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March 15, 1968

SPACECRAFT DISPERSION ANALYSIS FOR APOLLO 6 (A-2 OR AS-502/CSM-020)

By John T. Parker,
Guidance and Performance Branch

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PROJECT APOLLO

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SPACECRAFT DISPERSION ANALYSIS FOR APOLLO 6 (A-2 OR AS-502/CSM-020)

By John T. Parker

SUMMARY

This document presents the results of deviations and uncertainties for the Apollo 6 (A-2 or AS-502/CSM-020) final spacecraft dispersion analysis. These dispersions were achieved by propagating each error source from lift-off through the complete mission to 23 500-ft altitude.

The combined root-sum-squared (RSS) 3σ spacecraft deviations at entry interface are as follows:

Time, sec.	±125.0
Inertial velocity, fps	±29.1
Inertial flight-path angle, deg.	±0.562
Inertial azimuth, deg	±0.767
Geodetic latitude, deg	±0.324
Longitude, deg	±1.244
Altitude, ft	±0.0

The combined RSS 3σ spacecraft uncertainties at entry interface are as follows:

Time, sec.	0.0
Inertial velocity, fps	±9.072
Inertial flight-path angle, deg.	±0.182
Inertial azimuth, deg	±0.140
Geodetic latitude, deg	±0.103

Longitude, deg. . . . . ±0.334

Altitude, ft. . . . . ±6109.0

The RSS 3σ deviations from the nominal peak heating rate, the total heat load, and the maximum load factor are:

First maximum heating rate, Btu/ft^2/sec. . . . . ±121.93

Second maximum heating rate, Btu/ft^2/sec . . . . . ±12.12

Total heat load, Btu/ft^2 . . . . . ±3853.00

First maximum load factor, g . . . . . ±2.13

Second maximum load factor, g. . . . . ±1.96

The RSS 3σ uncertainties in the down-range and cross-range errors which describe the command module (CM) dispersion ellipsoid at 23 500-ft altitude are:

Down range, n. mi. . . . . ±3.08

Cross range, n. mi. . . . . ±51.74

As the flight-path angle approaches a 3σ steep value, the capability of the CM to reach the target becomes marginal because of the long entry range and relatively low lift-to-drag ratio (L/D). For example, for a nominal L/D and a 3σ steep flight-path angle, the spacecraft will fall short of the target by about 80 n. mi. This miss distance can increase to about 200 n. mi. as the L/D approaches the 3σ low value for the 3σ steep flight-path angle case. The following Y-gyro bias drift deviations reflect this nonlinearity.

Up range, n. mi. . . . . 80.

Down range, n. mi. 1.87

In conjunction with examining the mission for trajectory dispersions, an analysis was also made of the service propulsion system (SPS) performance. The results of this analysis indicated a 3σ SPS propellant dispersion of 400 lb.

INTRODUCTION

The purpose of the Apollo 6 final dispersion analysis is to verify the feasibility of the operational trajectory (ref. 1) by examining deviations and uncertainties caused by the major error sources. The Guidance and Performance Branch of the Mission Planning and Analysis Division (MPAD) performed the dispersion analysis assuming two state vector updates for the portion of the mission from lift-off to entry interface at 400 000-ft altitude. The Lunar Mission Analysis Branch of MPAD correlated the results at entry interface to 23 500-ft altitude using the primary entry guidance scheme.

Because of the large quantity of data presented in this analysis and the varying uses of this type data, no special emphasis was given to generating partial derivatives for special trajectory or vehicle parameters. Special assistance may be obtained from the Guidance and Performance Branch of the MPAD for any further reduction of the data of this analysis to satisfy individual needs.

The error sources used in this analysis were chosen as a result of careful examination of all possible sources and experience obtained in performing various dispersion analyses for Apollo 4 and Apollo 6. The error sources considered in this report cause nearly 100 percent of the possible dispersions.

The following analyses are included in this document:

(a) An analysis of the pertinent error sources and the combined RSS of the resulting dispersions (including deviations and uncertainties) based on the nominal Apollo 6 trajectory. The results include tables of pertinent trajectory parameters and the standard deviations of these parameters at various points in the trajectory. These tables also include the combined RSS of the dispersions on each of the parameters.

(b) An analysis of the effects of the pertinent error sources on the entry phase of the mission.

(c) An analysis of the effects of the pertinent error sources on the SPS propellant dispersions.

This report assumed that two state vector updates were made, as was previously agreed to by Flight Operations Directorate prior to the Apollo 4 mission. The resulting analyses and concurrence are given in references 2 and 3.

Mission Description

The sequence of events and state vectors for the Apollo 6 mission are presented in table I. Also included for interest is a diagram of the nominal mission profile (fig. 1). The following describes the various phases of the mission.

(a) Prelaunch phase - The Apollo 6 mission is to be launched from the Eastern Test Range, pad A of launch complex 39. The pad azimuth is 90° 12' from true north. Prior to launch, the inertial measuring unit (IMU) is aligned to a flight azimuth of 72° from true north. Platform release occurs at lift-off.

(b) Launch phase - The launch phase consists of nominal burns of the S-I, S-II, and S-IVB vehicles, culminating at injection of the spacecraft into a high apogee, earth-intersecting ellipse. During this phase the guidance and navigation (G&N) system monitors vehicle acceleration and, from these, computes an estimated position and velocity. A ground update of this estimated state vector is assumed to occur at second ignition of the S-IVB for the injection burn.

(c) High-altitude phase - Two guided SPS burns are separated by a long coast (approximately 6 hours) and followed by a 4-minute coast to entry. First SPS burn ignition occurs approximately 3 minutes after cutoff of the S-IVB injection burn and is proceded by S-IVB/CSM separation and an ullage burn. A second state vector update is assumed to occur approximately 13 minutes prior to the second SPS ignition. At this point a time of free fall (t_{ff}) calculation (i.e., time to 400 000-ft altitude) is begun and the "average g" guidance mode is initiated.

(d) Entry phase - This phase is the portion of the mission from entry interface 400 000 ft to drogue chute deployment at 23 500 ft. During this phase one of the primary spacecraft objectives, a test of the heat shield, will be accomplished.

METHOD OF ANALYSIS

The basic tool for the trajectory simulation was subprogram 32 of SG-GEM (ref. 4), which was modified to simulate the various spacecraft systems that were considered in this study by the addition of mathematical models of the particular systems.

The various error and deviation sources considered in this study originated from the following:

(a) G&N system errors - Figure 2 illustrates the IMU and the alignment of the gyros and accelerometers with respect to the platform axis. The G&N error sources and magnitudes were obtained from reference 5, and the IMU error model was obtained from reference 6.

(b) S-IVB injection dispersions - The S-IVB injection dispersions were obtained from reference 7. It is pointed out that this reference, although indicating approximately the same S-IVB injection dispersions caused by performance errors as all previous MSFC references, did give about twice the S-IVB injection dispersions caused by G&N errors in previous MSFC references. The effect of this was negligible on the trajectory dispersions but significant on the SPS propellant usage. However, the overall SPS propellant usage due to all considered dispersions resulted in about the same magnitude as indicated in the preliminary spacecraft dispersion analysis for AS-502 (ref. 8) because the two SPS burns decreased in total burn time, thus lessening the propellant effects due to accelerometer errors.

(c) Entry parameter errors - These errors consisted of deviations in L/D and atmosphere.

Each error source was considered to be independent and to have Gaussian distribution with a zero mean. All G&N and S-IVB injection errors were evaluated assuming a perfect state vector update. All errors, except when noted otherwise, are 3σ.

RESULTS

3σ SPS Propellant Dispersion

Table II gives the effects of the 3σ system errors on the SPS propellant consumption. The 400-lb flight propellant reserve is large enough to cover the dispersions.

Table III presents a breakdown of the nominal propellant usage by the service module. The usable propellant remaining after completion of the second SPS burn (see refs. 9, 10, and 11) is sufficient to accommodate dispersions, tolerances, and reserves, and still maintain an adequate propellant margin.

3σ Entry Interface Dispersions

The 3σ dispersions on the CM inertial velocity and inertial flight-path angle at entry are presented in tables IV and V. It can be seen from table IV that the major contributors to the entry velocity deviations are the Z-accelerometer bias and the Y-gyro drift errors. The major contributor to the entry inertial flight-path angle deviation is the Y-gyro drift.

The RSS 3σ deviations of the entry parameters for the independently applied perturbations are:

Time, sec	±125.0
Inertial velocity, fps.	±29.1
Inertial flight-path angle, deg	±0.562
Inertial azimuth, deg	±0.767
Altitude	0.0
Geodetic latitude, deg.	±0.324
Longitude, deg.	±1.244

The RSS 3σ uncertainties of the entry parameters for the independently applied perturbations are:

Time, sec	0.0
Inertial velocity, fps.	±9.072
Inertial flight-path angle, deg	±0.182
Inertial azimuth, deg	±0.140
Geodetic latitude, deg.	±0.103
Longitude, deg.	±0.334
Altitude, ft	±6109.0

3σ 23 500-ft Altitude Dispersions

The contribution of each error source and the combined RSS 3σ uncertainties are given in table VI. The values are presented in terms of final position and velocity in the topocentric reference frame (i.e., down range, cross range, altitude). The 3σ deviations of the down-range and cross-range positions are:

Down range, n. mi.	±3.08
Cross range, n. mi.	±51.74

As seen in table VI, the major contributor to the position uncertainties are the X and Z-gyro bias drift error. The G&N gyro bias drift errors are also the major contributors to the component velocity uncertainties.

From table VII the 3σ deviations in up-range, down-range, and cross-range positions are:

Up range, n. mi.	80.00
Down range, n. mi.	2.25
Cross range, n. mi.	±44.00

When the inertial flight-path angle approaches a 3σ steep value, the ability of the CM to reach the target becomes marginal. This is because of the long entry range and relatively low L/D. In the case involving a nominal L/D and a 3σ steep flight-path angle, the CM will fall short of the target by about 80 n. mi. If, however, the 3σ steep flight-path angle is combined with a 3σ low L/D the CM could fall short of the target by as much as 200 n. mi. In the case involving the 3σ low L/D, the CM can be controlled to the target, within the accuracy of the navigated position, with as much as a 1.1σ steep flight-path angle.

The ±0.18° uncertainty in entry flight-path angle does not significantly affect the performance of the reentry guidance. As the uncertainty is increased beyond this value the performance of the reentry guidance is rapidly degraded and miss distances caused by flight-path angle uncertainties are increased. This is discussed in more detail in references 12, 13, and 14.

3σ CM Total Heat and Heating Rate Deviations

For the nominal case, the total heat load is 41 134 Btu/ft^2, the first peak heating rate is 501.70 Btu/ft^2/sec, and the second peak heating rate is 58.75 Btu/ft^2/sec. The 3σ RSS deviations (table VIII) about the nominal case are:

Total heat load, Btu/ft^2	±3853.
First maximum heating rate, Btu/ft^2/sec.	±121.93
Second maximum heating rate, Btu/ft^2/sec	±12.12

3σ Load Factor Deviations

For the nominal case the first maximum load factor is 5.84 g, and the second maximum load factor is 3.96 g. The deviations (table VIII) in the load factor about the nominal case are:

First maximum load factor, g . . . . . ±2.13

Second maximum load factor, g . . . . . ±1.96

The results of this data indicate that the load factor deviations will not present an entry problem.

3σ G&N System Dispersions

It is evident that G&N system errors are the major contributors to the dispersions at entry interface and at 23 500-ft altitude. The greatest concern involving the heat shield test is the inertial flight-path angle deviation at entry interface (400 000 ft). The G&N Y-gyro drift error source accounts for 98 percent of this total deviation.

CONCLUSIONS

From the results of this study it was concluded that:

1. The SPS propellant reserves required are within the propellant budget limitations currently planned for the mission.

2. Because of the long entry range and low L/D the ability of the CM to reach the target becomes marginal as the entry inertial flight-path angle approaches the 3σ steep value. In this particular case, the CM will undershoot the target by about 80 n. mi. However, the heat shield test objectives will be met provided the CM does not undershoot the target by more than 100 n. mi. A combination of deviations involving a 3σ steep flight-path angle and 3σ low L/D can cause the CM to fall 200 n. mi. short of the target. However, this is not a 3σ case, and therefore the possibility of its occurring is very small.

The RSS 3σ ±0.18° uncertainty in the entry inertial flight-path angle does not significantly affect the landing accuracy at 23 500 ft. However, as the uncertainty increases beyond this the performance of the reentry guidance is rapidly degraded, and the dispersions at 23 500 ft are increased accordingly.

TABLE I.- SEQUENCE OF EVENTS

Event	Ground elapsed time, hr:min:sec	Spacecraft position			Inertial velocity vector		
		Longitude, deg	Geodetic latitude, deg	Altitude, ft	Velocity, fps	Flight-path angle, deg	Azimuth, deg
Lift-off	0:0	-80.6	28.6	90	1 341	0.0	90.0
Insertion	0:10:56	-55.5	32.6	627 816	25 568	-0.00	87.2
1st radar update	3:10:08	-88.0	32.5	675 303	25 549	-0.02	94.4
Injection	3:15:27	-60.6	27.2	1 045 324	35 581	6.77	108.7
1st SPS ignition	3:20:16	-37.9	17.5	3 255 979	33 916	18.64	117.9
1st SPS cutoff	3:24:39	-24.3	9.3	6 203 315	28 157	24.46	121.3
Apogee	6:21:53	44.7	-32.2	72 847 244	7 415	0.0	85.3
2nd radar update	9:08:30	88.6	2.8	14 281 182	23 548	-33.00	57.6
2nd SPS ignition	9:21:44	118.6	21.4	4 221 374	29 586	-20.78	64.9
2nd SPS cutoff	9:24:53	131.6	26.7	2 203 368	35 056	-17.52	70.5
Entry	9:28:09	155.0	31.9	400 000	36 500	-6.50	82.5
Drogue chute deployment	9:43:34	-157.14	27.4	23 500	1 485.9	-14.96	84.4

TABLE II.- EFFECT OF PERTINENT $\pm 3\sigma$ SYSTEM ERRORS ON SPS PROPELLANT CONSUMPTION

Error source	Propellant dispersion	
	Oxidizer, lb	Fuel, lb
Spacecraft G&N errors		
Y-gyro bias drift	85	42
X, Y and Z accelerometer bias	88	44
Launch vehicle G&N errors		
X[a] position	140	70
Y position	115	58
Z position	44	22
X velocity	130	65
Y velocity	20	10
Z velocity	64	32
Total required 3σ RSS propellant reserve	267	133

[a]X, Y, and Z corresponds to the earth-centered inertial plumbline coordinate system.

TABLE III.- SERVICE MODULE PROPELLANT USAGE

Usable propellant loaded, lb.	32 538.9
Consumed during first burn, lb.	-17 370.8
Remaining after first burn, lb.	15 168.1
Consumed during second burn, lb	-12 979.8
Remaining after second burn, lb	2 188.3
Total required 3σ propellant margin, lb	400.0
Mixture ratio tolerance, lb	754.
Loading tolerance, lb	164.
Operational reserves, lb.	328.
Propellant margin, lb	543.

TABLE IV.- $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT DEVIATIONS

[Two state vector updates]

(a) Time in seconds

CASE		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	7.587-01	7.651-01	1.192+02	1.196+02	1.081+02
BIAS	Y =	0.000	6.191-02	1.193+00	1.081+01	1.052+01	9.801+00
ERROR	Z =	0.000	2.424-01	3.149-01	8.315+00	9.575+00	1.608+01
ACCELEROMETER	X =	0.000	8.118-02	1.264+00	1.274+01	1.266+01	1.233+01
SCALE FACTOR	Y =	0.000	1.221-03	9.131-02	9.131-02	9.082-02	1.797-01
ERROR	Z =	0.000	7.935-03	5.859-03	5.859-03	2.246-02	1.953-01
GYRO BIAS	X =	0.000	0.000	1.226-01	6.123+00	5.711+00	5.357+00
DRIFT	Y =	0.000	4.272-03	9.766-04	5.400+01	5.441+01	5.846+01
ERROR	Z =	0.000	1.733-02	2.393-02	1.372+00	1.239+00	1.284+00
LNCH VEH PERFORMANCE	ERR	7.000+00	6.658+00	9.140+00	9.140+00	8.996+00	8.562+00
ROOT SUM SQUARE	=	7.000+00	6.706+00	9.342+00	1.318+02	1.334+02	1.250+02

(b) Inertial velocity in feet per second

CASE		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	2.005+01	7.020+02	3.315+01	1.443+01	1.052+00
BIAS	Y =	0.000	1.825+00	6.[illegible]7+01	8.590+00	7.765+00	3.467+00
ERROR	Z =	0.000	3.031+00	8.754+01	7.142+00	3.904+00	8.237+00
ACCELEROMETER	X =	0.000	2.165+00	8.167+01	6.392+00	2.460+00	4.146-01
SCALE FACTOR	Y =	0.000	3.516-02	7.039-01	9.282-01	7.319-01	2.861-02
ERROR	Z =	0.000	2.057-02	2.209+00	2.670+00	3.470+00	1.253+00
GYRO BIAS	X =	0.000	6.667-01	2.463+01	1.503+01	1.187+01	3.769+00
DRIFT	Y =	0.000	7.526+00	2.[illegible]1+02	4.219+01	5.284+01	2.736[illegible]
ERROR	Z =	0.000	1.365+01	6.327-01	1.827+01	1.403+01	1.613+00
LNCH VEH PERFORMANCE	ERR	1.321+02	1.276+02	3.[illegible]76+00	4.906+00	2.861+00	2.124-01
ROOT SUM SQUARE	=	1.321+02	1.301+02	7.720+02	6.261+01	5.862+01	2.906+01

TABLE IV.- $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT DEVIATIONS - Continued

[Two state vector updates]

(c) Inertial flight-path angle in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	1.185-02	1.096+00	3.012-03	5.028-02	4.831-02
BIAS	Y =	0.000	9.372-04	1.039-01	1.898-02	1.592-02	1.219-02
ERROR	Z =	0.000	2.000-[illegible]	1.476-01	5.837-02	2.137-02	2.271-02
ACCELEROMETER	X =	0.000	1.409-03	1.233-01	1.475-02	1.956-02	5.821-03
SCALE FACTOR	Y =	0.000	4.792-05	1.079-03	2.676-03	3.688-03	3.731-05
ERROR	Z =	0.000	7.830-04	3.943-03	8.771-03	1.275-02	5.225-03
GYRO BIAS	X =	0.000	4.751-03	3.414-02	4.700-02	5.908-02	5.237-02
DRIFT	Y =	0.000	5.093-02	3.976-01	1.646-01	3.111-01	5.561-01
ERROR	Z =	0.000	8.884-02	1.532-03	2.904-02	8.422-02	3.702-02
LNCH VEH PERFORMANCE	ERR	4.001-01	2.986-01	6.343-03	1.483-02	1.518-02	3.708-03
ROOT SUM SQUARE	=	4.001-01	3.165-01	1.161+00	1.839-01	3.334-01	5.623-01

(d) Inertial azimuth in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	3.460-03	2.009-01	2.199-01	3.525-01	1.586-01
BIAS	Y =	0.000	2.926-02	1.083-02	3.753-02	6.427-02	9.833-02
ERROR	Z =	0.000	6.428-04	2.297-02	1.675-01	1.922-01	1.146-01
ACCELEROMETER	X =	0.000	7.076-04	1.744-02	2.004-02	2.718-02	1.910-02
SCALE FACTOR	Y =	0.000	4.015-04	2.146-04	2.263-03	3.528-03	3.843-04
ERROR	Z =	0.000	3.052-05	6.652-04	9.156-03	1.477-02	9.025-03
GYRO BIAS	X =	0.000	1.787-02	1.174-02	6.141-02	1.892-01	1.779-01
DRIFT	Y =	0.000	5.674-03	4.966-02	2.180-01	3.150-01	6.995-01
ERROR	Z =	0.000	3.937-02	3.510-02	6.394-02	5.315-02	3.715-02
LNCH VEH PERFORMANCE	ERR	5.466-01	3.921-01	2.957-01	2.939-01	2.626-01	1.380-01
ROOT SUM SQUARE	=	5.466-01	3.957-01	3.641-01	4.688-01	6.072-01	7.675-01

TABLE IV.- $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT DEVIATIONS - Continued

[Two state vector updates]

(e) Geodetic latitude in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	2.093-02	1.943+00	2.729-01	2.526-01	3.647-02
BIAS	Y =	0.000	5.342-03	2.026-01	1.806-03	2.313-05	1.608-02
ERROR	Z =	0.000	7.618-03	2.855-01	2.005-01	1.435-01	2.050-02
ACCELEROMETER	X =	0.000	2.233-03	2.236-01	2.414-02	1.938-02	3.993-03
SCALE FACTOR	Y =	0.000	7.474-05	1.995-03	3.325-03	3.165-03	6.056-04
ERROR	Z =	0.000	2.534-04	7.833-03	1.092-02	1.025-02	1.667-03
GYRO BIAS	X =	0.000	1.617-03	6.519-02	4.723-02	1.561-02	5.861-02
DRIFT	Y =	0.000	2.713-03	6.811-01	2.686-01	2.456-01	1.607-01
ERROR	Z =	0.000	3.759-04	1.248+00	4.301-01	9.368-03	7.028-03
LNCH VEH PERFORMANCE	ERR	3.430-01	3.358-01	3.736-02	1.231-01	1.865-01	2.722-01
ROOT SUM SQUARE	=	3.430-01	3.366-01	2.444+00	6.245-01	4.218-01	3.245-01

(f) Longitude in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	3.211-02	-3.066+00	1.334-01	3.768-01	4.723-01
BIAS	Y =	0.000	1.837-03	-2.870-01	2.621-02	1.480-02	5.580-02
ERROR	Z =	0.000	1.225-02	4.525-01	4.950-01	4.928-01	1.328-01
ACCELEROMETER	X =	0.000	3.429-03	-3.464-01	1.837-03	7.998-03	1.521-02
SCALE FACTOR	Y =	0.000	3.076-05	-3.155-03	7.022-03	9.043-03	1.879-03
ERROR	Z =	0.000	4.098-04	-1.236-02	2.503-02	3.263-02	1.781-02
GYRO BIAS	X =	0.000	1.475-03	-6.633-02	1.237-01	1.300-01	1.256-01
DRIFT	Y =	0.000	3.282-03	1.074+00	3.769-01	5.247-01	1.116+00
ERROR	Z =	0.000	4.326-03	4.172-03	1.062-01	1.553-01	7.241-02
LNCH VEH PERFORMANCE	ERR	1.269+00	9.653-01	-2.078-01	9.840-02	9.088-02	1.649-01
ROOT SUM SQUARE	=	1.269+00	9.660-01	3.624+00	6.651-01	8.433-01	1.244+00

TABLE IV.- ±3σ SPACECRAFT POWERED FLIGHT DEVIATIONS - Continued

[Two state vector updates]

(g) Altitude in feet

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	9.083+03	1.449+06	2.679+04	5.391+04	0.000
BIAS	Y =	0.000	8.690+02	1.433+05	9.852+03	5.538+03	0.000
ERROR	Z =	0.000	1.694+03	1.867+05	1.550+04	6.037+03	0.000
ACCELEROMETER	X =	0.000	9.795+02	1.724+05	6.590+03	4.371+03	0.000
SCALE FACTOR	Y =	0.000	1.300+01	1.522+03	1.236+03	9.347+02	0.000
ERROR	Z =	0.000	5.025+01	4.693+03	3.589+03	2.879+03	0.000
GYRO BIAS	X =	0.000	2.352+02	5.224+04	1.932+04	1.053+04	0.000
DRIFT	Y =	0.000	2.479+03	6.267+05	4.844+04	3.235+04	0.000
ERROR	Z =	0.000	3.124+02	1.120+03	1.995+04	1.729+04	0.000
LNCH VEH PERFORMANCE	ERR	1.613+05	1.887+05	8.324+03	6.731+03	4.020+03	0.000
ROOT SUM SQUARE	=	1.613+05	1.887+05	1.961+06	6.539+04	6.542+04	0.000

(h) Apogee altitude in nautical miles.

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	2.173+02	2.164+02	2.185+02	0.000	0.000
BIAS	Y =	0.000	5.41[illegible]+00	5.263+00	5.465+00	0.000	0.000
ERROR	Z =	0.000	3.624.+01	3.502+01	3.574+01	0.000	0.000
ACCELEROMETER	X =	0.000	6.393+00	6.211+00	6.391+00	0.000	0.000
SCALE FACTOR	Y =	0.000	1.069-01	1.053-01	2.576-02	0.000	0.000
ERROR	Z =	0.000	2.577-01	2.523-01	2.206-01	0.000	0.000
GYRO BIAS	X =	0.000	1.842+00	1.788+00	1.827+00	0.000	0.000
DRIFT	Y =	0.000	2.160+01	2.096+01	2.163+01	0.000	0.000
ERROR	Z =	0.000	3.257+00	3.135+00	3.506+00	0.000	0.000
LNCH VEH PERFORMANCE	ERR	0.000	2.289-01	5.933-01	4.336-01	0.000	0.000
ROOT SUM SQUARE	=	0.000	2.186+02	2.204+02	2.226+02	0.000	0.000

TABLE IV.- ±3σ SPACECRAFT POWERED FLIGHT DEVIATIONS - Concluded

[Two state vector updates]

(i) Perigee altitude in nautical miles

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	1.027+00	1.123+00	1.013+00	0.000	0.000
BIAS	Y =	0.000	6.176-02	5.233-02	6.514-02	0.000	0.000
ERROR	Z =	0.000	1.443+00	1.432+00	1.431+00	0.000	0.000
ACCELEROMETER	X =	0.000	1.199-01	1.300-01	1.167-01	0.000	0.000
SCALE FACTOR	Y =	0.000	4.050-03	3.885-03	4.694-04	0.000	0.000
ERROR	Z =	0.000	5.694-02	5.702-02	5.764-02	0.000	0.000
GYRO BIAS	X =	0.000	3.512-01	3.545-01	3.428-01	0.000	0.000
DRIFT	Y =	0.000	3.833+00	3.866+00	3.827+00	0.000	0.000
ERROR	Z =	0.000	9.026-01	1.359+00	1.286+00	0.000	0.000
LNCH VEH PERFORMANCE	ERR	0.000	5.816-02	2.725-02	2.643-02	0.000	0.000
ROOT SUM SQUARE	=	0.000	3.942+00	4.506+00	4.421+00	0.000	0.000

(j) Time of free fall in seconds

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	3.012+02	2.993+02	1.842+01	2.284+00	0.000
BIAS	Y =	0.000	9.862+00	1.097+01	1.064+01	7.108-01	0.000
ERROR	Z =	0.000	2.527+02	2.537+02	1.164+01	7.742-01	0.000
ACCELEROMETER	X =	0.000	1.234+01	1.351+01	5.292-01	3.383-01	0.000
SCALE FACTOR	Y =	0.000	2.151-01	1.231-01	1.179-01	6.694-02	0.000
ERROR	Z =	0.000	2.686-01	2.646-01	2.636-01	2.222-01	0.000
GYRO BIAS	X =	0.000	4.748+00	4.628+00	1.402+00	3.745-01	0.000
DRIFT	Y =	0.000	2.623+01	2.631+01	2.134+00	3.596+00	0.000
ERROR	Z =	0.000	3.041+00	3.142+00	3.682+00	1.646+00	0.000
LNCH VEH PERFORMANCE	ERR	0.000	1.580+01	5.910-01	5.913-01	3.415-01	0.000
ROOT SUM SQUARE	=	0.000	3.947+02	3.936+02	2.224+01	4.735+00	0.000

TABLE V.- $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT UNCERTAINTIES

[Two state vector updates]

(a) Inertial velocity in feet per second

CASE		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	3.430+00	1.339+01	0.000	2.180-02	1.821+00	3.265 +00
BIAS	Y =	3.647-01	1.220+00	0.000	1.100-03	4.913-01	2.869 +00
ERROR	Z =	6.919-01	3.586-01	0.000	2.052+00	3.314+00	2.674 +00
ACCELEROMETER	X =	3.418-03	1.634+00	0.000	3.735-02	5.596-01	2.843 +00
SCALE FACTOR	Y =	3.418-03	4.492-02	0.000	3.589-02	8.740-02	2.850 +00
ERROR	Z =	3.418-03	4.028-02	0.000	3.955-02	1.344+00	2.794 +00
GYRO BIAS	X =	3.418-03	6.909-01	0.000	1.685-02	-3.525+00	2.860 +00
DRIFT	Y =	3.418-03	7.523+00	0.000	1.150-01	2.325+01	2.858 +00
ERROR	Z =	3.418-03	1.358+00	0.000	1.397-02	2.381+00	2.855 +00
LNCH VEH PERFORMANCE	ERR	5.859-03	3.320-02	0.000	3.760-02	5.322-02	2.803 +00
ROOT SUM SQUARE	=	3.518+00	1.558+01		2.055+00	2.399+01	9.072+00

(b) Inertial flight-path angle in degrees

CASE		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	1.219-03	2.189-04	0.000	5.680-02	6.338-02	1.076-01
BIAS	Y =	4.563-04	2.684-03	0.000	4.721-03	4.371-03	4.288-02
ERROR	Z =	5.818-03	2.811-02	0.000	1.029-02	2.002-02	6.373-02
ACCELEROMETER	X =	6.437-06	3.362-05	0.000	5.245-05	1.884-03	3.684-02
SCALE FACTOR	Y =	6.437-06	5.484-05	0.000	4.578-05	6.366-05	3.870-02
ERROR	Z =	6.437-06	1.005-03	0.000	4.411-05	9.220-04	3.944-02
GYRO BIAS	X =	6.437-06	5.085-03	0.000	8.750-05	1.373-02	2.336-02
DRIFT	Y =	6.437-06	5.559-02	0.000	2.453-04	1.443-01	1.780-01
ERROR	Z =	6.437-06	9.683-02	0.000	1.500-04	9.520-02	1.500-02
LNCH VEH PERFORMANCE	ERR	2.384-06	1.168-05	0.000	2.074-05	2.623-06	3.811-02
ROOT SUM SQUARE	=	5.962-03	1.153-01	0.000	5.792-02	1.858-01	1.821-01

TABLE V. - $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT UNCERTAINTIES - Continued

[Two state vector updates]

(c) Inertial azimuth in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	5.255-04	2.530-03	0.000	4.554-03	4.522-03	8.943-03
BIAS	Y =	6.231-03	2.961-02	0.000	6.069-02	5.788-02	3.231-02
ERROR	Z =	6.151-04	2.871-03	0.000	5.994-03	5.678-03	2.779-03
ACCELEROMETER	X =	1.240-05	2.623-04	0.000	7.534-05	2.575-04	4.839-03
SCALE FACTOR	Y =	1.240-05	4.539-04	0.000	7.629-05	3.357-04	5.445-03
ERROR	Z =	1.240-05	5.150-05	0.000	7.439-05	3.061-04	4.789-03
GYRO BIAS	X =	1.240-05	1.908-02	0.000	1.984-04	1.207-01	1.085-01
DRIFT	Y =	1.240-05	5.486-03	0.000	3.052-05	2.210-02	2.382-02
ERROR	Z =	1.240-05	4.265-02	0.000	1.221-04	8.588-02	7.721-02
LNCH VEH PERFORMANCE	ERR	1.526-05	5.150-05	0.000	9.060-05	1.030-04	4.821-03
ROOT SUM SQUARE	=	6.284-03	5.572-02	0.000	6.115-02	1.607-01	1.398-01

(d) Geodetic latitude in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	1.979-04	2.680-03	0.000	2.721-03	1.023-03	1.136-02
BIAS	Y =	3.216-04	4.063-03	0.000	2.666-02	4.476-02	7.981-02
ERROR	Z =	1.574-05	6.043-04	0.000	1.025-02	1.215-02	9.105-03
ACCELEROMETER	X =	4.053-06	2.036-04	0.000	2.074-05	2.050-05	1.067-02
SCALE FACTOR	Y =	4.053-06	5.841-05	0.000	1.812-05	7.868-05	1.035-02
ERROR	Z =	4.053-06	3.493-05	0.000	2.096-05	1.018-04	1.052-02
GYRO BIAS	X =	4.053-06	1.687-03	0.000	2.217-05	1.404-02	4.980-02
DRIFT	Y =	4.053-06	2.973-03	0.000	1.717-05	8.171-03	2.728-03
ERROR	Z =	4.053-06	1.956-04	0.000	1.693-05	1.271-02	3.449-02
LNCH VEH PERFORMANCE	ERR	9.061-06	2.193-05	0.000	1.955-05	2.074-05	1.085-02
ROOT SUM SQUARE	=	3.782-04	5.985-03	0.000	2.869-02	5.049-02	1.035-01

TABLE V. - $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT UNCERTAINTIES - Continued

[Two state vector updates]

(e) Longitude in degrees

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	3.314-04	3.678-03	0.000	1.666-03	6.830-03	1.597-01
BIAS	Y =	2.117-04	2.746-03	0.000	1.056-02	1.175-02	1.050-01
ERROR	Z =	6.485-05	1.967-03	0.000	3.020-02	5.259-02	3.115-02
ACCELEROMETER	X =	7.153-06	3.312-04	0.000	1.030-04	1.907-04	1.028-01
SCALE FACTOR	Y =	7.153-06	5.484-05	0.000	1.097-04	1.354-04	1.030-01
ERROR	Z =	7.153-06	2.050-05	0.000	1.163-04	4.921-04	1.017-01
GYRO BIAS	X =	7.153-06	1.557-03	0.000	1.097-04	6.989-03	1.157-01
DRIFT	Y =	7.153-06	3.600-03	0.000	1.297-04	1.040-02	9.380-02
ERROR	Z =	7.153-06	9.446-03	0.000	1.431-04	3.601-03	9.804-02
LNCH VEH PERFORMANCE	ERR	1.383-05	4.435-05	0.000	8.202-05	1.030-04	1.038-01
ROOT SUM SQUARE	=	3.992-04	1.139-02	0.000	3.203-02	5.587-02	3.340-01

(f) Altitude in feet

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	1.125+01	9.017+02	0.000	2.243+01	1.672+02	3.171+03
BIAS	Y =	8.500+00	2.633+02	0.000	1.288+01	7.284+01	7.913+02
ERROR	Z =	1.617+02	2.176+03	0.000	4.184+01	1.287+02	1.872+03
ACCELEROMETER	X =	5.000+00	5.425+01	0.000	3.750+01	1.482+02	6.123+02
SCALE FACTOR	Y =	5.000+00	2.025+01	0.000	3.225+01	4.150+01	6.468+02
ERROR	Z =	5.000+00	3.225+01	0.000	3.425+01	2.175+01	6.691+02
GYRO BIAS	X =	5.000+00	2.692+02	0.000	3.700+01	6.320+02	3.646+02
DRIFT	Y =	5.000+00	2.497+03	0.000	3.500+01	6.709+03	3.760+03
ERROR	Z =	5.000+00	4.362+03	0.000	3.550+01	4.437+03	2.684+03
LNCH VEH PERFORMANCE	ERR	8.250+00	2.600+01	0.000	3.700+01	4.025+01	6.486+02
ROOT SUM SQUARE	=	1.630+02	5.565+03	0.000	1.056+02	8.073+03	6.109+03

TABLE V. - $\pm 3\sigma$ SPACECRAFT POWERED FLIGHT DISPERSIONS UNCERTAINTIES - Continued

[Two state vector updates]

(g) Apogee altitude in nautical miles

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	5.417+03	2.176+02	0.000	8.111+00	0.000	0.000
BIAS	Y =	5.778+02	6.121+00	0.000	1.270+01	0.000	0.000
ERROR	Z =	1.245+03	9.430+00	0.000	1.370+01	0.000	0.000
ACCELEROMETER	X =	5.703-01	3.614+01	0.000	4.968-02	0.000	0.000
SCALE FACTOR	Y =	5.703-01	1.381-01	0.000	5.017-02	0.000	0.000
ERROR	Z =	5.703-01	3.286-01	0.000	6.482-02	0.000	0.000
GYRO BIAS	X =	5.703-01	1.859+00	0.000	4.675-02	0.000	0.000
DRIFT	Y =	5.703-01	2.152+01	0.000	3.822-01	0.000	0.000
ERROR	Z =	5.703-01	3.164+00	0.000	2.458-01	0.000	0.000
LNCH VEH PERFORMANCE	ERR	2.422-01	7.092-02	0.000	4.686-02	0.000	0.000
ROOT SUM SQUARE	=	5.588+03	2.218+02	0.000	2.037+01	0.000	0.000

(h) Perigee altitude in nautical miles

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	1.138-01	1.028+00	0.000	1.376-01	0.000	0.000
BIAS	Y =	1.071-02	5.749-02	0.000	8.691-04	0.000	0.000
ERROR	Z =	2.012-01	1.442+00	0.000	6.273-02	0.000	0.000
ACCELEROMETER	X =	1.440-03	1.275-01	0.000	1.330-03	0.000	0.000
SCALE FACTOR	Y =	1.440-03	4.836-03	0.000	8.779-04	0.000	0.000
ERROR	Z =	1.440-03	5.950-02	0.000	1.779-04	0.000	0.000
GYRO BIAS	X =	1.440-03	3.396-01	0.000	4.666-03	0.000	0.000
DRIFT	Y =	1.440-03	3.835+00	0.000	2.165-02	0.000	0.000
ERROR	Z =	1.440-03	6.700-01	0.000	1.492-02	0.000	0.000
LNCH VEH PERFORMANCE	ERR	2.880-03	4.159-03	0.000	5.488-04	0.000	0.000
ROOT SUM SQUARE	=	2.314-01	4.353+00	0.000	1.459-01	0.000	0.000

TALBE V. -±3σ SPACECRAFT POWERED FLIGHT DISPERSIONS UNCERTAINTIES - Concluded

[Two state vector updates]

(i) Time of free fall in seconds

CASE (SPACECRAFT GUIDANCE AND NAVIGATION ERRORS)		FIRST SPS IGNITION	FIRST SPS CUT OFF	CALCULATE TFF	SECOND SPS IGNITION	SECOND SPS CUT OFF	ENTRY
ACCELEROMETER	X =	0.000	3.034+02	0.000	2.361-02	7.304-01	0.000
BIAS	Y =	0.000	1.139+00	0.000	4.103-03	3.506-01	0.000
ERROR	Z =	0.000	2.482+02	0.000	5.038-02	2.371-01	0.000
ACCELEROMETER	X =	0.000	1.363+01	0.000	5.157-03	9.218-02	0.000
SCALE FACTOR	Y =	0.000	2.375-01	0.000	4.208-03	6.870-03	0.000
ERROR	Z =	0.000	3.813-01	0.000	3.555-03	4.681-02	0.000
GYRO BIAS	X =	0.000	4.740+00	0.000	8.797-03	6.792-01	0.000
DRIFT	Y =	0.000	2.623+01	0.000	2.340-02	7.322+00	0.000
ERROR	Z =	0.000	1.309+00	0.000	1.618-02	4.698+00	0.000
LNCH VEH PERFORMANCE	ERR	0.000	1.208-01	0.000	4.768-03	6.006-03	0.000
ROOT SUM SQUARE	=	0.000	3.929+02	0.000	6.480-02	8.756+00	0.000

TABLE VI.- $\pm 3\sigma$ UNCERTAINTIES AT 23 500-FT ALTITUDE

Error source	Final position errors, n. mi.			Final velocity errors, fps		
	Down range	Cross range	Altitude	Down range	Cross range	Altitude
G&N system errors						
Nominal	0.25	0.10	0.12	3.15	8.32	5.13
X-gyro drift	.152	25.16	1.34	4.71	206.11	29.91
Y-gyro drift	2.13	4.60	33.70	150.66	57.92	771.78
Z-gyro drift	2.18	44.98	2.97	34.38	728.50	36.89
Entry errors						
Small L/D (0.304)	0.01	0.02	0.04	1.05	1.58	.36
60° north winter atmosphere	0.29	0.11	0.13	13.1	4.77	7.01
Root sum squared error	3.08	51.74	33.86	155.19	759.37	773.29

TABLE VII.- $\pm 3\sigma$ DEVIATIONS AT 23 500-FT ALTITUDE

Error source	Final position errors, n. mi.				Final velocity errors, fps		
	Up range	Down range	Cross range	Altitude	Down range	Cross range	Altitude
G&N system errors							
X-gyro drift	0.53	.53	23.66	0	72.14	2.78	11.48
Y-gyro drift	80.00	1.87	5.34	0	65.00	96.06	20.72
Z-gyro drift	1.10	1.10	36.71	0	138.78	4.44	4.85
Entry errors							
Small L/D (0.304)	0.24	0.24	0.53	0	14.82	6.63	3.07
60° north winter atmosphere	0.11	0.11	0.25	0	15.56	2.06	5.59
Root sum squared error	80.01	2.25	44.00	0	170.73	96.45	24.42

TABLE VIII.- $\pm 3\sigma$ DEVIATIONS FROM THE NOMINAL HEATING PARAMETERS

Error source	First maximum heating rate, (B.t.u./ft^2)/sec	Second maximum heating rate (B.t.u./ft^2)/sec	Total heat load, B.t.u./ft^2	First maximum load factor, g	Second maximum load factor, g
X-gyro drift	5.11	8.03	831	0.15	0.98
Y-gyro drift	117.70	2.05	2013	2.04	1.14
Z-gyro drift	7.80	0.05	418	0.18	0.28
Small L/D (0.304)	25.41	8.56	2869	0.33	1.24
60° north winter atmosphere	16.73	2.24	1280	0.45	0.13
Root sum squared error	121.93	12.12	3853	2.13	1.96

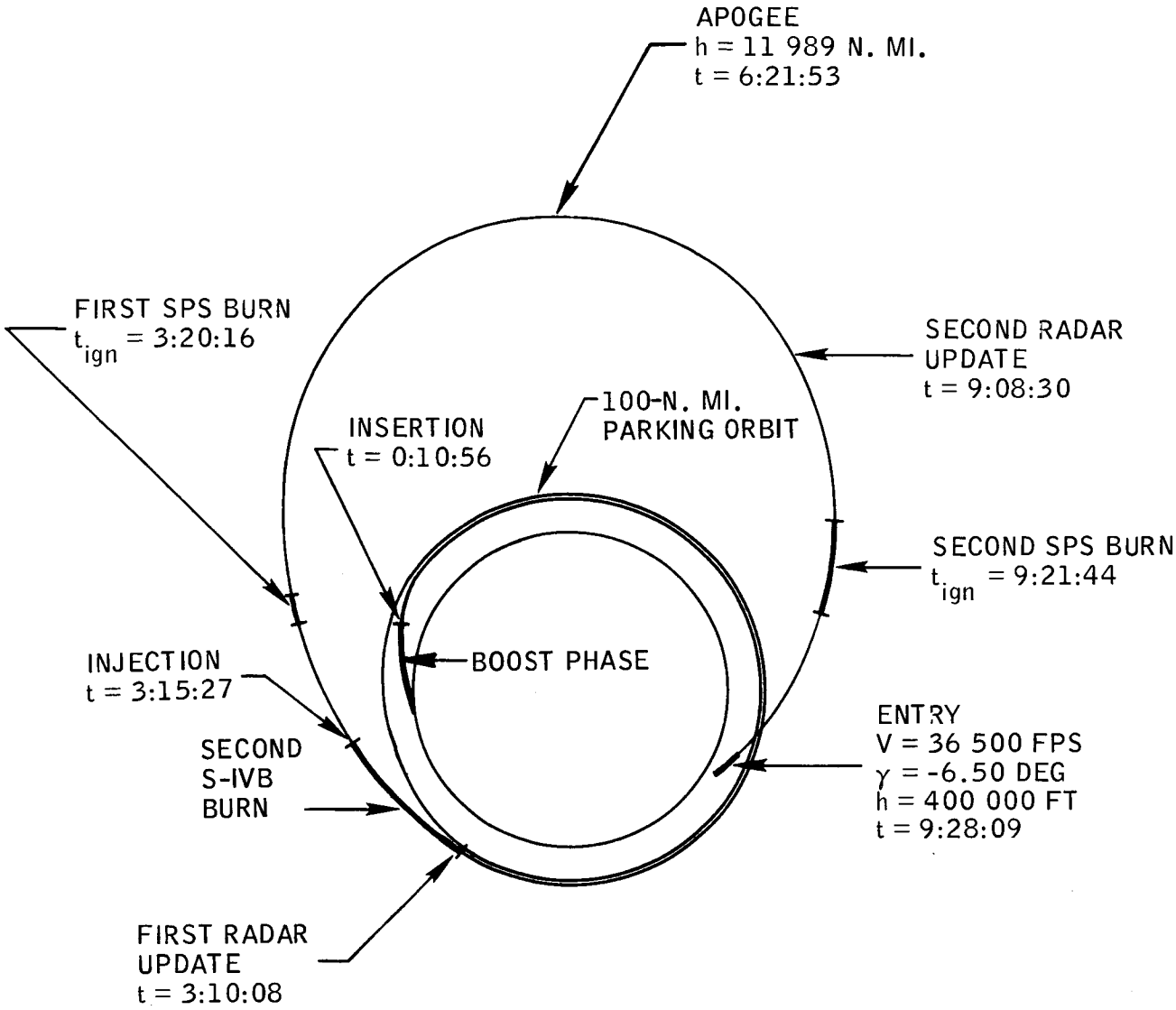
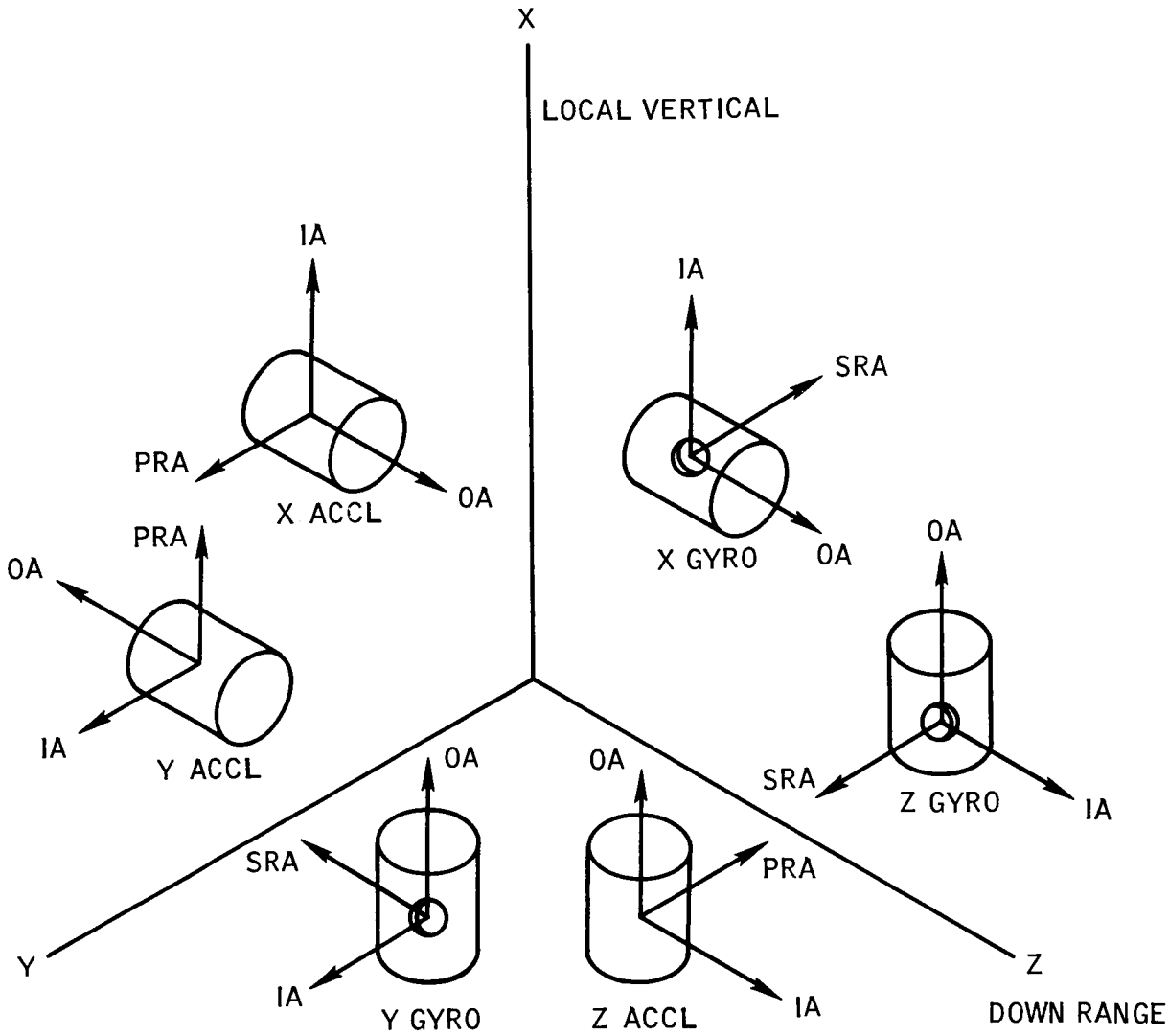


Figure 1.- Apollo 6 mission profile.



OA = OUTPUT AXIS
IA = INPUT AXIS
SRA = SPIN REFERENCE AXIS
PRA = PENDULOUS REFERENCE AXIS

Figure 2. - IMU stable member.

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