engines to go hardover. The maximum S-II engine hardover duration is 4.5 seconds for an S-IC engine-out at 159.6 seconds. The S-II stage base heating reevaluation analysis results (References 33, 34, 35, and 36) show that for S-IC engine-out induced S-II engine hardover inboard the flex curtain will withstand the resultant heating but due to direct impingement the S-II center engine hatband will fail within 3 seconds. Predictions indicate hatband failures may occur before abort can be implemented and will lead to catastrophic failure of the vehicle and subsequent crew loss. Therefore, for S-IC engine-out between 158.6 and 160.5 seconds, possible crew loss can occur during recovery transient in S-II flight.

No abort is required for an engine failure during S-II flight. POI and MCC is possible for any engine out at any time during S-II flight. No S-II/S-IVB staging problems result from one engine out.

S-IVB engine failure requires staging to the Command/Service Module. Mission Completion Capability is achieved for engine failures occurring within 36 sec of nominal S-IVB second burn cutoff.

EDS Abort Logic and Limits: During S-IC flight the first warning cue following thrust loss is the engine status light corresponding to the failed engine. For tower collision, the second cue will be astronaut physiology or the abort request light (i.e., after observation of the damage the ground may request an abort). For a control engine out prior to 50 seconds, the  $\Delta P$  should be ignored as a cue. For an engine failure between 50 and 120 seconds flight time, the second cue will be q-ball pressure of 3.2 PSID (100% on AOA). For an engine out after 120 seconds of flight time, the second cue will be the L/V overrate light. (In addition, the FDAI rate needles will exceed 10°/sec in pitch or yaw.)

No abort is required during S-II flight.

During S-IVB flight, the first cue will be the engine status light, and the second will be astronaut physiology or abort request light.

See Table 4 for a summary of this malfunction.

## LOSS OF THRUST, ONE ENGINE

S-IC Flight\*

Near Pad

Pad fall back

 $0 \leq t_{fail} \leq 0.2 \text{ sec}$ 

Holddown arm collision possible

 $0 \leq t_{fail} \leq 1.4 \text{ sec}$ 

Tower collision possible

 $0 \leq t_{fail} \leq 5.5 \text{ sec}$ 

Critical engines: No. 1 and 2 (tower side engines)

Negligible rate and error indication.

System status: engine status light.

Abort criteria: First cue: engine status light.

Abort cue: abort request light or physiology.

Inflight

Possible crew loss 158.6 sec  $\leq t_{fail} \leq 160.5$  sec

Possible vehicle loss.

Possible spacecraft breakup for explosive engine failures.

Possible control loss For any engine out with worst case wind phasing  $72 \leq t_{fail} \leq 82$  sec For engine No. 4 out  $0 \leq t_{fail} \leq 3.6$  sec, 110 sec  $\leq t_{control}$  loss  $\leq 140$  sec Possible false automatic abort on the 4 deg/sec rate limit For engine No. 2 or 3 out  $75 \leq t_{fail} \leq 80$  sec

POI Capability:

 $t_{fail} \ge 5.5$  seconds.

\*For catastrophic engine failures anytime in S-IC flight, loss of crew is highly probable.

TABLE 4 (Concluded)

LOSS OF THRUST, ONE ENGINE

MCC:

Center engine -  $t_{fail} \ge 20$  sec Upper engine -  $t_{fail} \ge 40$  sec Lower engine -  $t_{fail} \ge 30$  sec

Abort Criteria:

0-120 sec GET\*: (Automatic: pitch or yaw rate = 4 deg/sec or roll rate = 20 deg/sec
50-120 sec GET: Manual: first cue: engine status light abort cue: ΔP≥3.2 PSID
After 120 sec GET: Manual: first cue: engine status light abort cue: overrate light on at 9.2 deg/sec (pitch or yaw) or 20 deg/sec roll

S-II Flight

No abort required.

POI and MCC capability exists for S-II single engine failure.

S-IVB Flight

COI capability with SPS for loss of thrust in S-IVB first burn. MCC with SPS:  $t_{fail} \ge t_{NCO}^{**} - 36$  seconds

Abort criteria: First cue: Engine status light.

Abort cue: Astronaut physiology or abort request light.

\* Ground elapsed time

\*\*  $t_{NCO}$  - time of nominal cutoff for S-IVB second burn.

## 2. Loss Thrust, Two Engines

<u>Malfunction Description</u>: Thrust loss can occur in any two of the five F-1 engines during S-IC flight or any two of the five J-2 engines during S-II flight.

Malfunction Effects: Dual simultaneous engine failures before 1.8 seconds lead to holddown post collision, and before 7.4 seconds lead to tower collision for tower-side engine failures. Fallback occurs for all cases up to 30 seconds flight time. Automatic abort is initiated for two engine-out until 120 seconds of S-IC flight which is the recommended time for the crew to inhibit two engine-out auto abort. A dual engine out, especially simultaneous adjacent outboard, causes a large thrust unbalance and large dynamic transients. Structural breakup can occur in as little as 0.4 seconds after abort initiation. AS-512 S-IC dual adjacent engine-out between 147 and 161 seconds results in an S-II engine hardover condition. Crew loss can occur for dual adjacent sequential engine-out between 147 and 161 seconds due to the S-II base area overheating during the recovery transient. The S-II base heating re-evaluation analysis results (References 33, 34, 35, and 36) shows that direct impingement on the S-II stage center engine causes hatband failure within 3 seconds followed by a catastrophic vehicle failure.

S-II dual engine failures result in loss of control for early failures. The following summary shows the intervals of failure times for which dual engine-out combinations lose control during S-II flight.

S-II engine out positions	Control loss simultaneous	Control loss sequential*				
2 Upper	IGN to 420 sec	IGN to 475 sec				
2 Lower	IGN to 430 sec	IGN to 500 sec				
2 Side	IGN to 450 sec	IGN to 460 sec				
Center and one control	none	none				
2 Opposite	IGN to 220 sec**	IGN to 280 sec				
IGN = S-II ignition. *1st engine out at 165 seconds.						

EDS Abort Logic and Limits: During the first 120 seconds of S-IC flight, abort will be automatic (based on thrust O.K. pressure switch indication of 2 or more engines out). After that time, manual abort will be initiated on pitch or yaw overrates of 10 deg/sec.

\*\*No POI capability (220-275 sec).

During S-II flight, the first warning cue is two engine-out lights. The second cue is an attitude rate of 10 deg/sec pitch or yaw, or 20 deg/sec roll.

See Table 5 for a summary of this malfunction.

LOSS OF THRUST, TWO ENGINES

## S-IC Flight

Near-Pad

Earth Impact

<sup>0</sup> ≤ <sup>t</sup>fail ≤ <sup>30</sup> sec Holddown arm collision

 $0 \leq t_{fail} \leq 1.8 \text{ sec}$ 

Tower collision

 $0 \leq t_{fail} \leq 7.4 \text{ sec}$ 

Critical engines: No. 1, 2 (tower side engines)

Abort criteria: Automatic: thrust o.k. switch setting 2 or more engines out

#### In-Flight

Possible crew loss for adjacent sequential engines-out between 147 and 161 sec

Control loss probable

Adjacent control engines -  $t_{fail} \le 139$  sec Center and control engines -  $t_{fail} \le 120$  sec Opposite control engines -  $t_{fail} \le 120$  sec

POI capability:

Adjacent control engines -  $t_{fail} \ge 158$  sec Center and control engines -  $t_{fail} \ge 120$  sec Opposite control engines -  $t_{fail} \ge 120$  sec

## LOSS OF THRUST, TWO ENGINES (Concluded)

MCC:

Adjacent control engines -  $t_{fail} \ge 158 \text{ sec}$ Opposite control engines -  $t_{fail} \ge 120 \text{ sec}$ Center and control engines -  $t_{fail} \ge 120 \text{ sec}$ Abort criteria: automatic abort: thrust O.K. switch setting 2 or more engines out (0-120 sec) manual abort: First cue: engine status light Abort cue: overrate light on at 9.2 deg/sec, pitch or yaw. pitch or yaw rate of 10 deg/sec abort request light

## S-II Flight

Possible loss of control

POI capability for all controllable cases with the exception of: Opposite control engines -220 sec  $\leq t_{fail} \leq 275$  sec MCC:

Adjacent control engines -  $t_{fail} \ge 450$  sec Opposite control engines -  $t_{fail} \ge 400$  sec Center and control engine -  $t_{fail} \ge 300$  sec Abort criteria: First cue: engine status lights Abort cue: overrate light on at 9.2 deg/sec, pitch or yaw pitch or yaw rate of 10 deg/sec abort request light

## 3. One Actuator Hardover

<u>Malfunction Description</u>: One actuator hardover is the result of a thrust vector control system component failure that causes a control engine actuator to become fully extended or retracted in pitch or yaw. The affected engine retains the capability to respond to control commands in the plane of the non-malfunctioning actuator.

Malfunction Effects: A single S-IC actuator hardover during the first 1.1 second of flight will result in a base collision with the holddown arms. The failure of any of the yaw actuators in the positive direction before 2.7 seconds flight time results in a tower collision.

An S-IC actuator hardover with certain winds from 62 to 92 seconds or 102 to 105 seconds could result in loss of vehicle or loss of crew. False aborts can occur between 106 and 120 seconds due to exceedance of overrates.

An S-IC actuator hardover in pitch or yaw between 155 and 161.1 seconds causes the S-II engines to go hardover in the affected plane. The maximum S-II engine hardover duration is 13 seconds for an S-IC actuator hardover at 159.6 seconds. The S-II stage base heating reevaluation analysis results (References 33, 34, 35, and 36) show that direct impingement on the S-II stage center engine due to S-II engines hardover causes hatband failure within 3 seconds. Predictions indicate hatband failures will lead to catastrophic failure of the vehicle and subsequent crew loss. Therefore, for an S-IC actuator hardover between 156 and 160.9 seconds, possible crew loss can occur during the recovery transient.

During S-II flight, there is no loss of control for an S-II actuator hardover malfunction. A single S-II inboard directed actuator hardover results in an S-II stage base heating problem based on the S-II base heating reevaluation analysis results (References 33, 34, 35, and 36). For an actuator hardover inboard at hydraulic system unlock (TB<sub>3</sub> + 4 sec) the following sequence is predicted:

- (1) The center engine hatband fails within 10 seconds (TB<sub>3</sub> + 14 sec)
- (2) Flex curtain failure 8 seconds from actuator hardover (TB<sub>3</sub> + 12 sec)
- (3) Engine compartment components affected 3 seconds from curtain failure (TB<sub>3</sub> + 15 sec) (outboard engine)
- (4) Center engine beam damage 28 seconds from curtain failure (TB<sub>3</sub> + 40 sec)

It is predicted that an S-II actuator hardover inboard failure will lead to catastrophic failure of the vehicle and subsequent crew loss. Crew loss is predicted for this malfunction because the time to failure is less than the available warning time (approximately 25 seconds). An S-IVB actuator hardover causes the vehicle to exceed the pitch or yaw rates, consequently, requiring an abort.

EDS Abort Logic and Limits. Near pad abort will be based on astronaut physiology or the abort request light (i.e., after observation of the damage the ground may request abort.) After 50 seconds of flight the first cue is provided by a 5 degree attitude error and the second cue by a q-ball pressure of 3.2 PSID. Automatic abort results from exceeding the 4 degrees per second pitch/yaw rate limit. During S-II flight, abort will be initiated upon ground request. During S-IVB flight, the first cue is the overrate light which is lighted at 9.2 deg/sec, and the abort cue is the attitude rate which is 10 deg/sec.

See Table 6 for a summary of this malfunction.

## ONE ACTUATOR HARDOVER

## S-IC Flight

Near-Pad

Base collision possible

 $0 \leq t_{fail} \leq 1.1 \text{ sec}$ 

Tower collision possible

 $0 \leq t_{fail} \leq 2.7 \text{ sec}$ 

Critical modes: + yaw (any control engine).

Abort criteria: abort cue: abort request light or physiology.

#### In-Flight

Possible crew loss in early S-II flight 156.0 sec  $\leq t_{fail} \leq 160.9$  sec

Possible vehicle loss and crew loss

 $62 \leq t_{fail} \leq 92$  sec and  $102 \leq t_{fail} \leq 105$  sec

False Automatic abort  $106 \leq t_{fail} \leq 120$  sec

Abort criteria:

0-120 sec GET:\* Automatic: pitch or yaw rate = 4 deg/sec or roll rate = 20 deg/sec.

50-120 sec GET: Manual: First cue: 5 deg attitude error Abort cue:  $\Delta P \ge 3.2$  PSID

After 120 sec GET: No aborts required

S-II Flight

Possible crew loss for actuator hardover inboard  $\stackrel{>}{=}$  10 sec causes hatband failure followed by catastrophic failure of vehicle before abort can be implemented.

No loss of control for actuator hardover.

S-IVB Flight

Vehicle loss

Abort criteria: First cue: overrate light on at 9.2 deg/sec Abort cue: pitch or yaw rate = 10 deg/sec

\*Ground elapsed time

#### 4. One Actuator to Null

<u>Malfunction Description</u>: This malfunction is defined as any failure which causes a single actuator to remain at or near null ( $\pm 0.7$  degree) regardless of control commands or external forces placed on the actuator by the engine. The engine will respond to commands in the plane of the unfailed actuator.

<u>Malfunction Effects</u>: The failure will not cause tower collision or significantly affect vehicle performance for failures during S-IC or S-II flight. Abort is required during S-IVB first burn if the malfunction occurs earlier than 25 seconds in pitch and 15 seconds in yaw before nominal cutoff. Mission completion capability (MCC) exist during S-IVB second burn if the malfunction occurs earlier than 50 seconds in pitch and 40 seconds in yaw prior to nominal TLI cutoff.

EDS Abort Logic and Limits: No aborts are required in S-IC or S-II flight. During S-IVB burn the first cue is an attitude deviation and the second cue is the abort request light initiated from the ground. An attitude deviation of +20 deg during first burn (+45 deg during second burn) on the two S/C FDAI's is considered to be two independent cues.

See Table 7 for a summary of this malfunction.

#### ONE ACTUATOR TO NULL

#### S-IC and S-II Flight

No effect in this flight period.

#### S-IVB Flight

Abort is required during S-IVB first burn if the malfunction occurs earlier than 25 seconds in pitch and 15 seconds in yaw before nominal cutoff. MCC exist during S-IVB second burn if the malfunction occurs earlier than 50 seconds in pitch and 40 seconds in yaw prior to nominal TLI cutoff.

Abort criteria: First cue: attitude deviation\*. First burn +20 degrees Second burn +45 degrees

Abort cue: abort request light

\*The two S/C FDAI's are considered to be two independent cues.

## 5. Saturated Error/Rate Signal

Malfunction Description: The control signals to the flight control computer are attitude error and attitude rate. Saturated error/ rate signals means that a large and erroneous value of either attitude error or attitude rate has entered either the pitch, yaw or roll channel of the flight control computer. As a consequence, all control engines will move simultaneously to a fully deflected position.

Malfunction Effects: A saturated error or rate signal will result in rapid divergence at any flight time failure. A saturated error or rate signal prior to 1.5 seconds results in pad interference. A plus yaw saturated signal before 3.3 seconds or a minus yaw signal before 6.1 seconds will result in tower collision. Failures after tower clearance result in aborts on angular rates. Possible crew loss can occur for failures between 30 and 90 seconds of S-IC flight. Vehicle breakup will follow automatic abort for failure during the high-q region. Breakup will not occur for upper stage failures due to the absence of aerodynamic forces.

EDS Abort Logic and Limits: Automatic abort on 4 deg/sec rate in pitch or yaw or 20 deg/sec in roll will occur within 1.0 second after the malfunction during the time that automatic abort is active. Sudden structural failure will occur after abort for malfunctions between 30 and 120 seconds. After 120 seconds, automatic abort may be manually inhibited by the crew. After automatic abort is inhibited, the first cue for manual abort will be the launch vehicle overrate light, which is turned on at 9.2 deg/sec in pitch or yaw or 20 deg/sec in roll. The second cue will be spacecraft attitude rate indication of 10 deg/sec in pitch or yaw or 20 deg/sec in roll.

For failures during S-II or S-IVB flight, the first cue will be the launch vehicle overrate light, which is turned on at 9.2 deg/sec. The second cue will be a spacecraft attitude rate indication of 10 deg/sec. For the special case of saturated error signal in S-II where the maximum pitch and yaw rate is 9.1 deg/sec, the abort cue will be 20 degrees attitude deviation.

See Table 8 for a summary of this malfunction.



#### SATURATED ERROR/RATE SIGNAL

#### S-IC Flight

Near-Pad

Pad interference possible

 $t_{fail} \leq 1.5$  seconds

Tower collision possible

+ Yaw t<sub>fail</sub> < 3.3 seconds

- Yaw t<sub>fail</sub>  $\leq$  6.1 seconds

Abort criteria: automatic: pitch or yaw rate = 4 deg/sec or roll rate = 20 deg/sec.

#### In-Flight

Control loss certain

Crew loss possible 30 sec  $\leq t_{fail} \leq 90$  seconds

Abort criteria: automatic: Pitch or yaw rate = 4 deg/sec roll rate = 20 deg/sec manual: First cue: overrate light on at 9.2 deg/sec (pitch or yaw) or 20 deg/sec roll Abort cue: pitch or yaw rate = 10 deg/sec or roll rate = 20 deg/sec

#### S-II and S-IVB Flight

Control loss certain

Abort	criteria:	First o	cue:	*overrate light on at 9.2 deg/sec (pit	:ch
				or yaw) or 20 deg/sec (roll)	
		Abort o	cue:	*pitch or yaw rate of 10 deg/sec, or	
				roll rate = 20 deg/sec	
				abort request light	

\*For the special case of saturated error signal in S-II where the maximum pitch and yaw rate is 9.1 deg/sec, the abort cue will be 20 degrees attitude deviation.

#### 6. Loss of Inertial Attitude

Malfunction Description: For the AS-509 vehicle, Reference 27 indicated that 35 percent of all inertial platform failures were undetectable by the LVDC. Due to changes in the AS-510 and subsequent vehicles (redundant 6D10 battery), this figure has been reduced to 8 percent, with corresponding reductions in crew, vehicle, and mission loss criticalities.

Platform outputs are monitored by the LVDC and tested for reasonableness. On successive failures of the RT, the LVDC issues a guidance reference failure discrete. This discrete lights the L/V GUID light on the S/C instrument panel. It also turns on the L/V overrate light prior to automatic abort deactivation.

A backup system permits S/C takeover of guidance. Switchover to the backup system is performed manually upon detection of the L/V GUID light.

Malfunction Effects: Undetectable malfunctions occurring before 6.1 seconds can cause tower collision. For failures not detected by the RT, switchover to backup guidance will not take place since the L/V GUID light will not be lighted and abort will be necessary when EDS lights are violated. This can result in possible crew loss for undetectable failures between 30 and 90 seconds of S-IC flight.

The early malfunction effects are manifested by divergence from the planned attitude time history and flight path (possibly even inland) with consequent violation of range safety limits. If malfunction is detected by the RT, crew will initiate spacecraft takeover of guidance, navigation, and control.

If the failure occurs near or before the high-Q region of flight, rapid divergence and subsequent loss of control can occur. Loss of inertial attitude in S-II and S-IVB stage flight could result in a gradual divergence. Gradual divergence from trajectory and attitude will eventually require an abort or spacecraft takeover if the RT has failed.

EDS Abort Logic and Limits: If a platform failure results in failure of the RT between liftoff and 120 seconds, the guidance failure light and L/V overrate light will be lighted, and the crew will initiate spacecraft takeover. If the failure does not result in failure of the RT, an abort will be initiated automatically when the EDS rate limit is exceeded or manually when the attitude error and  $\Delta P$  limits are exceeded. If a platform failure results in failure of the RT after 120 seconds, the guidance failure light will be lighted, and the crew will initiate takeover. If the failure does not result in failure of the RT, an abort will be initiated when EDS rate limits are exceeded or when the limit lines of Figure 1 (page 10) are crossed.

See Table 9 for a summary of this malfunction.

## LOSS OF INERTIAL ATTITUDE

#### S-IC Flight

Near Pad

Pad interference possible  $t_{fail} \leq 1.5$  seconds

Tower collision + yaw t < 3.3 seconds - yaw t fail < 6.1 seconds fail =

#### In-Flight

the.

Possible crew loss for undetectable failure between 30 and 90 sec of S-IC flight

Control loss possible if S/C takeover not effected immediately.

Abort criteria: Automatic: Pitch or yaw rate = 4 deg/sec or roll rate = 20 deg/sec

> Manual: First cue: (a) Overrate light on at 9.2 deg/sec (pitch or yaw) or 20 deg/sec roll

- (b) Pitch, yaw or roll error ± 5 deg
- Abort cue: (a) Pitch or yaw rate = 10 deg/sec or roll rate = 20 deg/sec

(b)  $\Delta P = 3.2 PSID$ 

## S-II and S-IVB Flight

Divergence from intended flight path.

For undetectable failures abort on FDO and EDS limits.

MCC exists for detectable RT failures that do not lose control at switchover.

## 7. Erroneous Attitude Error Signal

<u>Malfunction description</u>: Erroneous attitude error signal is considered to be any failure that causes a false attitude error signal between zero and saturated to the flight control computer (FCC). The most probable failure mode and the one considered is an attitude error which fails to zero. This failure does not preclude rate control.

<u>Malfunction Effects</u>: Erroneous attitude error signal will not result in tower collision. The early malfunction effects are manifested by divergence from the planned attitude time history and flight path (possibly even inland) with consequent violation of range safety limits.

If the failure occurs near or before the high-q region of flight, rapid divergence and subsequent loss of control will occur. After 120 seconds in S-IC flight and in upper stage flight, attitude divergence is slower and no structural breakup occurs. Gradual divergence of trajectory and attitude will eventually require abort.

EDS Abort Logic and Limits: During S-IC flight up to 120 seconds, safe aborts are provided by the 4 degrees per second automatic abort setting. Manual abort cues are 5 degrees attitude error, 3.2 PSID q-ball pressure (between 50 and 120 seconds) and 10 deg/sec overrate (after 120

Abort cues for failures during S-II stage flight prior to COI capability and S-IVB flight are  $\pm$  20 degrees attitude deviation ( $\pm$  45 degrees for S-IVB second burn) and abort request light.

Early staging cues for failures during S-II after COI capability are  $\pm$  85 degrees pitch attitude error or  $\pm$  45 degrees yaw attitude error, actuator not responding (ground), and FDO limits or verified trajectory deviation (ground).

During S-II and S-IVB flight the abort request light is initiated by the ground when the vehicle's trajectory crosses FDO limit lines. POI can be achieved for failures 85 seconds prior to S-IVB cutoff. During second burn, mission completion capability exists for failures 50 seconds prior to insertion.

See Table 10 for a summary of this malfunction.

#### ERRONEOUS ATTITUDE ERROR SIGNAL

#### S-IC Flight

Near-Pad

Tower collision does not occur

In-Flight

Control loss certain

Rapid divergence in high-q for  $t_{fail} \leq 120$  sec

Abort criteria; Automatic abort: 4 deg/sec in pitch or yaw Manual Abort Cues: 5 deg p/y attitude error Q-ball pressure of 3.2 PSID (between 50 and 120 sec) 10 deg/sec overrate (after 120 sec)

#### S-II and S-IVB Flight

Divergence from intended flight path.

Gradual divergence except for failure immediately following IGM initiation.

Early staging cues (failures during S-II flight after COI capability)

- a.  $\pm 85$  degree pitch attitude error or  $\pm 45$  degree yaw attitude error.
- b. Actuator not responding (ground)
- c. FDO limits or verified trajectory deviation (ground)

POI can be achieved for failures occurring 85 seconds prior to S-IVB first burn cutoff.

MCC can be achieved for failures occurring 50 seconds prior to insertion.

Abort criteria: First cue: attitude deviation: \* S-II & S-IVB first burn + 20 degrees S-IVB second burn + 45 degrees Abort cue: abort request light

\* The two S/C FDAI's are considered two independent cues.

## 8. Loss of Attitude Rate Signal

Malfunction Description: Any failure that causes a false vehicle rate indication of zero degrees per second to the flight control computer is defined as a loss of attitude rate signal. This failure does not preclude attitude control.

Malfunction Effects: This malfunction is characterized by divergent attitude oscillation for all flight stages. Malfunctions during S-IC burn are complicated by aerodynamic forces which cause more rapid loss of control. In a high-q region, vehicle structural breakup eventually occurs. This failure results in attitude oscillations between 0.08 and 0.16 Hertz depending on control gains and launch vehicle inertial characteristics.

During S-II flight, vehicle loss of control occurs approximately 45 seconds after the malfunctions.

During S-IVB flight, loss of control in the pitch/yaw plane prior to insertion is dependent on the time of the malfunction. POI capability during first burn exists 30 seconds prior to cutoff for loss of pitch rate and 100 seconds prior to cutoff for loss of yaw rate. During second burn, mission completion capability exists 50 seconds prior to cutoff for loss of pitch or yaw rate. Loss of roll rate does not cause loss of mission during S-IVB flight.

EDS Abort Logic and Limits: During S-IC flight, safe automatic aborts are provided by the 4 degrees per second overrate setting. During the high-q region of flight, automatic abort on rates will occur before exceeding 3.2 PSID. After automatic abort is disabled, manual abort cues are 5 degrees attitude error and L/V rate light. Manual abort cues for failures during S-II and S-IVB flight are the L/V rate light and 10 degrees per second rate on S/C FDAI.

See Table 11 for a summary of this malfunction.

#### LOSS OF ATTITUDE RATE SIGNAL

S-IC Flight

Near-Pad

Tower collision does not occur.

In-Flight

Control loss certain

Abort criteria:

Automatic: 4 deg/sec pitch or yaw attitude rate Manual: First cue: 5 degrees attitude error Abort cue: overrate light on at 9.2 deg/sec,pitch or yaw, or 20 deg/sec, roll.

#### S-II and S-IVB Flight

Control loss certain in S-II flight.

POI can be achieved for failures occurring 30 seconds in pitch and 100 seconds in yaw prior to S-IVB first burn cutoff.

MCC can be achieved for failures occurring 50 seconds in pitch or yaw prior to insertion.

MCC can be achieved for loss of roll rate during S-IVB flight.

Periodic oscillations which do not diverge occur in coast periods. Abort criteria: First cue: overrate light on at 9.2 deg/sec, pitch or yaw or 20 deg/sec, roll. Abort cue: 10 deg/sec pitch or yaw attitude rate or 20 deg/sec roll attitude rate.

#### 9. Loss of Inertial Velocity

Malfunction Description: Modifications have been made to the backup acceleration profile logic resulting in less perturbed orbits and increased mission completion capability for loss of inertial velocity (reference no. 28).

The accelerometer failure modes considered are X, Y or Z axis accelerometer failures to zero. The primary accelerometer failure is to a zero output. The hardware failures producing this condition can be accelerometer servo-loop failures, signal conditioner failures or Launch Vehicle Data Adapter (LVDA) ladder failure. The LVDC Accelerometer Processing Loop (APL) tests for zero or unreasonable accelerometer data. After erroneous output is detected, pre-stored F/M data is substituted for the failed axis. If subsequent data from the failed axis should pass the tests in the APL, that information will be used in place of the backup F/M data.

For all previous vehicles it has been assumed that the accelerometer failures occur at the beginning of the major loop. However, the actual phasing of a particular accelerometer failure within its major loop is random and the orbit obtained is a function of this phase relationship. The orbit is also affected by the phasing, within the major loop, of the major flight events (i.e., S-IC CECO, S-IC OECO, S-II 90% thrust, S-II CECO, S-II OECO, and S-IVB 90% thrust). These effects combine to produce a band of orbit apogee/perigee conditions possible for any particular accelerometer failure time.

Malfunction Effects: The primary effect of an accelerometer malfunction is the trajectory deviation which results from using backup accelerometer data. This malfunction can occur in all stages but will produce trajectory deviations only during S-II and S-IVB flight and can result in a band or envelope of off nominal orbit apogee/perigee conditions.

A safe parking orbit (perigee  $\geq 70$  n. mi.) is achieved for X- accelerometer failures for nominal and +3 $\sigma$  performing L/V, and after 140 seconds for -3 $\sigma$  performing L/V.

Z-accelerometer failures during boost-to-orbit (BTO) result in elliptical orbits with perigee altitude of 90 n mi for nominal and  $+3\sigma$  performing L/V and  $\geq 70$  n mi perigee after 100 sec for a  $-3\sigma$  performing L/V.

Y - accelerometer failures result in nearly circular 90 n. mi. orbits with an insignificant error in inclination and descending node angles for nominal and  $+3\sigma$  performing vehicle.

If an accelerometer fails in boost to orbit, a navigation update will be performed. With the navigation update, TLI can be achieved if the backup F/M closely approximates the acceleration profile of the S-IVB stage's second burn.

All out-of-orbit accelerometer failures result in perturbed TLI conditions requiring midcourse correction.

EDS Abort Logic and Limits: No abort is required for a nominal or a  $+3\sigma$  performing L/V for single axis accelerometer failures to zero assuming that a navigation update is performed following into-orbit malfunctions. However, Mode IV abort may be required for low performing L/V.

See Table 12 for a summary of this malfunction.

#### S-IC Flight

No loss of control

POI capability (a minimum 70 n mi orbit):

X-accelerometer failure:

Nominal and +3 performing  $L/V - t_{fail} \ge 0$ 

 $-3\sigma$  performing L/V  $-t_{fail} \ge 140$  sec

Y-accelerometer failure:

Nominal and  $\pm 3\sigma$  performing L/V - t<sub>fail</sub>  $\geq 0$ 

Z-accelerometer failure:

Nominal and  $+3\sigma$  performing L/V - t<sub>fail</sub>  $\geq 0$ 

-  $t_{fail} \ge 100$  sec

```
-3 performing L/V
```

Mission Completion Capability

X-accelerometer failure:

No capability for  $-3\sigma$  performing L/V before 140 sec. Mode IV abort may be required for  $-3\sigma$  performing L/V before 140 sec.

Y-accelerometer failure:  $t_{fail} \ge 0$ 

Z-accelerometer failure:

No capability for  $-3\sigma$  performing L/V before 100 sec. Mode IV abort may be required for  $-3\sigma$  performing L/V before 100 sec.

Abort criteria: no abort required in S-IC flight.

## LOSS OF INERTIAL VELOCITY (Concluded)

## S-II and S-IVB Flight

Attitude deviations can result

POI capability for all failures; however, subnominal orbit may result.

Mission completion capability: (vehicle performing in the  $+3\sigma$  range)

Abort criteria: No abort required in S-II and S-IVB flight.
### 10. Loss of FCC Switch Points

<u>Malfunction Description</u>: Loss of FCC (flight control computer) switch points is assumed to be any malfunction that causes early gain switching or late (or no) gain switching.

Malfunction Effects: An early control system gain change (in the high q region) may result in vehicle loss when combined with certain quartering headwinds in the 95 percentile wind rose. A late gain change does not require abort even if no gain change occurs during S-IC flight.

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

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

EDS Abort Logic and Limits: The abort cues for this malfunction are 5 degrees attitude error and q-ball pressure of 3.2 PSID during S-IC flight. No abort is required during S-II and S-IVB flight.

See Table 13 for a summary of this malfunction.

# LOSS OF FCC SWITCH POINTS

S-IC Flight

1.

Abort Criteria: First Cue: 5 deg attitude error Abort Cue:  $\triangle$  P > 3.2 PSID

S-II and S-IVB Flight

No abort required

## 11. Propellant Utilization (PU) Failure

Malfunction Description: This failure is defined as that causing the PU valve in the S-II or S-IVB stage to assume some position other than that predetermined. No malfunctions are considered to occur before PU unlock. It is assumed that all valves fail simultaneously to high or low stop for the remainder of the flight.

<u>Malfunction Effects</u>: POI capability is retained for all PU failures. Mission Completion Capability (MCC) is retained during S-II flight for PU failure to the low position after 255 sec and to the high position at any time. For S-IVB first-opportunity reignition, a PU failure to the low position during the first or second burn will not prevent MCC. For S-IVB second-opportunity reignition, a PU failure to the low position during all of the first burn will result in loss of MCC. Failure to achieve MCC is due to propellant depletion. No loss of MCC will result from PU failure to the high position during S-IVB first and second burns.

EDS Abort Logic and Limits: For failures prior to POI and S-IV second burn which require an abort in parking orbit or during S-IV second burn, the abort request light will be the abort cue.

See Table 14 for a summary of this malfunction.

#### PROPELLANT UTILIZATION FAILURE

### S-IC Flight

No effect in this flight period.

## S-II Flight

Trajectory deviations Perturbed burn times

POI can be achieved for failures to low or high stop at anytime.

MCC can be achieved for failures to low stop after 255 seconds and failures to high stop at anytime.

#### S-IVB Flight

Perturbed burn times Propellant depletion

POI can be achieved for failure to low or high stop at anytime.

First opportunity reignition: MCC can be achieved for failure to low or high stop at anytime during first or second burn.

## Second opportunity reignition:

MCC can be achieved for failure to high stop during first burn or failure to low or high stop at anytime during second burn.

Abort criteria: Abort cue: Abort request light.

12. Loss of One APS Module

Malfunction Description: This failure is one which prevents any of the three reaction jets on one APS module from firing.

Malfunction Effects: During S-IVB powered flight, the roll control becomes sluggish, but loss of control does not result. During parking orbit, control will be lost, and abort will be required if attitude is not controlled by the RCS. TD&E will be attempted if it appears feasible.

EDS Abort Logic and Limits: No abort is required since control can be maintained by means of the RCS.

See Table 15 for a summary of this malfunction.

## LOSS OF ONE APS MODULE

## S-IC and S-II Flight

No effect in this flight period.

## S-IVB Flight

No abort required

Roll control sluggish.

## Parking Orbit

Crew will stabilize the vehicle with CSM RCS

### Translunar Coast

Crew will stabilize the vehicle with CSM RCS TD&E will be attempted if it appears feasible.

### 13. Loss of Both APS Modules

Malfunction Description: This failure is defined as one which prevents the firing of any of the reaction control jets on both APS modules.

<u>Malfunction Effects</u>: Loss of both APS modules during S-IVB powered flight may result in loss of control. If excessive roll torques (>60 newton-meters) are present, loss of control occurs due to the interaction with pitch and yaw when the roll attitude error exceeds 135 degrees. During S-IVB powered flight the abort cues are EDS rate limits.

During coast flight the vehicle is uncontrollable and slow attitude divergence occurs. Mission rules call for RCS takeover. The cues are ground command (i.e., APS manifold pressure below 100 PSIA) and loss of attitude control (i.e., when either pitch/yaw error and rate exceed  $\pm 2.5^{\circ}$  and  $\pm 0.3^{\circ}$ /sec, respectively, or roll error and rate exceed  $\pm 3.5^{\circ}$  and  $\pm 0.5^{\circ}$ /sec, respectively). TD&E will be attempted if it appears feasible.

EDS <u>Abort Logic and Limits</u>: In order to improve mission completion capability it is recommended that CSM-RCS takeover be inititated during powered flight on + 20 degrees roll attitude deviation and ground command (i.e., APS manifold pressure below 100 psia). The first cue will be large roll attitude deviations, and the abort cue will be the abort request light.

See Table 16 for a summary of this malfunction.

## LOSS OF BOTH APS MODULES

## S-IC and S-II Flight

No effect in this flight period.

## S-IVB Flight (First and Second Burns)

Possible loss of control: \*

Abort criteria: Manual: First cue: overrate light on at 9.2 deg/sec, pitch or yaw Abort cue: 10 deg/sec pitch or yaw attitude rate

## Parking Orbit

Loss of control if vehicle is not controlled by CSM-RCS

## Translunar Coast

Loss of control if vehicle not controlled by CSM-RCS.

TD&E will be attempted if it appears feasible.

\* CSM-RCS takeover may be used to maintain roll control.

### 14. Staging and Sequencing Malfunctions

<u>Malfunction Description</u>: Staging or sequencing malfunctions are defined as any malfunction which causes premature staging, lack of staging, or complications during staging.

Malfunction Effects: Staging malfunctions which would prevent S-IC/S-II separation would require an abort. The removal of the four S-II stage ullage motors and four of the eight S-IC stage retrorockets on AS-510 reduced clearances during S-IC/S-II first plane separation. AS-511 and AS-512 staging and sequencing malfunctions have been affected due to the addition of four S-IC stage retrorockets which yields an improvement in S-IC/ S-II separation clearance. Failure of an S-IC stage retrorocket does not impair the flight. Delay or failure of the S-IC/S-II interstage to separate could cause the thermal environment limits in the S-II boattail area to be exceeded which will probably lead to an explosion. The mission rule for S-II second plane separation failure at TB3 + 31.7 seconds is for the crew to abort prior to TB3 + 1 minute 45 seconds. Failure to jettison the launch escape tower (LET) does not cause any problems during powered flight; however, primary mission will probably be lost. Staging malfunctions which would prevent S-II/S-IVB staging would require an abort. Failure of either an S-II stage retrorocket or an S-IVB stage ullage motor would not impair the flight.

EDS Abort Logic and Limits: In the event no S-IC/S-II or S-II/S-IVB staging occurs, the first cue is physiological and the abort cue will be the abort light initiated by the ground. No abort is required for a retrorocket out during S-IC/S-II staging or an ullage or retrorocket out during S-II/S-IVB staging. No abort is required for failure to jettison LET except in parking orbit. If the S-IC/S-II interstage fails to jettison, the first cue is the S-II separation light, and the abort cue will be the abort request light initiated by the ground.

See Table 17 for a summary of this malfunction.

# STAGING AND SEQUENCING MALFUNCTIONS

## S-IC Flight

No effect in this flight period.

### S-II Flight

No abort required for LET jettison failure.

Abort certain if S-IC/S-II separation does not occur.

Abort certain if S-IC/S-II second plane separation does not occur. (The crew is to abort prior to TB3 + 1 minute 45 sec.)

Abort criteria: First cue: S-II sequencing light or physiological. Abort cue: abort request light.

#### S-IVB Flight

No abort required for LET jettison failure

Abort certain if S-II/S-IVB separation does not occur

Mission lost if S/C separation does not occur

Abort criteria: First cue: center engine status light. Abort cue: abort request light.

### Parking Orbit

Probable mission loss for LET jettison failure Tower must be removed before leaving parking orbit. Abort criteria: abort cue: abort request light

## D. Abort Cues and EDS Limit Summary

The automatic abort cues and EDS limits are summarized in Table 18. The recommended Manual abort cues and EDS limits for S-IC flight are summarized in Table 19. The recommended Manual abort cues and EDS limits for S-II and S-IVB flight are summarized in Table 20. In the judgment of the Flight Limits Sub-Panel, these limits are the optimum limits in that they yield maximum crew safety possible for a minimum probability of aborting a good launch vehicle.

STAGE	FLIGHT TIME (SEC)	ABORT CUE	EDS LIMIT
S-IC	0 < t < 120	Pitch or Yaw Attitude Rate Roll Attitude Rate Thrust OK Switch (89% of Rated Thrust) Structural Wires between I. U. and CM	+4 deg/sec +20 deg/sec Two or More Eng. Out Failure of 2 out of 3 will indicate structural failure between the 2 components.
S-IC, S-II and S-IVB	t > 120	AUTOMATIC ABORT NOT REQUIRED	

TABLE 18 AUTOMATIC\* ABORT CUES AND EDS LIMIT SUMMARY

\* Manual abort cues and EDS Limit summary are on the following pages.

## S-1C STAGE MANUAL ABORT CUES AND EDS LIMIT SUMMARY

FLIGHT TIME (sec)	ABORT CUE	EDS LIMIT
Near Pad 0 < t < 50	L/V Rate Light* S/C Attitude Rate Indicator: Pitch or Yaw Attitude Rate Roll Attitude Rate Engine Status Light Abort Request Light Voice Request	+ 4 deg/sec + 20 deg/sec
<ul> <li>L/V Rate Light*</li> <li>S/C Attitude Rate Indicator: Pitch or Yaw Attitude Rate Roll Attitude Rate         Pitch or Yaw Attitude Error***         Q-Ball ΔP****         Engine Status Light         Abort Request Light     </li> </ul>		+ 4 deg/sec + 20 deg/sec + 5 deg 3.2 PSID (100%)
L/V Rate Light* S/C Attitude Rate Indicator: Pitch or Yaw Attitude Rate Roll Attitude Rate Pitch or Yaw Attitude Error*** Engine Status Light Abort Request Light		<u>+</u> 10 deg/sec <u>+</u> 20 deg/sec <u>+</u> 5 deg

\*L/V Rate Light Settings:

Ditch or Yow	4( <u>+</u> 0.49) deg/sec	Liftoff to Automatic Abort Deactivation (120 sec)
Attitude Rate	9.2( <u>+</u> 0.8) deg/sec	Automatic Abort Deactivation (120 sec) to <b>S-I</b> VB Cutoff
Roll Attitude Rate	$20 \begin{pmatrix} +1.9\\ -2.0 \end{pmatrix} deg/sec$	Liftoff to S-IVB Cutoff

\*\* This period will be delayed for engine out prior to this time.

\*\*\* Use as first cue only,

\*\*\*\* For a control engine out prior to 50 seconds,  $\Delta P$  should be ignored as an abort cue.

S-II & S-IVB STAGE MANUAL ABORT CUES AND EDS LIMIT SUMMARY

FLIGHT STAGE	ABORT CUE	EDS LIMIT
S-II & S-IVB First Burn	L/V Rate Light* S/C Attitude Rate Indicator: Pitch or Yaw Attitude Rate Roll Attitude Rate Pitch or Yaw Attitude Deviation** FDO Display Engine Status Light Abort Request Light Voice Request	+ 10 deg/sec + 20 deg/sec + 20 deg Limit Exceeded
S-IVB Second Burn L/V Rate Light* S/C Attitude Rate Indicator: Pitch or Yaw Attitude Rate Roll Attitude Rate Pitch or Yaw Attitude Deviation Engine Status Light Abort Request Light Voice Request		<u>+</u> 10 deg/sec <u>+</u> 20 deg/sec <u>+</u> 45 deg

\*L/V Rate Light Settings:

Pitch or Yaw Attitude Rate	9.2 ( <u>+</u> 0.8) deg/sec	
Roll Attitude Rate	$20 \begin{pmatrix} +1.9\\ -2.0 \end{pmatrix} deg/sec$	

\*\* Attitude deviation should be ignored as an abort cue following an S-IC or S-II engine out.

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