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FLIGHT TEST ENGINEERING ANALYSIS REPORT

FOR

THE HEXAGON PROGRAM SATELLITE VEHICLE NUMBER NINE ~~(S)~~

Contract [redacted] ~~(S)~~

Prepared and Submitted by the
Satellite Vehicle Integrating Contractor

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FOREWORD

This report describes the performance of the ninth HEXAGON Program Satellite Vehicle (SV-9). The vehicle was launched 29 October 1974, and after a 129 day primary mission and an 11 day solo mission, was deboosted on Rev 2274 on 18 March 1975.

This report does not explicitly cover the solo mission; however, results from solo are used as appropriate when they contribute substantially to the understanding of primary mission events.

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SECTION I

SUMMARY

1.1 INTRODUCTION

The ninth HEXAGON Satellite Vehicle (SV-9) was placed in a 85.5 X 156 nm orbit by the Titan IIID booster on 29 October 1974. The nominal orbit, 87.5 X 156 nm, was established on the first orbit adjust. Ascent events were nominal and proper stabilization of the SV allowed the initiation of deployment of the solar arrays at the first station contact, INDI. The SSU (-Y) Subsatellite was properly ejected on Rev 13. The S3-1 Subsatellite (+Y) was properly ejected on Rev 15. The panoramic camera operated throughout the mission and its RVs were recovered on Revs 310, 894, 1364 and 2094 which occurred on Mission Days 20, 56, 85 and 130. All four RVs were caught in the air. All of the film was transported into the RVs. There were no anomalies to report in the Panoramic Camera System. All the mapping camera operations were normal and 100 percent of the film was transported to RV 5 which was aerially recovered on Rev 958. Solo tests were run and the SV was deorbited on Rev 2274 (during Mission Day 141).

SECTION II
MISSION OVERVIEW

2.1 PREFLIGHT PLANNING

Mission 1209 was the third of the HEXAGON Block II vehicles and panoramic camera systems. No special flight experiments were scheduled except for the installation of four pressure transducers inside the shroud in an attempt to measure the pressure within the shroud at the moment of shroud separation. All four RCS Tanks were filled and supplied the thrusters throughout the flight with no transfer of fuel from the OA Tank.

2.2 PREFLIGHT CONSTRAINTS

The Mission 1209 orbit was designed to:

A. Maintain solar angle (Beta) within -8° to +30° for the planned 130 days.

B. Have orbit adjusts to occur on a three-day cycle with every third orbit adjust (OA) to be a positive and negative burn for close control of perigee argument.

2.2.1 Panoramic Camera System Constraints

The following were the constraints imposed on the panoramic cameras:

A. Rewind velocity limited to 5 inches/second.

B. No 30° scans at ±45° scan centers.

2.3 LAUNCH BASE

The SV was delivered to the launch pad and mated to the BV on 9 October 1974. The vehicle was launched on 29 October 1974 at 11:30:00.7 PST at the opening of the launch window.

2.4 ASCENT

The BV successfully injected the SV into an 85.50 X 156.00 nm orbit. The targeted and achieved orbits and deviations were as follows:

	Targeted	Achieved	Deviation
Apogee Altitude (nm)	87.270	85.502	-1.768
Perigee Altitude (nm)	152.535	156.002	+3.467
Period (min:sec)	88:53.82	88:57.60	0:03.78
Eccentricity	0.009238	0.009909	+0.000671
Argument of Perigee (degrees)	158.585	168.557	+9.972
Inclination (degrees)	96.788	96.692	-0.096

The deviations except inclination were corrected at the first OA on Rev 63.

2.5 ORBIT AND RECOVERY

2.5.1 1209-1 (Nineteen Days Duration)

On SV-9, the DBS Antenna was deployed 2000 seconds after liftoff. Solar array deployment was executed over INDI on Rev 1 with normal deployment and erection. Since the solar arrays were at the optimum position of +18° for the initial beta angle of +9.0 degrees no repositioning was necessary. The SSU Subsatellite was ejected on Rev 13 and the S3-1 Subsatellite was ejected on Rev 15.

Operational photography began on Rev 6 following successful completion of constant velocity and health checks. Approximately 29,500 feet of film per camera (including the pre-launch footage on the take-up) were exposed and stowed in RV-1.

Post flight analysis (PFA) of the recovered film showed the overall quality of the acquired photography to range from very good to poor with the majority rated as fair to good. The poor imagery was attributed to hazy or inclement weather. There was a definite preference for the aft camera imagery when compared to the forward.

On-Orbit Adjust Assembly (OOAA) adjustments were established for the aft looking camera to correct a skew angle error determined by subjective analysis.

2.5.2 <u>1209-2 (Thirty-Six Days Duration)</u>

Normal operational photography continued throughout this segment. The PFA determined OOAA adjustment on the aft camera was implemented on Rev 350. Approximately 29,500 feet of forward-looking camera film and 26,900 feet of aft-looking camera film were exposed and stowed in RV-2. This included 5650 feet of SO-255 (color) on the aft-looking camera.

PFA showed overall quality of acquired photography to range from very good to poor but with the majority rated as good. Aft-looking camera performance continued to be better than that of the forward-looking camera. The poor imagery for the most part was attributed to hazy or inclement weather. The quality of the color photography (SO-255) compared to previous color ranged from good to fair with the majority rated good. The SO-255 material had an apparent underexposure of 1/3 to 1/2 stop, and a three count (0.10 log E) exposure increase was made on RV-3 for the remaining SO-255 material. In addition, a one command step advance of OOAA on the forward-looking camera was determined to be required.

2.5.3 <u>1209-3 (Twenty-Nine Days Duration)</u>

The Panoramic Camera System exhibited nominal performance throughout RV-3 with no anomalies or malfunctions. The OOAA adjustment of the forward-looking camera indicated by PFA was made on Rev 979.

Approximately 28,510 feet of forward-looking camera film and 27,500 feet of aft-looking camera film were exposed and stowed in RV-3. The aft-looking camera included 3400 feet of SO 130 (IR) and 2500 feet of SO-255 (color) film.

PFA showed overall quality of the acquired photography to range from very good to poor with the majority rated as fair. The quality was comparable to previous winter missions.

2.5.4 <u>1209-4 (Forty-Five Days Duration)</u>

Normal photographic operation was obtained throughout this segment with no anomalies or malfunctions.

Forward and aft-looking camera film depletion occurred on Rev 2086 and Rev 2090 respectively.

Approximately 29,240 feet of forward-looking camera film and approximately 27,750 feet of aft-looking camera film were exposed and stowed in RV-4.

PFA showed image quality to range from very good to poor with the majority rated fair. The quality was comparable to previous winter missions and the best of RV-4 indicated a slight decrease from the best of RV-3.

2.6 ANOMALY SUMMARY

Significant anomalies are listed chronologically in Table 2-1. The list includes a brief description of the anomaly and its effect on the mission. A more detailed discussion can be obtained in the reference paragraphs.

TABLE 2-1

SUMMARY OF ANOMALIES

Day	Description	Impact	Reference Paragraph
1	REA Temperature Monitors erratic.	Basis for thrust comparison changed. Redesign and improved inspection for Block III.	3.3.4
2	Spurious command to tape recorder.	Command sequence corrected. No data lost.	3.7.3
47	ECS executed two spurious commands.	Weak uplink signal at 1.2° elevation angle. Decoders turned off 30 - 60 seconds before fade rest of mission.	3.7.6.1
56	Damaged RV-2 parachute load loop.	No mission impact. Hook tine split webbing.	5.3
64	High roll rate during Camera A ops with one OB.	REA 3 and REA 7 unbalance induced pitch which suppressed roll control. Single OB mono ops curtailed until transfer to RCS 2.	3.2.3
69	HSA inhibit one side. Roll and yaw diverging attitude.	Transfer to ACS 2. Failed HSA is Block I design. Corrective modifications are in Block II units.	3.2.3 and 3.2.6
79	REA 7 valve had transient leak.	Self healed in one rev. No mission impact.	3.3.4
81	REA 2, 5, 7 and 8 seriously degraded.	Transfer to RCS 2. Thruster degradation was as expected.	3.3.1
85	RV 3 Main Parachute hit - parachute torn.	Endangered RV recovery. No other mission impact.	5.3
96	Horizon Sensor moon intercept.	Moon induced signal produced pitch. No mission impact.	3.2.9
136	IRA failure to start.	Solo test. Failure under study.	3.2.7
140	FCEA failure.	Vehicle tumbled. Deboost with Lifeboat II. Failure under study.	3.2.8

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SECTION III
SATELLITE BASIC ASSEMBLY SUBSYSTEMS

3.1 INTRODUCTION

The following paragraphs summarize the performance of the Satellite Basic Assembly (SBA) subsystems as verified from flight data.

3.2 ATTITUDE CONTROL SYSTEM (ACS)

The ACS performed within specification. Low force level RCS thruster operation and a PACS HSA inhibit impacted payload operations as described in Paragraph 3.2.3. An IRA and FCEA failure which occurred during solo are described in Paragraphs 3.2.7 and 3.2.8.

3.2.1 BV/SV Separation

BV/SV separation was completed at approximately 534.0 seconds vehicle time (vehicle time started 67.26 seconds prior to liftoff). Master Clear Off, which enables the pitch, roll and yaw integrators to accumulate angle, was at 501.8 seconds and SECO, which terminates BV attitude control, occurred at 522.0 seconds vehicle time. The SV attitude changes from SECO to BV/SV separation and the attitude and rates as measured at BV/SV separation are shown in Table 3-1. This table also presents the times in which the SV attitudes and rates came back within the specified limits following BV/SV separation.

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TABLE 3-1

BOOSTER VEHICLE/SATELLITE VEHICLE (BV/SV) SEPARATION

Axes	RATE AND ATTITUDE AT BV/SV SEPARATION						CAPTURE			
	RATE (deg/sec)		ATTITUDE (DEGREES)				ATTITUDE		RATE	
			H/S at Sep		SECO to Sep					
	Spec	Actual	Spec	Actual	Spec	Actual (5) H/S/ Integrator	Spec (1) (deg)	Actual (2) (Time in sec)	Specified (3) (deg/sec)	Actual (Time in sec)
Pitch	±0.752	-0.203	+13.0 to -21.7	-0.984	±3.5	-0.76/-1.89	±0.70	(6)	±0.014	(6)
Roll	±0.786	-0.240	±10.6	3.20	±3.5	1.44/1.81	±0.70	(6)	±0.021	(6)
Yaw	±0.752	0.125	+11.1 to -11.4	---	+4.5 to -3.5	---/1.68	±0.64	(6)	±0.014	(6)

(1) Attitude in degrees to be achieved in 1500 seconds.
(2) Actual time required to achieve specified attitude (switch to fine mode + settling time).
(3) Rate in degrees/second to be achieved in 1500 seconds.
(4) Actual time required to achieve specified rate.
(5) Relative to the local horizontal.
(6) Nominal performance indicating the pointing requirements are satisfied was observed at a nominal settling time of 520 seconds after the commanded switch to fine mode (673 seconds after separation). The total 1193 seconds is well within the spec of 1500 seconds and no closer study was performed.

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3.2.2 Subsatellite/SV Separations

3.2.2.1 SSU/SV Separation

SSU/SV separation events of Rev 13 were as follows:

Event	Vehicle Time (sec)
Start Negative Yaw Maneuver	70317.2
Stop Yaw	70355.4
Separation	70415.2
Start Positive Yaw Maneuver	70432.4
Stop Yaw	70470.6

The ACS parameters just prior to the instant of separation (70415.2 seconds vehicle time) are presented in Table 3-2.

TABLE 3-2

RATE AND ATTITUDE PARAMETERS AT SSU/SV SEPARATION

Parameter	Specified	Actual
Pitch Horizon Sensor	+1.0 deg	+0.12
Roll Horizon Sensor	±1.0 deg	-0.80
Roll Integrator	---	-0.16
Yaw Integrator	---	+0.02
Pitch Integrator	---	+0.15
Yaw Attitude	(-26.9 deg desired)	-26.1 deg (1)
Pitch Gyro Rate (2)	±0.1 deg/sec	-0.06 deg/sec
Roll Gyro Rate	±0.1 deg/sec	+0.03 deg/sec
Yaw Gyro Rate	±0.1 deg/sec	+0.02 deg/sec
Maximum Rates Following Separation		
Pitch Gyro Rate		-0.07 deg/sec
Roll Gyro Rate		+0.22 deg/sec
Yaw Gyro Rate		+0.09 deg/sec

(1) By rate integration of yaw gyro rate.

(2) Geocentric program rate is connected.

3.2.2.2 S3/SV Separation

S3/SV separation events of Rev 15 were as follows:

Event	Vehicle Time (sec)
Start Negative Pitch Maneuver	78612.8
Stop Pitch	78638.8
Separation	80032.4
Start Positive Pitch Maneuver	80033.6
Stop Pitch	80059.6

The ACS parameters just prior to the instant of separation (80032.4 seconds vehicle time) are presented in Table 3-3.

TABLE 3-3

RATE AND ATTITUDE PARAMETERS AT S3/SV SEPARATION

Parameter	Specified	Actual
Roll Horizon Sensor	±1.0 deg	-0.02 deg
Roll Integrator	---	+0.04 deg
Yaw Integrator	---	-0.12 deg
Pitch Integrator	---	-0.12 deg
Pitch Attitude	(-18.3 deg desired)	-19.0 deg (1)
Pitch Gyro Rate (2)	±0.1 deg/sec	-0.001 deg/sec
Roll Gyro Rate	±0.1 deg/sec	+0.001 deg/sec
Yaw Gyro Rate	±0.1 deg/sec	-0.001 deg/sec
Maximum Rates Following Separation		
Pitch Gyro Rate (3)		---
Roll Gyro Rate		-0.49 deg/sec
Yaw Gyro Rate		-0.23 deg/sec

(1) By rate integration of pitch gyro rate.
(2) Geocentric program rate is connected.
(3) Pitch maneuver immediately follows separation.

3.2.3 Payload Operations

a.

b. An out-of-spec roll rate was observed for the Camera 1, Side A operation (OP 408) on Rev 1026. The estimated RCS force levels were as follows:

Thrust Valve 1	3.3 lb	Thrust Valve 5	2.6 lb
Thrust Valve 2	2.7 lb	Thrust Valve 6	3.5 lb
Thrust Valve 3	3.8 lb	Thrust Valve 7	1.8 lb
Thrust Valve 4	3.8 lb	Thrust Valve 8	2.7 lb

The positive roll rotation of the A side optical bar creates a negative response in vehicle rate. This requires Thrust Valves 3 and 7 to fire to counter that negative roll rate. Since Thrust Valve 3 force level was 2.0 pounds greater than Thrust Valve 7, a negative pitch disturbance was created. Thrust Valve 3 will then stop firing until the vehicle attitude and rate return within the pitch switching lines. This, in effect, slows the response to the initial roll disturbance. Roll disturbances in general will excite the solar array, and the 10-second period of the solar array is quite evident during much of the operation. More normal RCS thruster force levels (about 2.5 pounds) would allow timely control of the large roll disturbances created by single optical bar operation. Single optical bar mono operations can be controlled within rate specs at RCS thruster force levels below 2.5 pounds; however, a more reasonable force balance must exist between coupled thrusters. The 0.021 degree/second roll rate spec is to be maintained from 6.6 seconds after steady operation (camera) until closing of the shutter on the last frame. Steady state operation is assumed to have occurred one second prior to the first frame reference time of 2111377.609 (433656.0 seconds vehicle time). Rate spec must, therefore, be met subsequent to 433662.6 seconds vehicle time. The maximum roll rate during this time was -0.029 degrees/second.

c. The HSA inhibit that occurred on Rev 1113 caused loss of the roll horizon sensor signal until transfer to RACS on Rev 1118. The inhibit occurred at 58429 seconds vehicle time. The two operations (OP 454 and 455) taking place during this time are shown below with estimates of the maximum vehicle attitude offset.

Rev	FT+ to FT- (vehicle time)	MAXIMUM ATTITUDE (DEGREES)		
		Pitch (1) Horizon Sensor	Roll (1) Horizon Sensor	Yaw Integrator
1113.4	58525 - 58626	+0.40	-0.20	+0.13
1114.4	63910 - 63992	+0.24	-0.16	> -0.5

(1) From RACS.

The yaw offset on OP 455 of Rev 1114 was the only attitude parameter that may have been beyond the 0.64 degree yaw attitude pointing requirement.

A study investigating the linear stability of the SV with the loss of either the pitch and/or roll horizon sensor signal was done subsequent to an HSA inhibit problem on SV-5. The results of that study shows:

a. If the roll horizon sensor signal is lost, the roll and yaw attitude will diverge oscillatory with a period of $\frac{2\pi}{\omega_o}$. In a typical simulation the amplitude reached six degrees after two orbits.

b. If the pitch horizon sensor signal is lost, linear stability is maintained if the center of pressure is behind the center of gravity.

A plot of the RACS roll horizon sensor output for the time of horizon sensor inhibit (Revs 1113 to 1118) is shown in Figure 3-1. In 2.2 revs the roll offset was in excess of 5 degrees (Rev 1115) which corresponds with the simulation results. The yaw performance is assumed to be similar to the roll, separated by several hundred seconds due to gyrocompassing time constants. Linear stability was maintained in pitch during the 5 rev span, as the maximum offset observed was 0.36 degrees.

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NOTE: Solid lines are flight data. The dashed lines are data interpolations to better show the probable performance during the HSA inhibit.

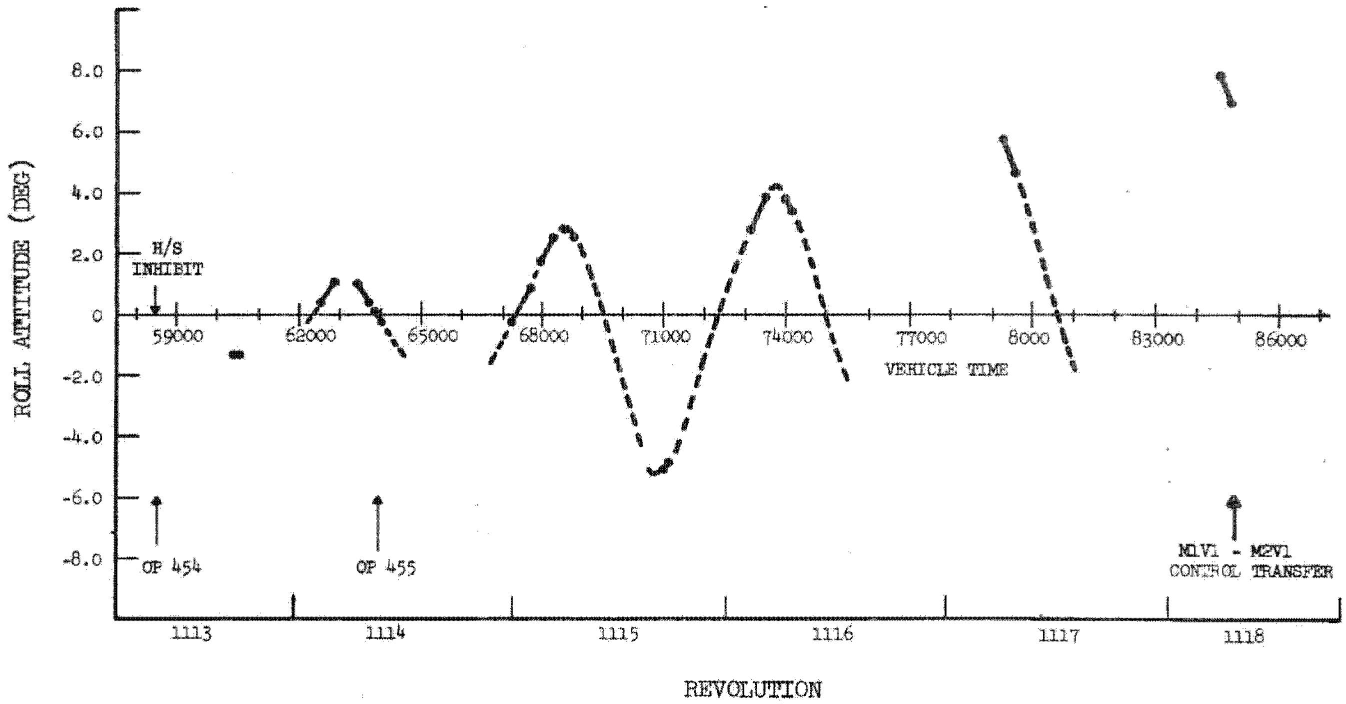

Figure 3-1

Roll History Following HSA Inhibit

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d. A stellar photography experiment was supported on Rev 1769. The 11 frame sequence (each side) was done while the SV was performing a positive pitch maneuver. The center of format was intended to occur at 65.7 degrees south latitude asending which was 200463.5 seconds vehicle time. The pitch gyro rate was integrated from the beginning of the pitch maneuver to 200463.5 seconds and the estimated pitch attitude was 163.9 degrees. The desired angle at the mid point reference time (200463.5) was 170 degrees. This would result in pointing the A side camera 180 degrees from nadir at the time of interest.

3.2.4 Mapping Camera Module (MCM) Operations

3.2.4.1 MCM Calibration Maneuvers

The calibration maneuvers on Rev 941 consisted of a negative pitch maneuver of 167.9 degrees, followed by four inertial periods, one for each calibration. The durations of the inertial periods were 177, 177, 177, and 233 seconds for calibrations 1, 2, 3 and 4 respectively.

A positive pitch maneuver was performed after the calibrations to return to nose forward horizontal flight. Upon return to geocentric control (horizon sensor connect) the pitch offset was -2.4 degrees indicating successful execution of the calibration sequence. Table 3-4 presents the four calibrations.

3.2.4.2 MCM Recovery

The ST-RV (RV-5) recovery is performed with the SV yawed 180 degrees and pitched down, with the release taking place along the SV X-axis. The vehicle rate and attitude parameters at RV-5 separation (vehicle time 71393.8 on Rev 958) are listed in Table 3-5.

3.2.5 Recovery

The pitch down maneuvers preceding RV-1 through RV-5 separations were all within specification and are summarized in Table 3-6. The RV separation performance summary is shown in Table 3-7.

TABLE 3-4

MAXIMUM VEHICLE RATES DURING MCM CALIBRATION

Calibration	Duration ST+ to ST- (sec)	Vehicle Time at Frame 1 (sec)	Settling Time (sec)	MAXIMUM VEHICLE RATES DURING CALIBRATION (DEG/SEC)		
				Pitch	Roll	Yaw
1	166	821797.385	352.18(1) 23.78(2)	-0.005	-0.019	-0.002
2	166	822339.201	21.6	0.007	-0.002	0.001
3	166	822883.201	21.6	0.007	0.002	0.001
4	252 (3)	823427.200	21.6	0.007	-0.005	0.001
Specified	Not to exceed 300	---	600 Allowed	±0.014	±0.021	±0.014

(1) Time from start of pitch down maneuver to Frame 1.
(2) Time from removal of geocentric rate to Frame 1.
(3) Time from ST+ to start of return pitch maneuver.

TABLE 3-5

RATE AND ATTITUDE PARAMETERS AT RV-5 SEPARATION

Axes	ATTITUDE (DEGREES)			Rate (deg/sec)
	Desired	Actual	Source	
Pitch	-64.6 ±3.0	-63.4	PDWN	+0.069 (1)
Roll	±1.0	-0.04	H/S	0
Yaw	±1.0	+0.12	Integrator	-0.001

(1) Includes orbital rate.

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TABLE 3-6

PITCH DOWN PERFORMANCE PRECEDING RECOVERY VEHICLE SEPARATION

RV/Rev	PITCH DOWN ANGLE		MANEUVERING TIME TO < 0.1 DEG/SEC		PITCH DOWN COAST RATE		
	Desired ±3.0 deg	Actual (PDWN)	Spec (sec)	Actual (sec)	Command Rate (deg/sec)	Coast Rate Expected (deg/sec)	Coast Rate Actual - PGR (deg/sec)
1/310	-34.0	-33.1	150	82.0	-0.705	-0.75 ±0.05	-0.71
2/894	-38.1	-37.2	150	92.0	-0.705	-0.75 ±0.05	-0.71
3/1364	-38.5	-37.7	150	85.0	-0.705	-0.75 ±0.05	-0.70
4/2094	-40.6	-39.6	150	89.0	-0.705	-0.75 ±0.05	-0.70

TABLE 3-7

SUMMARY OF RV/SV SEPARATION PERFORMANCE

RV/Rev	Peak Pitch Rate (deg/sec)	Max. Pitch Integrator Angle (degrees)	Induced Impulse By RV (lb-sec)	Pitch Down Prior to Sep (deg)	Pitch Up Following RV Sep to Removal of Manvr Cmd (deg)	Pitch Inertia After Sep (slug-ft^2)	RV Moment Arm (ft)	ROLL ANGLE	
								Spec (deg)	Meas H/S (deg)
1/310	1.33	6.4	125.0	-33.1	99.1	145950	27.1	±1.0	-0.04
2/894	1.39	8.4	136.0	-37.2	97.5	122226	21.8	±1.0	-0.04
3/1364	1.41	6.3	131.8	-37.7	97.8	93718	17.5	±1.0	-0.06
4/2094	1.24	7.5	153.1	-39.6	40.3 (1)	82798	11.7	±1.0	-0.06

(1) Pitch up to horizontal flight.

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3.2.6 HSA 1009 Anomaly

The HSA inhibit described in Paragraph 3.2.3 was a single head inhibit (ie, roll) since the pitch HSA output responded to attitude change whereas roll remained clamped at about 50% of telemetry bandwidth level. Control was transferred to RACS on Rev 1118 with PACS enabled to permit observation of its operation. The sensor remained in inhibit until Rev 1129 when it cycled in and out of inhibit rapidly and then remained out of inhibit until Rev 1146. Inhibit then reoccurred. From Rev 1148 to 1472, the HSA did not exhibit the inhibit anomaly but occasional negative going pitch and roll transients as shown in Figure 3-2 were observed. After Rev 1472, no further anomalies were seen in the PACS HSA performance.

Failure mode analysis and simulation tests indicate the most probable causes of failure are:

a. A defective comparator module in the right radiance channel (A6 or A4)

b. A cracked solder joint at comparator module to printed circuit board (daughter board) interface.

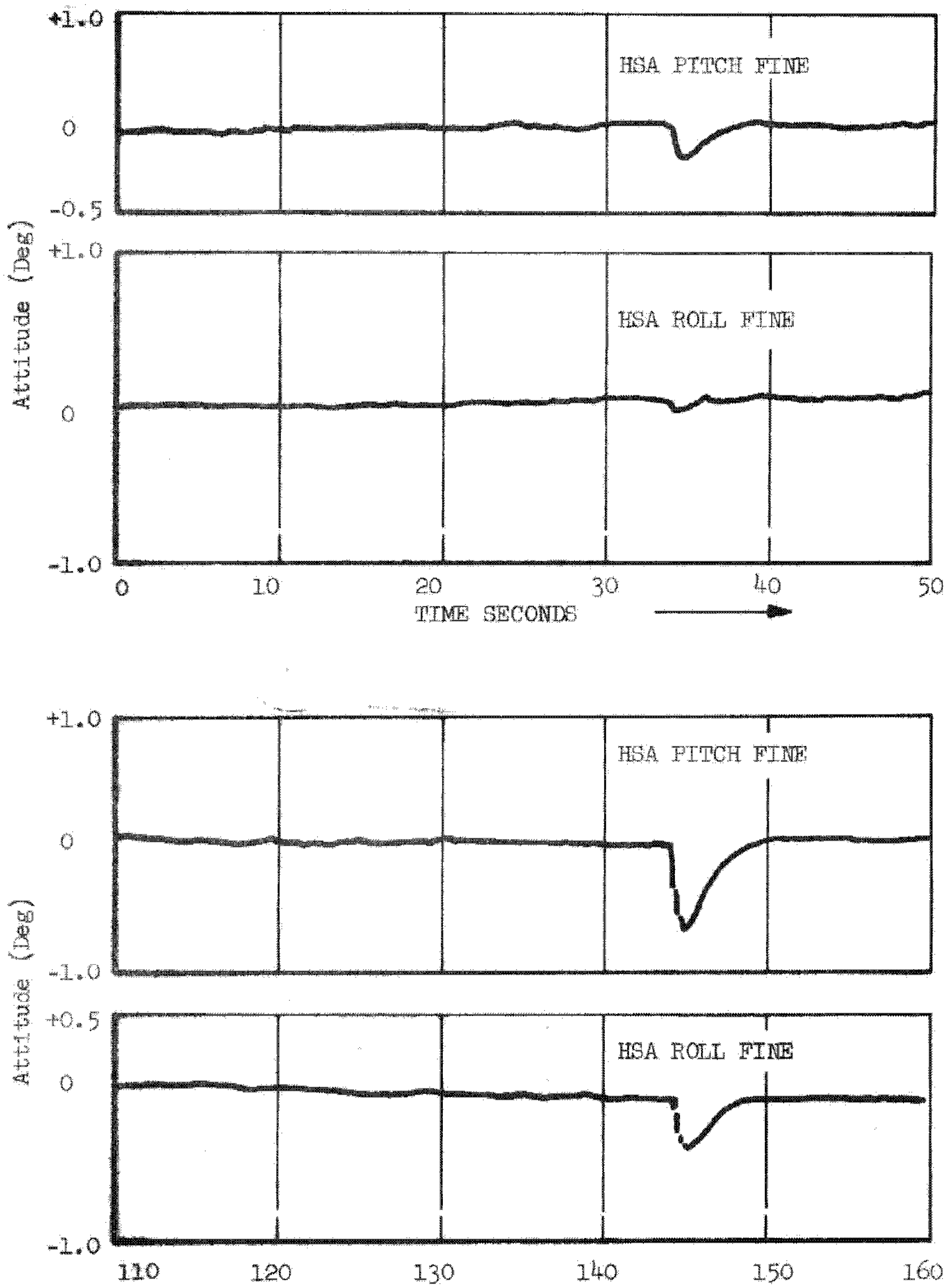
c. A cracked solder joint at daughter board to mother board interface.

HSA 1009 is a Block I unit. Design modifications were incorporated in Block II addressing the failure causes indicated above. No Block II flight anomalies have been observed to date.

3.2.7 IRA 1025 Anomaly

During the solo phase of flight an IRA Ferrotic gyro start-up capability test was conducted.

On Rev 2195 the PACS system was deactivated and allowed to cool for ten revs. On Rev 2205 the PACS IRA was activated. The SMCD monitor did



NOTE: 1. Both examples shown were selected from Rev 1240 data.
2. Similar transients occurred from Rev 1148 to Rev 1472.

Figure 3-2

Typical Transients During HSA 1009 Anomaly not respond, indicating an out-of-limits spin motor current. Repeated attempts to enable the IRA and obtain the correct SMCD response were unsuccessful. Subsequent tests showed that the yaw gyro, Serial Number X30, did not reach synchronous speed, and probably did not start, since no rate response to vehicle yaw maneuvers could be observed.

The anomaly is currently under investigation. Suggested probable causes are:

a. Moisture lockup due to condensation as the gyro cools down of the high relative humidity in the gimbal gas.
b. Contamination in the spin motor aggravated by cool down.
c. An open or shorted motor lead.

3.2.8 FCEA 1023 Anomaly

RACS controlled the vehicle without anomaly or malfunction from Rev 1118 until Rev 2266 when the digital rate, analog rate and the attitude monitors all dropped to a zero telemetry voltage level. These monitors remained at the zero level although the horizon sensor indicated that vehicle rates were present and that the vehicle was tumbling. The zero level monitors are all processed in the FCEA and powered by regulated voltages derived in the FCEA. The discrete monitor for "gyro-compassing Off" also read zero instead of the nominal 5 volts it ordinarily reads.

This FCEA (Serial Number 1023) failure is currently under study.

3.2.9 Horizon Sensor Moon Intercept

A horizon sensor moon interception was observed on Revs 1546B, 1549K, 1550K real time and 1552 playback. A horizon sensor output error occurs when the moon is sufficiently close to the earth limb. The observed moon intrusions did not occur during any payload operations. Maximum vehicle pitch excursions were about 0.4 degrees which is within the 0.7 degree requirement. A pitch horizon sensor output error of 0.8 degrees maximum was possible for the +12.6 beta angle of SV-9, but was not observed.

3.3 REACTION CONTROL SYSTEM (RCS)

3.3.1 Flight Summary

SV-9 was loaded with a total of 523 pounds of RCS hydrazine, 252 pounds in Tanks 1 and 2 and 271 pounds in Tanks 3 and 4. RCS 1 (primary) thrusters were operated from Tanks 1 and 2 with an initial pressure of 233 psia blowing down to 98 psia at near depletion on Rev 989 when all four tanks were manifolded together. The 98 psi in Tanks 1 and 2 was manifolded with the 273 psi in Tanks 3 and 4 to give a four tank system pressure of 142 psi. The RCS 1 thrusters provided control until Rev 1311 when the expected degradation of thrust led to transfer to the RCS 2 (standby) thrusters. RCS 2 provided control from Rev 1311 to Rev 2266 when the FCEA failure described in Paragraph 3.2.8 occurred. The pressure of the four tank system had dropped to 97 psia with 46 pounds of RCS fuel remaining. Deboost on Rev 2274 was under control of Lifeboat.

Two anomalies were noted:

1. Two RCS 1 thruster temperatures were erratic throughout the flight (see Paragraph 3.3.4).
2. A temporary thruster valve leakage occurred on Rev 1267 (see Paragraph 3.3.5).

3.3.2 Propellant Consumption

RCS propellant consumption averaged 3.5 pounds/day through Day 90, which was close to the preflight estimate of 3.7 pounds/day based on prior flight experience. From Day 90 to the end of the mission, the RCS averaged 2.4 pounds/day due to reduced payload activity and higher altitude.

3.3.3 Thruster Performance

Three RCS 1 thrusters (REAs 2, 5 and 8) operating at low duty cycle and one (REA 7) operating at a high duty cycle indicated performance degradation prior to transfer to RCS 2 on Rev 1311. Figure 3-3 is a history of the normalized thrust. The measured thrust is normalized to a constant

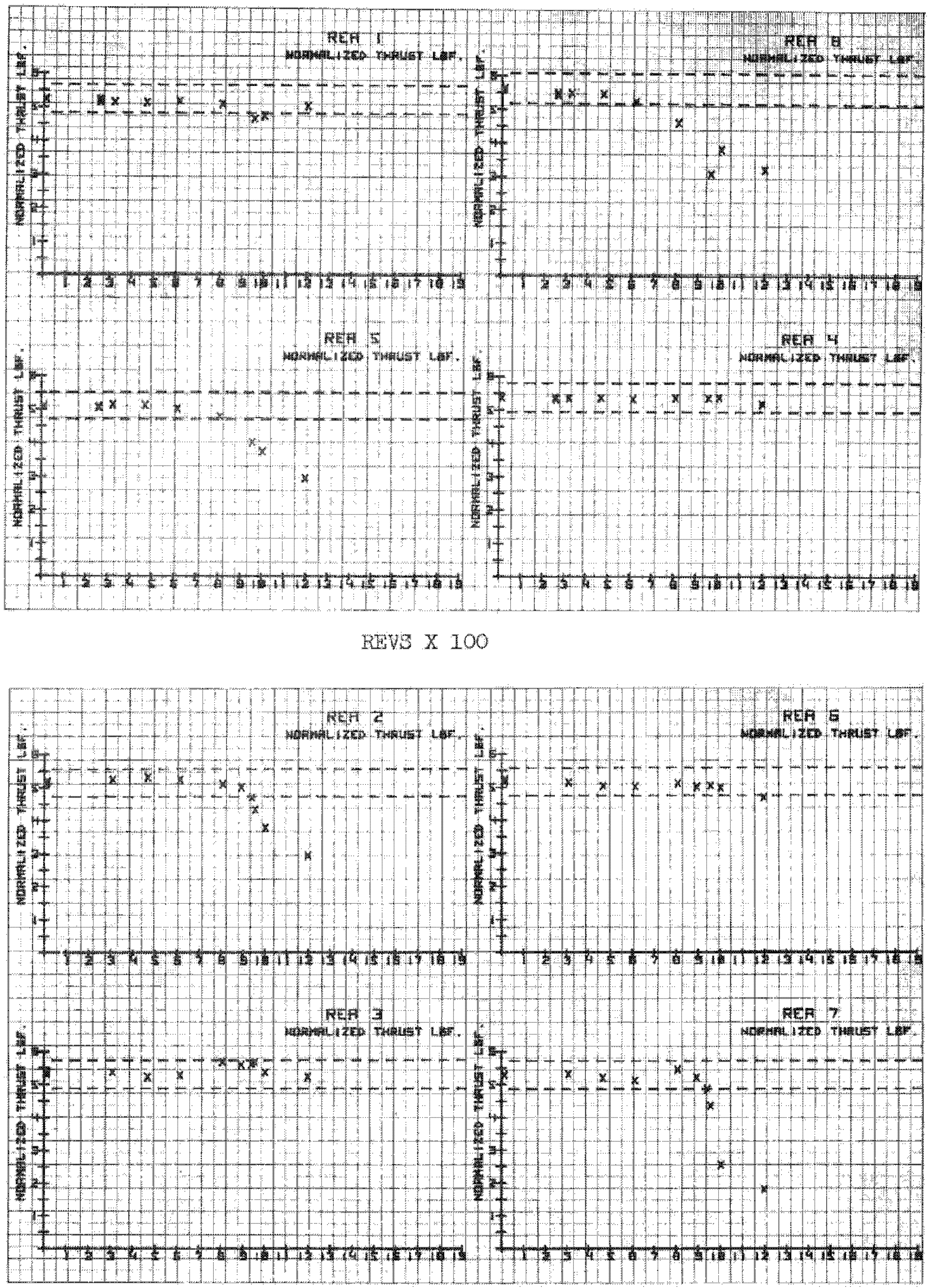


Figure 3-3

Normalized Thrust History

feed pressure and data is selected at a consistant thruster temperature. The dashed lines represent a ±8% tolerance band around the initial baseline thrust calculation.

Table 3-8 shows the thrust values for RCS-2 calculated from gyro rates together with the thrust attained during the qualification tests at the fuel supply pressures available on the pertinent revs. Comparison has been made for a pair of thrusters since the calculated differences between the pair of thrusters are thought to be overly sensitive to errors in the roll characteristics. No degradation in thrust was evident for RCS 2.

Thruster pulse counts are presented in Table 3-9 for both RCS 1 and RCS 2. Accuracy of total pulse counts on the more active thrusters is estimated to be ±30% while the less active thruster pulse counts can only be used as a trend indication. Pulse count sample frequency and rate do not permit a more accurate compilation.

3.3.4 Anomalous Thruster Temperature Readings

The nozzle temperature monitors for REA 5 (B055) and for REA 8 (B058) of RCS 1 displayed anomalous behavior whenever thruster temperatures approached the 600° to 1000°F range. B055 exhibited this behavior during BV/SV separation and later, during the Rev 13 Subsatellite activities, both B055 and B058 gave erratic readings inconsistent with thermal time constants expected with the thruster firings. The anomalous data was characteristic of variable resistance resulting from a break in the thermocouple wire. A previous history of some thermocouple lot failures and the apparent triggering of the phenomena at higher temperatures suggests that the most probable location of the break was the thermocouple lead/junction/adapter area where thermocouple expansions due to changing temperature levels would produce such readings.

Since the B055 and B058 data was unreliable, the steady state thrust analysis utilized the pressure read 3.5 seconds after valve opening rather than the pressure read at 900°F to establish thrust levels. Either reading provides a satisfactory basis for detecting the deterioration of thrust of the REAs. Redesign of the thermocouple and improved inspection techniques have been instituted for the Block III REMs.

TABLE 3-8

RCS 2 THRUSTER PERFORMANCE

Rev	Feed Pressure (psia)	THRUSTER - LBF				Qual Thrust Level Per Pair (lbf)
		1 8	2 3	4 5	6 7	
1330	127	---	2.9 (6.7) 3.8	---	3.9 (7.0) 3.1	(6.8)
1443	122	2.7 (5.7) 3.0	---	3.3 (6.1) 2.8	---	(6.5)
1735	114	2.8 (5.8) 3.0	---	2.6 (5.6) 3.1	---	(6.2)
1769	114	---	2.7 (6.4) 3.7	---	3.4 (5.7) 2.3	(6.2)
2094	106	---	2.1 (5.7) 3.6	---	3.6 (5.5) 1.9	(5.8)
2222	99	3.0 (5.5) 2.5	---	2.5 (5.1) 2.6	---	(5.5)

() Bracketed values are for two thrusters.

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TABLE 3-9

THRUSTER PULSE COUNTS

Thruster Number	RCS-1 THRUSTERS (REV 0 TO 1311)		RCS-2 THRUSTERS (REV 1311 TO 2094)	
	Total Pulses	Pulses/Day	Total Pulses	Pulses/Day
1	24,000	290	21,000	430
2	13,000	160	7,000	140
3	128,000	1570	77,000	1590
4	5,000	60	Negligible	Negligible
5	7,000	90	Negligible	Negligible
6	9,000	110	2,000	50
7	156,000	1920	63,000	1520
8	5,000	60	4,000	100

3.3.5 Temporary Thruster Valve Leakage

Figure 3-4 shows the baseplate temperature anomaly on Rev 1267 indicating valve leakage. Investigation showed that immediately after a three pulse firing on REA 7, the vehicle moved to and remained at the positive pitch dead band attitude limit. At this time REA 2 started firing at approximately 0.6 cps causing its temperature to increase. The duty cycle gradually decayed to normal during the next 5500 seconds. After the three pulse firing, REA 7 indicated a constant chamber pressure of 3 psi without a recorded valve actuation. The above indicated that the REA 7 thrust chamber valve started to leak and that the leak gradually sealed off during the following 5500 seconds.

Two failure modes were possible:

1. Contamination on the thruster valve soft seat.
2. Seat shift causing the poppet to ride on the seat retainer ring instead of the teflon seat.

In ground tests, catalyst particles have been found in the valve and the amount appeared a function of the life/duty cycles; however, no liquid leakage has been detected. Also a seat shift in the valve has been experienced after exposure to extended time at temperature but again no liquid leakage was detected. Both failure modes could be expected to exhibit the self-healing characteristic observed.

Although corrective action was not necessary in this case, valve opening via maneuver commands may have washed particles off the seat, and/or heated the teflon seat to cure the valve leakage. The Block III valve has a different design uitlizing a teflon poppet and a hard seat which precludes a seat shift. Also this design features a longer stroke which makes it less sensitive to particulate contamination.

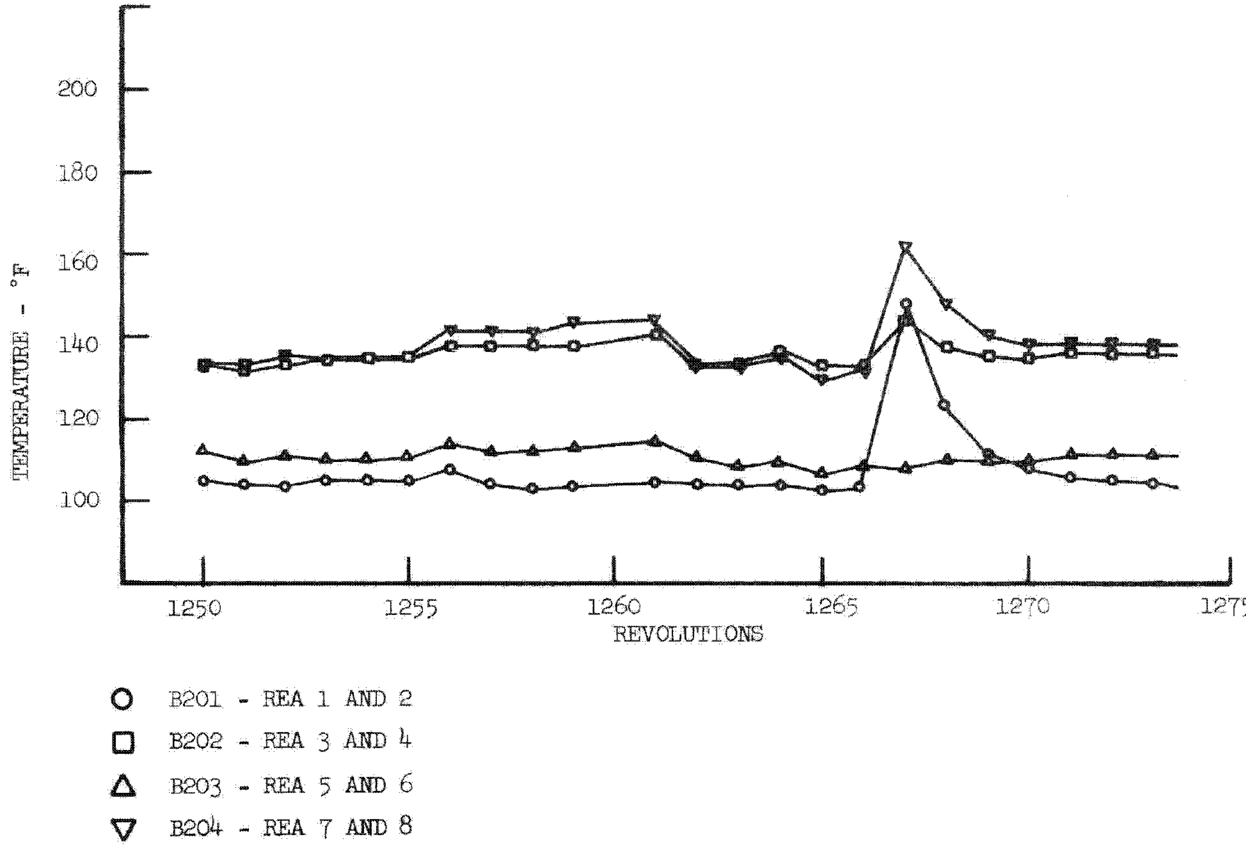


Figure 3-4

REM Baseplate Temperatures During REA 7 (Primary) Valve Leak

3.4 ORBIT ADJUST SYSTEM (OAS)

3.4.1 Orbit Control

The Orbit Adjust System was fired a total of 55 times during the mission. The orbit adjust firings were all normal and engine performance was well within specifications. The catalyst bed resistance factor history was similar to that of other flight engines, exhibiting an initial increase followed by a period of decline. As can be seen in Table 3-10, OAs occurred every three days with an adjustment of perigee location every nine days.

3.4.2 Deboost

The deboost was successfully accomplished with a 120 second firing, followed in five seconds by a firing which went to propellant depletion approximately 343 seconds later. The depletion burn time indicated the remaining fuel had been estimated within ten pounds. Engine performance was normal throughout the deboost and depletion operations.

3.4.3 Propellant Usage

During the active phase of the mission (130 days) the OA propellant usage averaged 22.8 pounds per day for 51 engine starts. For the total mission 3301 pounds were expended for a total impulse of approximately 780,000 pounds-second; these figures are the highest to date for the OA System.

TABLE 3-10

ORBIT ADJUST SYSTEM PERFORMANCE

OA Firing Number	Revolution Number	Impulse Delivered (lb-sec)	Planned Δ V (ft/sec)	Achieved Δ V (ft/sec)	Δ V Error (percent)
1	63	9567	13.9	14.01	+0.8
2	112	16341	23.89	24.00	+0.5
3	159	13132	19.26	19.35	+0.5
4	224	24289	35.93	35.94	0
5	257	16981	24.89	25.23	+1.4
6	259	14309	-21.45	-21.33	-0.6
7	286	23313	35.07	34.89	-0.5
8	322	7006	-11.26	-11.35	+0.8
9	370	15868	25.60	25.78	+0.7
10	420	13150	21.26	21.44	+0.9
11	467	25806	42.96	42.27	-1.6
12	469	12665	-21.00	-20.83	-0.8
13	517	9379	15.21	15.48	+1.7
14	565	14196	23.23	23.49	+1.1
15	614	22641	37.41	37.64	+0.6
16	616	11533	-19.15	-19.25	+0.5
17	662	9732	16.03	16.29	+1.6
18	711	15774	26.25	26.49	+0.9
19	759	8564	14.37	14.44	+0.5
20	808	27524	46.14	46.56	+0.9
21	810	14423	-24.50	-24.52	+0.1
22	857	13569	22.92	23.15	+1.0
23	906	11009	20.47	20.54	+0.3
24	961	13870	26.48	26.60	+0.5
25	1003	18889	36.18	36.40	+0.6

OA Firing Number	Revolution Number	Impulse Delivered (lb-sec)	Planned Δ V (ft/sec)	Achieved Δ V (ft/sec)	Δ V Error (percent)
26	1005	11251	-21.72	-21.77	+0.2
27	1052	11803	22.70	22.91	+0.9
28	1100	9390	18.05	18.29	+1.3
29	1149	13851	26.92	27.07	+0.6
30	1198	21047	41.30	41.34	+0.1
31	1200	10921	-21.50	-21.54	+0.2
32	1246	9627	18.77	19.05	+1.5
33	1295	14120	27.89	28.04	+0.6
34	1344	11339	22.59	22.61	+0.1
35	1393	8309	18.31	18.38	+0.4
36	1441	23544	51.96	52.36	+0.8
37	1443	10285	-23.16	-22.99	-0.7
38	1490	7266	16.10	16.29	+1.2
39	1538	11730	26.44	26.39	-0.2
40	1587	12053	27.18	27.22	+0.1
41	1636	10528	23.87	23.87	0
42	1684	11760	26.70	26.76	+0.3
43	1733	24348	55.58	55.74	+0.3
44	1735	10547	-24.55	-24.28	-1.1
45	1782	8200	19.05	18.94	-0.6
46	1830	11751	27.08	27.24	+0.6
47	1879	9704	22.63	22.58	-0.2
48	1928	10780	25.02	25.17	+0.6
49	1976	10877	25.51	25.50	0
50	2025	9803	23.10	23.07	-0.2
51	2074	11537	27.39	27.26	-0.5
52	2122	12511	33.37	33.47	+0.3
53	2242	3358	9.00	9.04	+0.4
54 & 55	2274	63700*	-175.00*	Deboost	---

* Approximate values

3.5 LIFEBOAT II SYSTEM

Lifeboat magnetometer prediction checks are shown in Table 3-11. The Q magnetometer has an instrumentation bias of between -3 and -6 milligauss so that a data correction is required. The equivalent attitude error of the Q sensor is less than 0.7 degree. The P magnetometer equivalent attitude error was less than 0.8 degree and the R magnetometer equivalent attitude error was less than 1.3 degrees. The three rate gyros were within 0.08 degree/second of the rates monitored on the ACS gyros.

Due to failure in both ACS systems, vehicle deboost was successfully accomplished under Lifeboat control on Rev 2274.

No anomalies were noted.

TABLE 3-11

LIFEBOAT MAGNETOMETER PREDICTION CHECKS

Rev	Mode *	MILLIGAUSS						DEGREES PER SECOND					
		Q Magnet		R Magnet		P Magnet		Y Axis Gyro		X Axis Gyro		Z Axis Gyro	
		Observed	DG Map	Observed	DG Map	Observed	DG Map	Observed	ACS	Observed	ACS	Observed	ACS
18	N-S, RX	-17	-20	Not in Use		Negative Saturation		-0.08	-0.068				
265°N	S-N, RX	-29	-20	Not in Use		Not in Use							
162.5°W	S-N, DB	-17	-20	Positive Saturation		Not in Use							
	S-N, DB	-17	-19	209	215	Not in Use							
310 63°N 152°W	N-S, RX	-20	-20	Not in Use		· 174	179						
958 52°N 142°W	N-S, DB	41	38	61	56	Not in Use							
894 62°N 157°W	N-S, RX	17	-17	Not in Use		183	187						
1364 58°N 152°W	N-S, RX	-26	-28	Not in Use		169	173	1.35	1.31	0.12	0.04	0.04	0.02
2094	N-S, DB	-29	-23	Positive Saturation		Not in Use							
60°N 154°W	N-S, RX	-23	-23	Not in Use		195	202	0.85	0.89	0.02	0.03	0.06	0.04

* DB indicates deboost, RX indicates recovery.

3.6 ELECTRICAL DISTRIBUTION AND POWER (EDAP)

3.6.1 Solar Arrays

Solar arrays were extended on Rev 1. Power output from each leg exceeded the specification value. Degradation from normal orbit environments was 4.7% after 125 days (Rev 1995) in orbit. After RV-4 on Rev 2094 the solar array received more contamination on the back side from the RV retro as the SV was stopped in a level attitude rather than being positioned with the solar array edge on to the retro. This raised solar array peak temperatures 5°F which increased degradation to 6.9% on Rev 2106.

3.6.2 Main Bus Voltage

The main bus voltage varied from a low of 26.3 to a high of 31.0 volts. The allowable range is 25.5 to 33.0 volts. Low range voltage was obtained during payload operations with a bus load of 74 amperes. High voltage data was gathered during charge cycles.

3.6.3 Power Capability and Usage

Power usage ranged from 319 to 420 amp-hours/day. The 420 amp-hours on Day 1 exceeded the 400 (calculated) amp-hour/day capability. K2's occurred on Rev 5 (Day 1) indicating the generating capability exceeded the 400 amp-hours/day during heavy bus loads. Excess capacity was demonstrated with K2's occurring on essentially every rev after Rev 9 except those with heavy payload operations.

3.6.4 Type 29 Battery Performance

All Type-29 Batteries operated in a desirable environment (44°F to 49°F) and performed normally throughout the mission.

3.6.5 Pyro Battery Performance

Pyro Battery 1 stabilized at 50°F which minimized self discharge during the mission. Lift off capacity was 11.62 amp-hours. After 130 flight days the usage for instrumentation (4.0 ampere-hours) and self discharge

(1.53 amp-hours) totaled 5.53 amp-hours. This left a residual capacity of 6.09 amp-hours. Remaining cell life after 130 days of flight was 3.5 days.

Pyro Battery 2 followed the same pattern with the exception of 11.36 amp-hours capacity at lift off.

3.6.6 Lifeboat Battery Performance

The Lifeboat Battery operated normally in a 48°F environment throughout the entire mission. A total of 183 amp-hours remained at the end of 130 mission days from an initial 370 amp-hours at launch. Remaining cell life was 5.5 days.

3.7 TRACKING, TELEMETRY AND COMMAND (TT & C)

3.7.1 Tracking

Excellent tracking performance was maintained throughout the flight mission. During the last day of solo (Rev 2266 to 2274), the FCEA failure (see Paragraph 3.2.8) caused the vehicle to tumble. Due to the vehicle antenna pointing angles with respect to the ground station, the down link data was not always available at the tracking station. Commands were sent to the vehicle only when the down link was available to verify vehicle command accept.

Because of the low transmitter signal strength reported on SV-8, health tests were performed once a week on SGLS 2. To obtain consistent data, the signal strength was measured and plotted (Figure 3-5) at the 5° elevation angle for acquisition and fade using the 46 foot antenna at HULA. This data exhibits good signal strength for the entire flight; therefore, precautions taken on SV-9 and up for maintaining pressure in the rf cavity of the transmitter are believed to be adequate.

3.7.2 Telemetry

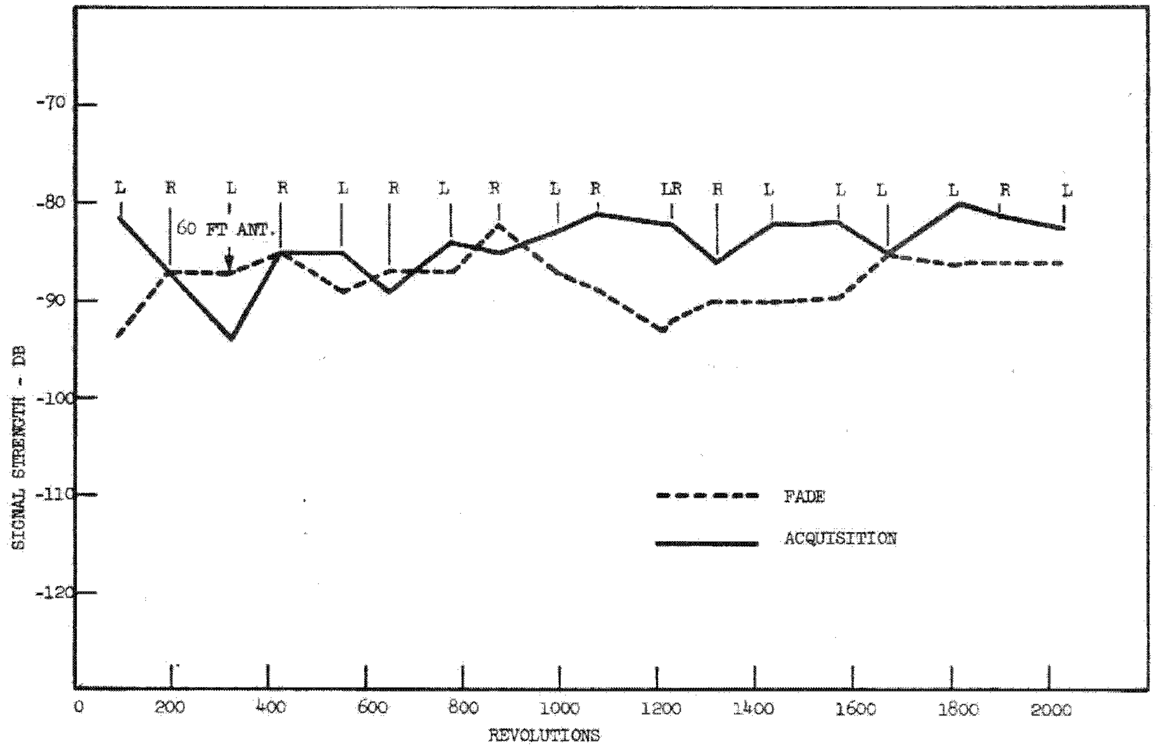
The telemetry performance was excellent for the entire flight. There were no anomalies reported.

3.7.3 Tape Recorder

The tape recorder performance was satisfactory throughout the flight. There was a software problem during the MCS/LB 2 health test on Rev 18. Tape Recorder 1 was in the record mode and Tape Recorder 2 was selected for commanding by the TT & C Off Command. Therefore, when the TT & C Off Command was executed at POGO, Tape Recorder 1 continued to record until HULA. There was no flight data lost and the command sequence has since been corrected.

3.7.4 Instrumentation

There were no instrumentation anomalies at lift off.



NOTES:

- o Values are given at 5° elevation at acquisition and fade.
- o L and R indicate the side of SV facing the station.

Figure 3-5

SGLS 2 Signal Strength History at HULA, 46 Foot Antenna

Telemetry monitor B055 (thruster temperature monitor 5) and B058 (thruster temperature monitor 8) exhibited anomalous operation. This anomaly appeared on both sides of the PCM Remote Unit. Additional investigation indicates that the problem is in the thermocouple (see Paragraph 3.3.4).

3.7.5 T & T Equipment On/Off Cycles and Operating Time

The utilization through Rev 2094 of T & T equipment is summarized below:

Primary Equipment	On/Off Cycles	Operating Time in Hours
SGLS 1	2,426	234.10
T/R 1	9,898	506.18
PCM 1	10,240	656.36
PCM 2	21	2.26
SGLS 2	39	1.17
T/R 2	71	9.43

3.7.6 Command

3.7.6.1 GFE Command System (ECS)

The ECS executed all required SPC in memory and real time commands. The total SPC commands loaded during the primary mission were 283,501 of which 129,830 were executed. The remainder were erased.

The ECS executed two spurious real time commands at GUAM Rev 759. The analysis indicated that these commands were not a result of discrepant vehicle hardware but were caused by a deterioration of the up link signal level.

Real Time Command ER 57 (Left Solar Array Position Drive-Stop) was accepted and executed by Decoder B and 100 milliseconds later Decoder A accepted and executed one of three internal commands which resulted in Decoder A being placed in a non-real-time state. During this portion of the pass, the vehicle was at an elevation of approximately 1.2 degrees and a range of approximately 880 nm. The nominal up link signal strength at the vehicle receiver was -75 dBm, based on a 50 watt transmitter output, 43.7 dBm gain in the 60 foot antenna and -165 dB space loss (combined space loss, cable loss, and antenna polarization loss). Coincident with the anomaly the down link signal strength experienced a drop of approximately 35 dB for a duration of approximately 0.5 seconds. Assuming the up link was similarly attenuated, the vehicle receiver input signal would drop to the vicinity of -110 dBm. A decrease in signal level of this magnitude was up link modulation, "S" pulses, present will cause the receiver to squelch. Under these conditions the receiver will output data and bit read pulses randomly until squelch has been activated. Within a 0.5 second of the first accept of the spurious command, there were 5 different occasions where both decoders stopped outputting reject telemetry for a period of 5 to 20 milliseconds indicating the absence of "S" pulses from the VCTS Receiver. This also indicates the receiver was in the process of squelching.

The following options could be implemented to minimize reoccurrence of this situation:

1. Limit period of up link modulation to greater than 5° elevation.
2. Turn decoders off after the last command transmission on each pass and before reaching 5° elevation at fade.
3. Turn off "S" pulse modulation at known antenna pattern holes.
4. Increase up link rf power levels.

The vehicle hardware, command system and VCTS were not discrepant since the anomaly occurred beyond the design baselines of 5° minimum elevation angle and maximum range of 850 nm. For the remainder of the flight the decoders were turned off 30 to 60 seconds before fade and no other spurious commands were executed.

3.7.6.2 Minimal Command System (MCS)

The MCS executed all commands correctly during the primary mission.

3.7.6.3 Remote Decoder/Backup Decoder

The Remote Decoder was used for each of the five recoveries. The performance of both channels was verified from telemetry to be proper in each case.

The Backup Decoder operational capability was verified during the health check.

3.7.6.4 Command System

Usage summary through Rev 2094

System	Total Operating Time (Hours)
ECS	3095.0
MCS	4.0
Remote Decoder	13.77
Backup Decoder	0.05

3.8 MASS PROPERTIES

A history of SV mass properties throughout the flight are tabulated in Table 3-12.

3.9 PREFLIGHT WINDS ALOFT LOADS ANALYSIS

Table 3-13 presents a chronological tabulation of the pre-launch winds aloft loads analyses for SV-9. The results are plotted in Figure 3-6 The analyses are based on observations made at the Vandenberg Air Force Base using Rawinsonde balloon soundings on the 28th and 29th of October 1974.

The T-24 hour simulation analysis resulted in a "hold-continue count" recommendation due to excessive booster TVC fluid usage.

The T-12 hour analysis again indicated excessive use of TVC fluid and also indicated that the TVC fluid dumper time off constraint was exceeded and the recommendation was "hold-continue count".

The T-8.5 hour analysis showed the TVC fluid usage within limits; however, the TVC fluid dumper time off was excessive resulting again in a recommendation of "hold-continue count".

The succeeding T-6 and T-3 hour analysis indicated that all parameters were within acceptable limits. Consequently, the recommendation was "go for launch".

3.10 SOLAR ARRAY

Deployment and erection of the left (-Y) solar array is shown in Figure 3-7 and for the right (+Y) solar array in Figure 3-8. The arrays were deployed at the first station pass, INDI. Since +18° is the initial position and is the optimum for the range of beta angle (9 to 12.5 degrees) flown during the basic mission, no repositioning of the solar arrays was done until the solo experiments.

TABLE 3-12

SV-9 MASS PROPERTIES

Description	Weight (lb)	Center of Gravity (inches)			Moment of Inertia (slug-ft^2)			Product of Inertia (slug-ft^2)		
		$\bar{X}$	$\bar{Y}$	$\bar{Z}$	I_x	I_y	I_z	I_{xy}	I_{xz}	I_{yz}
At Launch	26163	1970.9	1.06	3.10	7694	213817	213806	-1092	-902	-69
Separated from S2	23162	1981.3	1.17	3.47	5516	182206	182120	-1044	-678	-71
Arrays Deployed +18°	23162	1981.7	1.17	3.76	6690	183328	184380	-1041	-1058	-87
After -Y Subsatellite Ejection	22620	1985.8	2.22	4.15	6411	179808	180674	-152	-721	-169
After +Y Subsatellite Ejection	22010	1990.8	1.24	4.85	6129	175271	176065	-1018	-110	-55
Prior to Drop 1	21432	1977.4	1.27	5.54	6081	178621	179431	-1106	32	-59
After Drop 1	19888	2000.7	1.36	4.63	5883	145950	146893	-976	-1244	-54
Prior to Drop 2	18812	1982.3	1.44	5.70	5808	143385	144358	-1093	-1341	-60
After Drop 2	17273	2003.7	1.57	4.66	5609	122226	123332	-973	-2359	-54
After Drop 5	16806	2012.1	1.63	5.59	5472	108456	109684	-897	-1198	-62
Prior to Drop 5	16102	1997.0	1.70	5.88	5461	107211	108416	-982	-602	-63
After Drop 3	14566	2017.1	1.87	4.67	5263	93718	95054	-870	-1403	-56
Prior to Drop 4	13586	1998.6	2.01	5.81	5220	88961	90299	-983	-1365	-63
After Drop 4	12063	2014.6	2.27	4.36	5023	82798	84266	-892	-1906	-54
Prior to Deboost	11887	2012.1	2.30	4.43	5022	81681	83150	-907	-1935	-55
End Deboost	11605	2008.1	2.36	4.54	5020	80028	81498	-931	-1980	-55

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TABLE 3-13

WINDS ALOFT ANALYSIS SUMMARY

	BALLOON RELEASE TIME					
	T-30	T-12	T-8.5	T-6	T-3	T-0
	3800 RUN AT STC TIME					
	T-24	T-8.5	T-6.0	T-3	T-1	---
SV Structural Loads:						
Bending Mom, % Limit Load	93.23	60.04	62.79	56.63	57.33	61.19
Critical SV Station	1,902.	1,902.	1,902.	1,902.	1,902.	1,902.
Elapsed Time, seconds	26.2	16.85	18.38	16.68	22.66	53.42
Altitude, feet	8,000.	3,398.	3,999.	3,336.	6,000.	33,001.
SRM Side Force:						
% Allowable	67.51	79.66	81.43	63.97	53.57	58.36
SRM Number	2	1	1	1	1	1
Pitch or Yaw	Yaw	Pitch	Pitch	Pitch	Pitch	Pitch
TVC Usage for Control:						
% Allowable Expended	133.63	115.34	96.65	75.85	70.18	81.39
SRM Number	1	1	1	1	2	1
Expended, pounds	2,004.5	2,238.81	1,875.92	1,472.3	1,236.6	1,579.78
Vehicle Response						
Maximum $\alpha\bar{q}$, % allowable	81.58	48.64	27.11	26.79	26.42	32.73
Maximum $\alpha\bar{q}$, deg-psf	2,738.55	1,692.73	930.9	937.6	924.61	1,405.15
Elapsed Time, seconds	39.38	29.52	29.78	29.47	29.54	53.42
Altitude, feet	18,150.3	10,139.2	10,312.	10,113.	10,156.	33,000.

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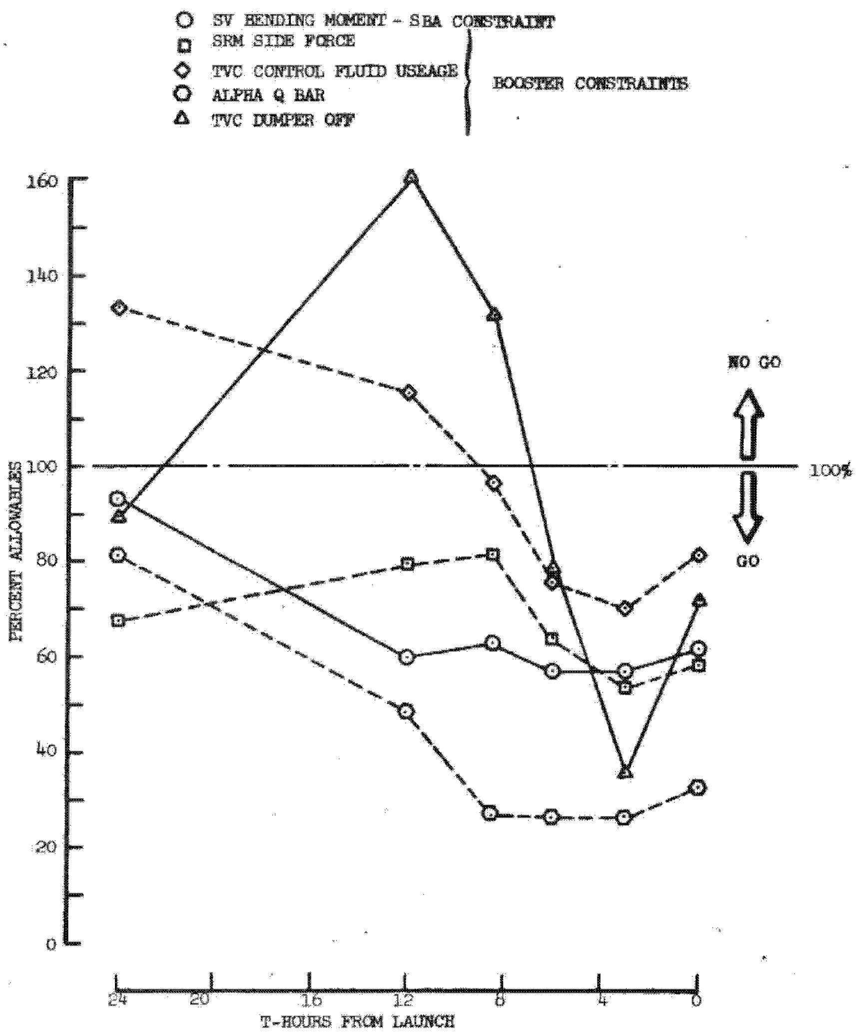

Figure 3-6

Critical Launch Parameter Summary

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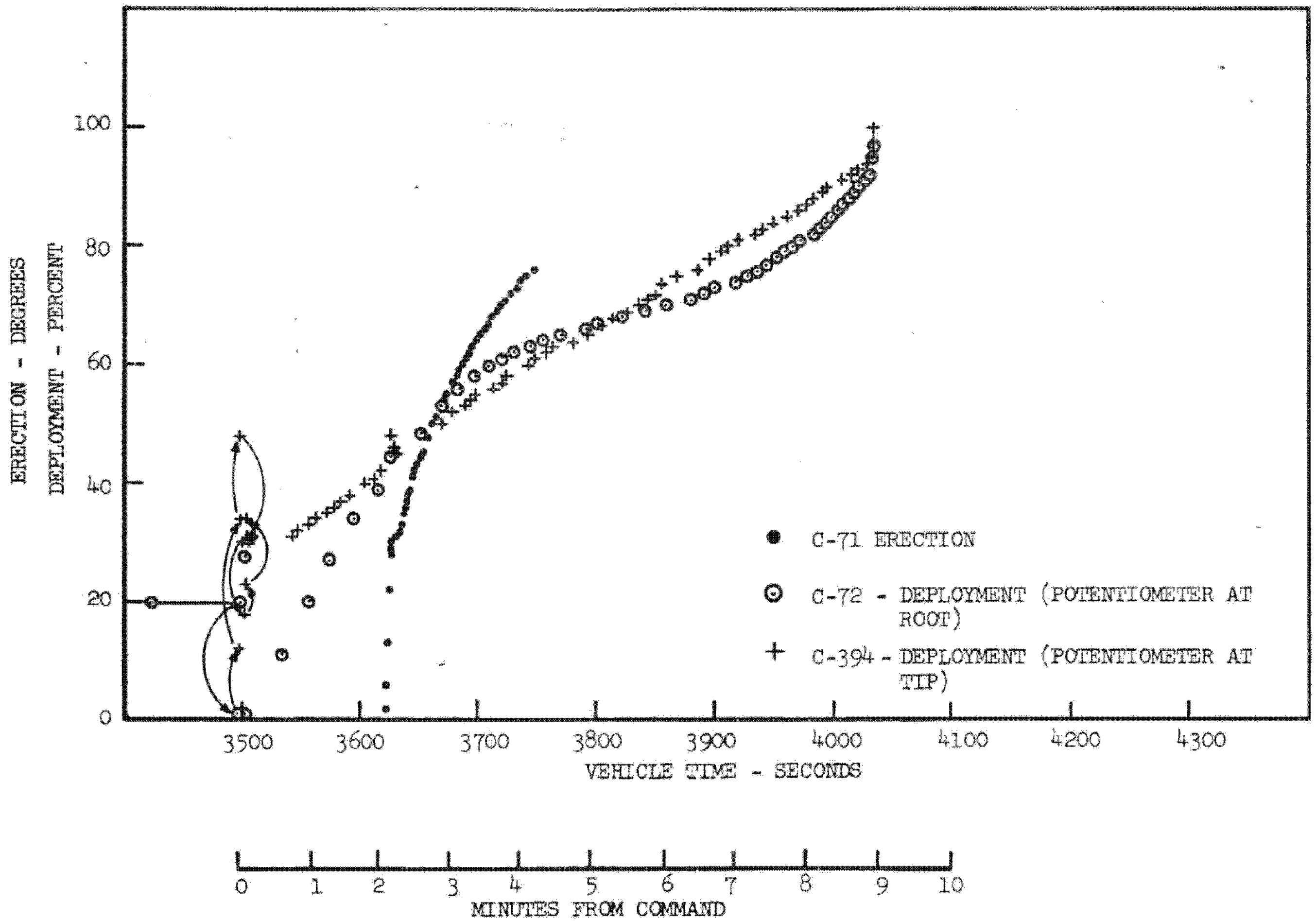

Figure 3-7

Left Solar Array Erection and Deployment Time Histories

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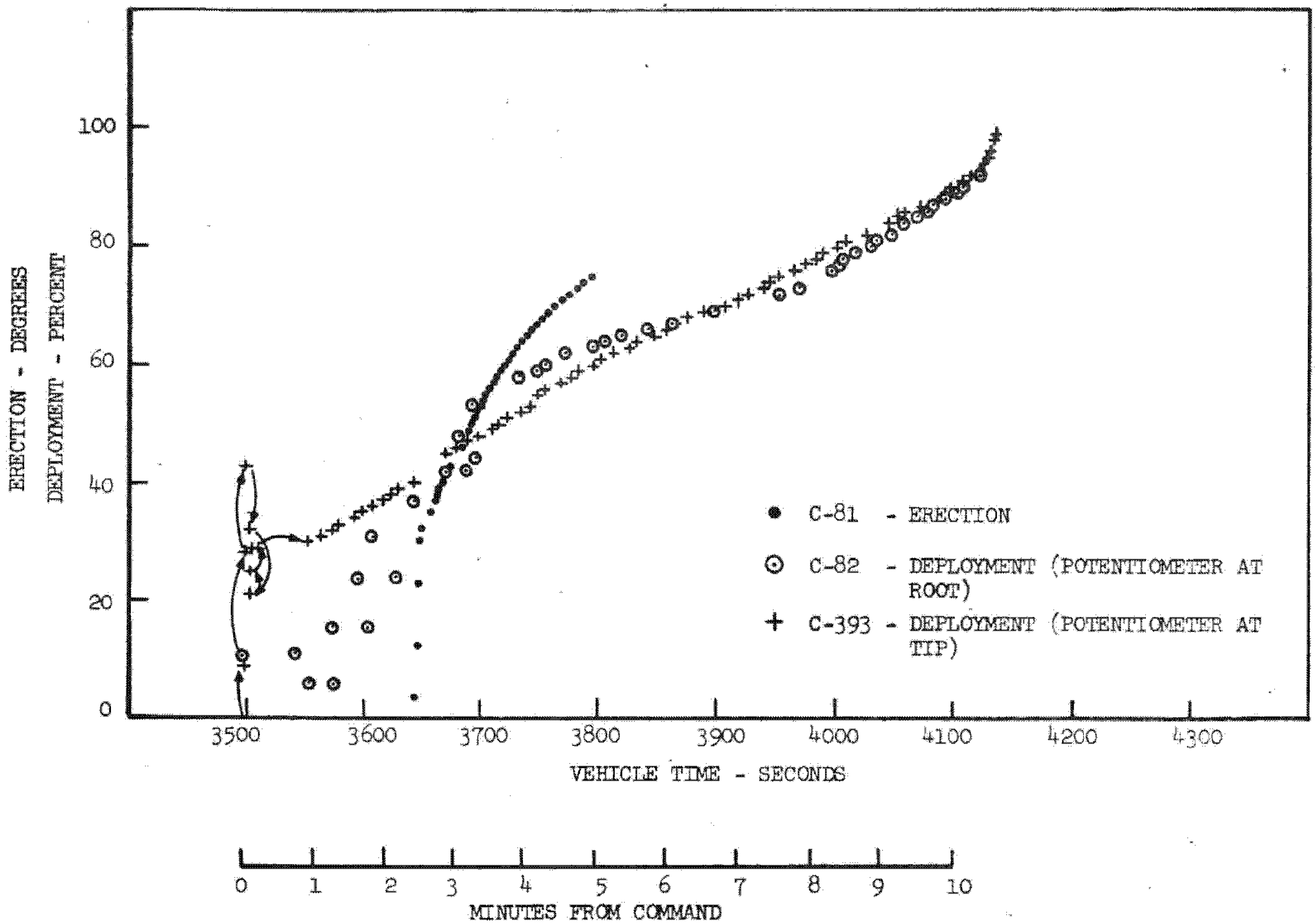

Figure 3-8

Right Solar Array Erection and Deployment Time Histories

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3.11 THERMAL CONTROL

3.11.1 Mid and Forward Sections Including MCM

The flight temperatures of the Mid Section, Forward Section and MCM are summarized in Table 3-14. This data indicates that the average section temperatures were well within the required design limits. No design changes are forthcoming as a result of SV-9 flight experience.

3.11.2 Active Thermal Control (ATC)

The ATC reference temperature in the Mid Section experienced a range of approximately 5°F (Figure 3-9). The change in orbital beta angle that occurred during the flight would, by itself, have produced an obvious change in temperature. The seasonal change in solar constant in conjunction with payload activity could account for the variations experienced during flight. The RV heater control zones, which are actively controlled to the reference temperature, were generally at the desired temperature.

3.11.3 Aft Section

Acceptable Aft Section temperature control was maintained throughout the flight. All equipment temperatures remained within design limits as shown in Table 3-15.

The thermal design of the SV-9 Aft Section was the same as SV-8. The resulting flight temperatures were similar to SV-8 at the beta angle range of 9 to 12.5. There were no thermal anomalies.

Figure 3-10 shows orbit average flight temperatures with pre flight predictions made for two cases: (1) uncontaminated surface properties and (2) contaminated properties representative of the maximum levels of contamination observed in previous flights. Booster contamination degradation appears to be similar to that of other flights.

TABLE 3-14

THERMAL DATA SUMMARY

Vehicle Section	Parameter	Design Limits	SV-9 Actual*
Mid Section	T_{TCA}	49/91	72
	T_{SU}	47/93	74
	$T_{SU} - T_{TCA}$	5/-4	2
Forward Section	T_{FWD}	47/93	72/80
	$T_{FWD} - T_{TCA}$	±20	0/8
MCM	T_{ENC}	32/69	55
	T_{TP}	30/85	51
	DBS Panel	32/90	63

DEFINITIONS

T_{TCA} = Orbit average radiation temperature of the TCA compartment structure.

T_{SU} = Orbit average radiation temperature of the SU compartment structure.

T_{FWD} = Orbit average temperature of each Forward Section bay based on the average temperature of the bulkheads.

T_{ENC} = Orbit average temperature of the MCM enclosure.

T_{TP} = Orbit average satellite recovery vehicle take-up temperature.

Temperatures are in °F.

* Based on orbit average temperatures from Rev 279 (stabilized thermal conditions). During remaining portions of the flight these temperatures generally did not change more than 2°F.

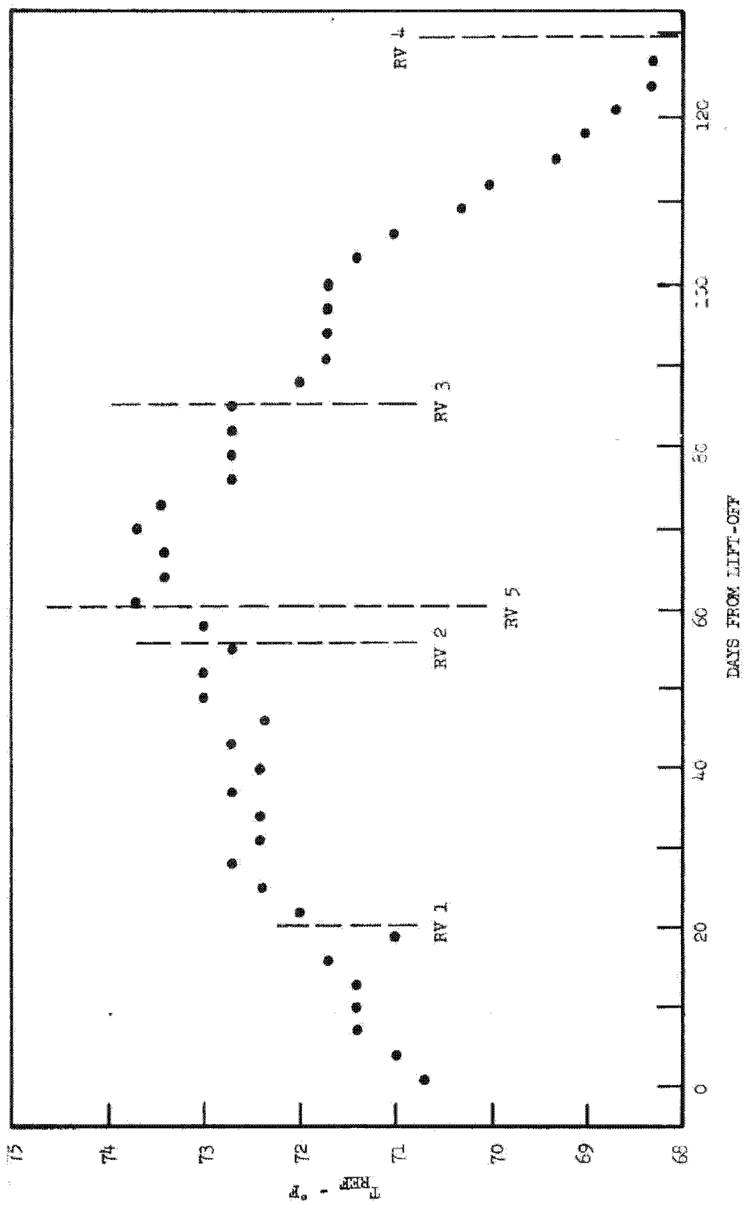

Figure 3-9
SV-9 T_{REF} History

TABLE 3-15

AFT SECTION CRITICAL COMPONENT TEMPERATURES (°F)

Critical Component	Design Limits	SV-9 Predictions (1)	SV-9 Actuals (2)
Power Distribution Junction Box	-30/165	82	76/90
Charge Current Controller 2	-30/170	90	95/106
Type 29 Batteries Bay 3	35/70	48 (3)	43/49
Type 30 Battery	30/90	48 (3)	47/51
Type 31 Batteries	40/90	48 (3)	48/54
Type 29 Batteries, Bay 4	35/70	48 (3)	43/51
Horizon Sensor Assembly Heads	0/130	67/71 (4)	77/87
Inertial Reference Assembly	50/130	97/104 (4)	104/117
Pulse Code Modulator Master	-30/170	80	80/110
Tape Recorders	20/120	77	78/94
Transmitters	-30/170	77/85	80/110
Extended Command System Clocks	40/153	98/104	100/107
Programmable Memory Unit A	-40/145	86	80/92
Programmable Memory Unit B	-40/145	96	87/103
Inertial Reference Assembly Gyros	50/200	---	139/168
Minimal Command System	-40/149	71	70/81
Reaction Control System Tanks	40/140	73/83	70/100
Plumbing Bay 6	35/140	80/82	78/105
Plumbing Bay 12	35/140	84/92	77/101
Orbit Adjust Tank	70/100	83	71/99
Positional Drive Assembly	-30/160	60	55/98
Solar Arrays	-125/225	---	-87/157
Quad Valve	40/200	---	108/193

(1) Predicted temperature is for the orbit average; multiple temperatures indicate multiple points.
(2) Stabilized temperature ranges not including launch transients.
(3) Temperature controlled by heaters.
(4) Prediction assumes dual IRA operation (Primary ACS and Redundant IRA only).

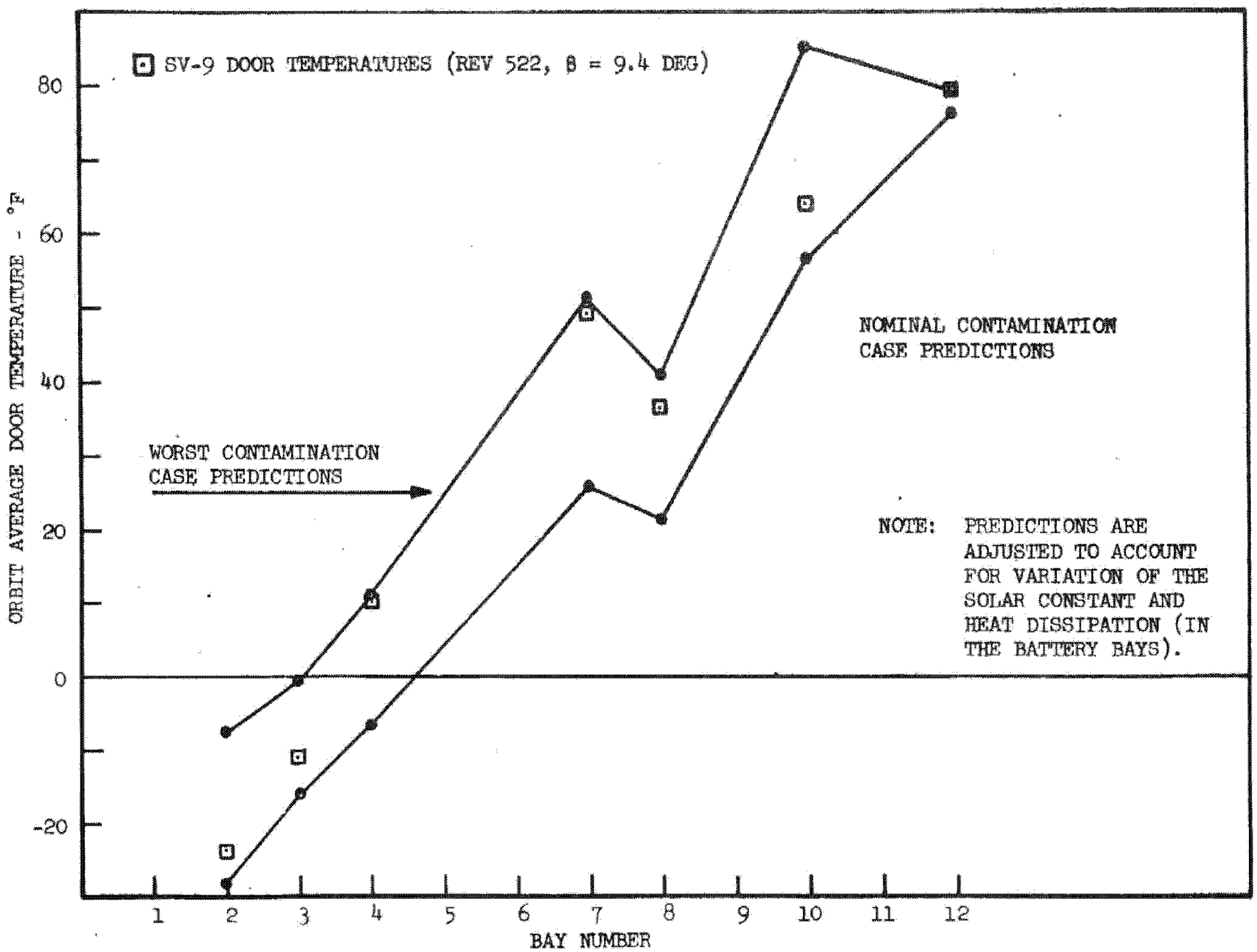

Figure 3-10

SV-9 Equipment Door Temperatures

3.12 SHROUD SEPARATION

Four pressure transducers were installed in the shroud in an attempt to determine if a pressure exists in the shroud at the time of separation. The transducers were calibrated to read 0 to 0.5 psia and did measure a change of that magnitude; however, the sensing element cannot hold the calibration for the length of time between the last possible calibration and launch. The data did not prove either the presence or absence of pressure at separation. Since no suitable instrumentation is known, this line of investigation for the cause of the apparent initial acceleration of the shroud will not be continued at this time.

SECTION IV
PAYLOADS

4.1 SENSOR SUBSYSTEM

4.1.1 Coarse Film Path

Both coarse film paths (supply, loopers, steerers, articulators and take-ups) exhibited nominal operation throughout the mission.

4.1.2 Fine Film Path

Both fine film paths performed nominally throughout the mission.

4.1.3 Command and Control

The command and control subsystem functioned normally throughout the mission. On-orbit Adjust Assembly (OOAA) and exposure adjustments as determined by post flight analysis were made.

4.1.4 Optical Bar Performance

Mechanical and optical performance of both optical bars was nominal throughout the mission with the optical performance of the aft looking camera being superior at all times.

4.1.5 Pneumatics Subsystem

The pneumatics subsystem performed normally throughout the mission.

4.2 TERTIARY PAYLOADS

[redacted] Vehicle support of the system (power, commands, telemetry, tape recorder, thermal environment, etc) was provided satisfactorily in accordance with the system requirements. Five calibrations were successfully accomplished through Rev 1701.

4.3 SUBSATELLITES

Two subsatellites were carried into orbit on SV-9. A 599 pound subsatellite system (541.9 pound separable) was carried on the -Y side of the Forward Section. Separation of this subsatellite occurred at Rev 13.5 following an SV yaw maneuver of -26.9°. Separation occurred at a 15° south latitude on a descending mode. All subsatellite separation events occurred in the normal sequence within specified limits and the subsatellite went on to achieve its desired orbit.

A 620.9 pound SAMSO S3-1 Subsatellite (608.6 pound separable) was carried on the +Y side of the Forward Section. Separation of this subsatellite occurred at Rev 15.3 (56.8° north latitude on a descending mode) following a 18.3° pitch down maneuver of the SV. The S3-1 Subsatellite separation sequence was accomplished within desired tolerances and constraints, and the subsatellite achieved its intended orbit.

4.4 MAPPING CAMERA SUBSYSTEM

The operation and performance of the fifth ST Camera System flown on SV-9 is considered excellent.

SECTION V
REENTRY VEHICLE SUMMARY

5.1 SUMMARY

The recovery statistics are shown in Table 5-1 and Figure 5-1. Performance of the RV subsystems is summarized in Table 5-2. All RV on-orbit and reentry events occurred as planned and the RV flights followed the predicted trajectories. All four recoveries terminated in aerial retrieval at locations near the PIP.

The take-up core pins were sheared only on RV-2. Aerial retrieval loads exceeding the core pin strength are expected.

All subsystems performed satisfactorily and met all mission requirements, see Table 5-2.

5.2 REENTRY VEHICLE PERFORMANCE

All RV on-orbit functions were normal and occurred on time. The SV provided a satisfactory pitch angle for each RV separation. All other SV/RV interface functions were nominal.

The RVs were adequately spin stabilized during the vacuum coast phase and aerodynamically stable during the atmospheric phase of the reentry trajectory. Figure 5-1 shows the entry conditions at the time of drogue deployment which are also within the design envelope. Post flight examination revealed two recovery operational variances which are noted in Paragraph 5.3. RV-4 was not available for inspection in time for this report; however, all available data indicated nominal satisfactory performance.

TABLE 5-1

RV RECOVERY SUMMARY

	RV-1	RV-2	RV-3	RV-4
RV Serial Number	37	38	39	40
Recovery Rev Number	310	894	1364	2094
Recovery Date	17 November 1974	23 December 1974	21 January 1975	7 March 1975
Payload Weight (lb) (Measured Weight from Recovered RV)				
Forward	230.6	230.8	222.6	228.3
Aft	230.9	226.7	233.4	217.0
Unbalance Percent	0.1	1.5	3.9	4.1
SV Orbit (h_p X h_a/ω_p)*	87.9 x 164.6/136.6	87.7 x 153.3/126.6	87.9 x 160.0/120.5	88.2 x 161.7/103.3
SV Pitch Angle (degree)	-33.1	-37.2	-37.7	-39.6
Nominal PIP Latitude (°N)	25.5	23.0	18.0	18.0
Impact Location Error (EPPD Versus Teapot Evaluation)				
Overshoot (nm)	8.0	5.1	19.5	7.5
Undershoot (nm)				
Cross Track (nm)	6.2 E	0.9 W	3.1 E	2.1 E
Recovery (Aerial)				
Altitude (ft)	7,600	13,350	13,900	12,700
Parachute Condition	Normal	Normal	1 cone & 1 main tear	1 cone & 2 main tears
Retrieval Pass	2	1	1	1
RC/Payload Condition	Good	Good	Good	Good

* h_p = Altitude of Perigee (nm), h_a = Altitude of Apogee (nm), ω_p = Argument of Perigee (deg)

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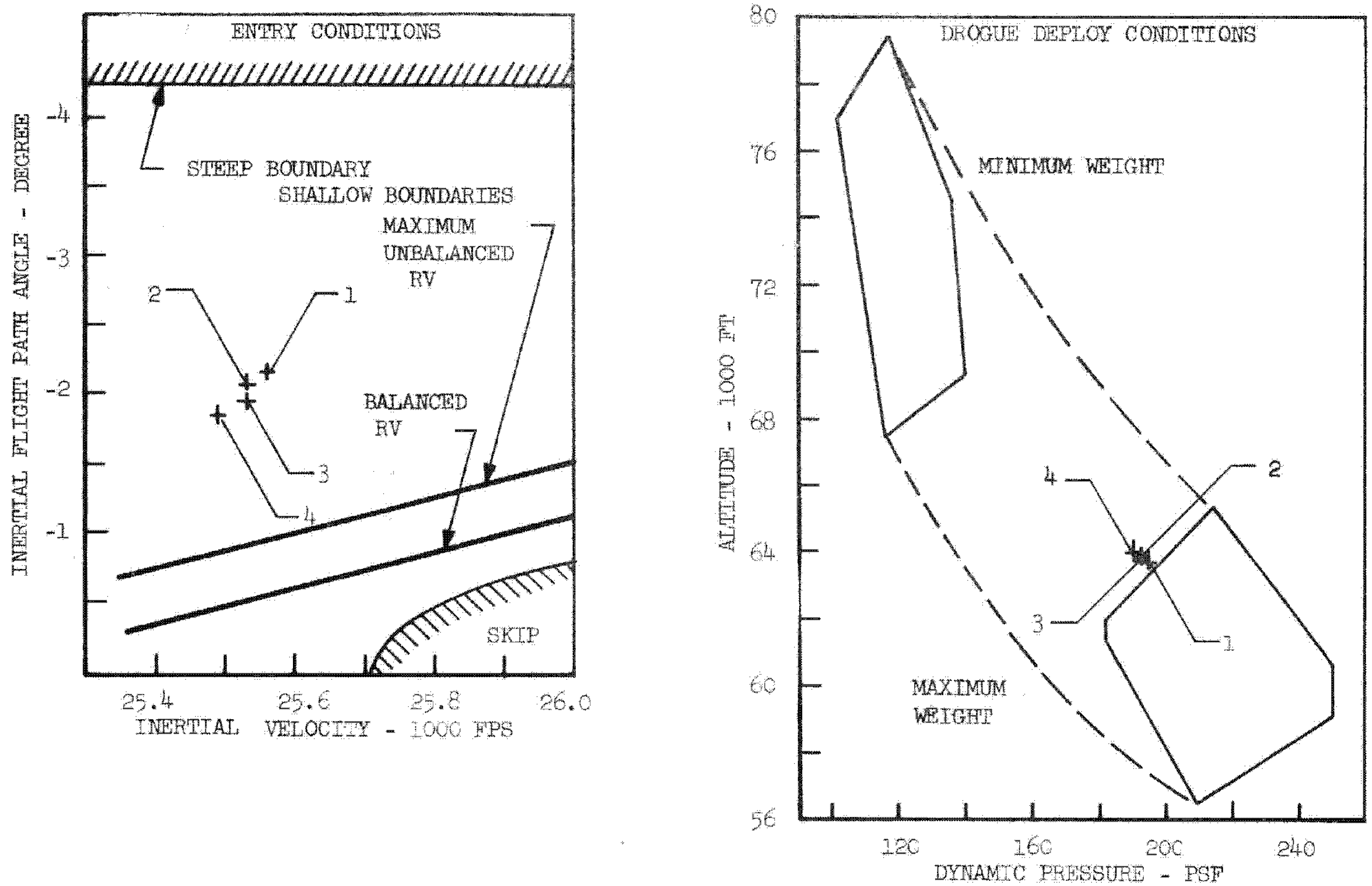

Figure 5-1

SV-9 Reentry Parameter Comparisons

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TABLE 5-2

RV SUBSYSTEM PERFORMANCE SUMMARY

RV SUBSYSTEM/FUNCTION	PERFORMANCE ASSESSMENT
On Orbit Thermal Protection	Normal o T_{PL} Container = $T_{REF}{}^{+0°F}_{-5.6°F}$ o Power Usage (Watts/RV) Maximum 19.3 (First day in orbit) Stabilized 7.7 (Ninth day in orbit) Allowable 20
Trim and Seal	Normal*
Electrical Power and Distribution	Normal* o All batteries activated. o All voltages at least 24.8 volt open circuit voltage.
Structure	Normal*
Pyro Subsystem	Normal*
Spin Stabilization	Normal
Retro Motor	Normal
Tracking, Telemetry and Instrumentation	Normal
Heat Shield	Normal
Base Thermal Protection	Normal*
Sequential	Normal*
Recovery	Normal

* RV-4 not available for post flight inspection at time of report. Inspection expected to prove normal performance similar to other RVs.

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5.3 REENTRY VEHICLE SUBSYSTEM PERFORMANCE

Post flight analysis indicated that aerial retrieval of RV-2 and RV-3 resulted in marginal recoveries.

- RV-2 Damaged Load Loop
  On RV-2, a retrieval hook tine split the two ply load loop and a single ply of webbing carried the entire load. The ply was severely damaged since the design is based on engagement of both plies. Failure would probably have resulted in reinflation of the parachute and a water recovery. Redesign of the hook tine from the current 0.125 inch radius to a larger radius would minimize this type occurrence but could introduce other problems.

- RV-3 Main Parachute Retrieval Hit
  Review of the recovery aircraft film coverage of the RV-3 retrieval showed the lower load loop and hooks made contact with the main canopy instead of the cone and were responsible for the excessively torn condition of the main chute. Recovery was made by hook engagement of the 4,000 pound parachute vent band instead of the 18,000 pound load loop. A successful recovery using the parachute vent band is not considered reliable. If the hook had torn through the band, reinflation of the parachute would have been unlikely and could have resulted in loss of the RV.

5.4 STELLAR TERRAIN RECOVERY (RV-5)

RV-5 (S/N 1805) was successfully recovered on Rev 958. Recovery statistics are shown in Table 5-3. All RV subsystems performed normally. The SV provided a satisfactory pitch angle after a yaw reverse and all other interface functions were nominal.

The predicted impact point (PIP), the estimated point of parachute deployment (EPPD) and the air snatch point are shown in Figure 5-2. The miss distance between the PIP and EPPD was calculated to be 0.49 nm short and 1.03 nm East of the ground track. The capsule was recovered at 14,600 feet on the first pass.

TABLE 5-3

ST-RV (RV-5) RECOVERY SUMMARY

Recovery Rev	958
Date	27 December 1975
Payload Weight (100%)	67.82 pounds
SV Recovery Orbit	
Perigee (nm)/Apogee (nm)/Argument Perigee (deg)	87.26/151.96/116.64
SV Pitch Angle (after yaw around) (deg)	-64.7

	PIP	EPPD	Air Catch
Latitude	23° 59.7'	24° 00'	23° 59'
Longitude	149° 07.2'	149° 06'	149° 00'
Altitude	---	---	14,600 ft

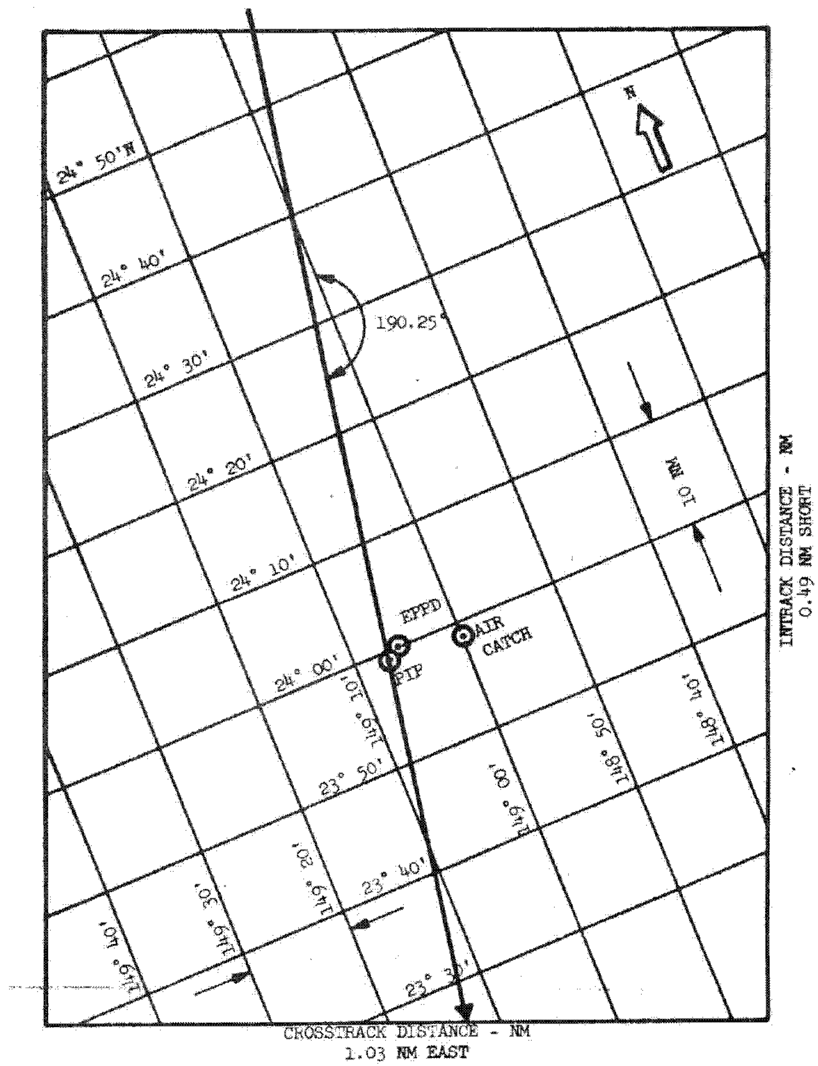

Figure 5-2

ST-RV (RV-5) Recovery Locations

APPENDIX A
GLOSSARY OF TERMS

ACS	Attitude Control System
ATC	Active Thermal Control
BV	Booster Vehicle
BV/SV	Booster Vehicle/Satellite Vehicle
DBS	Doppler Beacon System
ECS	Extended Command System
EDAP	Electrical Distribution and Power
EPPD	Estimated Point of Parachute Deployment
FCEA	Flight Control Electronics Assembly
GFE	Government Furnished Equipment
H/S	Horizon Sensor
HSA	Horizon Sensor Assembly
IRA	Inertial Reference Assembly
LB II	Lifeboat
MCM	Mapping Camera Module
MCS	Minimal Command System
OA	Orbit Adjust
OB	Optical Bar
OOAA	On-Orbit Adjust Assembly
PACS	Primary Attitude Control System
PCM	Pulse Code Modulator
PDWN	Pitch down

APPENDIX A (Continued)

PFA	Post Flight Analysis
PIP	Predicted Impact Point
PST	Pacific Standard Time
RACS	Redundant Attitude Control System
RCS	Reaction Control System
REA	Reaction Engine Assembly
REM	Reaction Engine Module
Rev	Revolution
RV	Reentry Vehicle
SBA	Satellite Basic Assembly
SECO	Stage II Engine Cut-Off
Sep	Separation
SGLS	Space Ground Link System
SMCD	Spin Motor Current Detector
Solo	Systems Engineering Test after fourth RV separation
SPC	Stored Program Command
SRM	Solid Rocket Motor
SS	Sensor Subsystem
SSU	Subsatellite Unit
ST	Stellar Terrain
ST-RV	Stellar Terrain-Reentry Vehicle
SV	Satellite Vehicle
T/R	Tape Recorder
TT & C	Telemetry, Tracking and Command
TVC	Thrust Vector Control
VCTS	Vehicle Command and Transponder Set