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FLIGHT TEST ENGINEERING ANALYSIS REPORT

FOR

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

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SV-8 45 Day Report

Prepared and Submitted by the

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FOREWORD

This report describes the performance of the eighth HEXAGON Program Satellite Vehicle (SV-8). The vehicle was launched 10 April 1974, and after a 105 day primary mission and a 4 day Solo mission, was deboosted on Rev 1768 on 28 July 1974.

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 eighth HEXAGON Satellite Vehicle (SV-8) was placed in a nominal 85.5 X 165 nm orbit by the Titan IIID Booster on 10 April 1974. 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 S73-7 Subsatellite (+Y) was properly ejected on Rev 3. The SSU (-Y) Subsatellite was properly ejected on Rev 13. The Panoramic Camera operated throughout the mission and its RVs were recovered on Revs 228, 681, 1118 and 1701 which occurred on Mission Days 15, 43, 70 and 106. RV 1 was recovered from the water when the hook failed to engage a load loop, but the other three RVs were caught in the air. All of the film was transported into the RVs including 2600 feet of SO-255 color and 3000 feet of FE-3916 infrared color film in RV 4. Operations were normal until Rev 1268 when a failure of the take-up integrator servo to reset required elimination of nested operations for the remainder of the mission. All the mapping camera operations were normal and 100 percent of the film was transported to RV 5 which was aerially recovered on Rev 973. Solo Tests were run and the SV was deorbited on Rev 1768 (during Mission Day 110).

SECTION II
MISSION OVERVIEW

2.1 PREFLIGHT PLANNING

Mission 1208 was the second of the HEXAGON Block II vehicles and panoramic camera systems. With the leak free performance of the soft seat RCS valves proven on 1207, all the RCS Tanks were filled on 1208 and both the Primary and Backup Thrusters were initially supplied from RCS Tanks permitting a closer watch on RCS fuel usage. The OA Tank was loaded with 3205 pounds of fuel. The Quantic Horizon Sensor Experiment was carried on this mission and was conducted throughout the mission including Solo. In addition, sixteen accelerometers and an FM/FM Telemeter package were utilized on ascent to ascertain dynamic characteristics of the SV.

2.2 PREFLIGHT CONSTRAINTS

The Mission 1208 orbit was designed to:

A. Maintain solar angle (Beta) within -8° to +30° for the planned 93 days.

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

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 12 March 1974. The vehicle was launched on 10 April 1974 at 12:20:00.7 PST at the opening of the launch window.

During launch pad testing, a leak was determined between the High Pressure Isolation Valve (HPIV) and the regulator on the "A" side of the Sensor Subsystem Pneumatics Module. Since the leak was small it was decided to launch "as is" without any repair. To assure that leakage remained low, it was planned to close the A-side HPIV for the duration of the mission immediately after uncage and optical bar stowing.

2.4 ASCENT

The BV successfully injected the SV into an 85.55 X 164.72 nm orbit. The achieved orbit was close to nominal with the deviations shown as follows:

Apogee Altitude (nm)	+2.315
Perigee Altitude (nm)	+0.100
Period (second)	+1.96
Eccentricity	+0.000204
Argument of Perigee (degrees)	-2.235
Inclination (degree)	+0.010

2.5 ORBIT AND RECOVERY

2.5.1 1208-1 (Fourteen Days Duration)

Solar Array deployment was executed over INDI on Rev 1 with normal deployment and erection. At KODI Rev 1 the Solar Arrays were repositioned from +18° to 0°. The S-73-7 Subsatellite was ejected on Rev 3 and the SSU Subsatellite was ejected on Rev 13. The DBS Antenna was deployed on Rev 12.

Operational photography began on Rev 5 following successful completion of constant velocity and health checks. The A side High Pressure Isolation Valve of the SS pneumatics module was closed just after SV/BV separation as planned to minimize gas loss from the leak detected prior to launch. Normal operation was exhibited throughout this segment. No gas leakage was detected, the consumption rate being the nominal 0.023 pounds per minute. Approximately 28,000 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. The very good imagery was obtained only on the aft looking camera with clear weather. Focus and On-Orbit Adjust Assembly (OOAA) adjustments were established for the forward and aft looking cameras respectively.

2.5.2 <u>1208-2 (Twenty-Eight Days Duration)</u>

Normal operational photography continued throughout this segment. PFA determined focus and OOAA adjustments were implemented on Revs 281 and 291 respectively. The HPIV-A was opened during Revs 480 through 496 to obtain more data on the pneumatics leak. Evaluation indicated a very small, low pressure leak which could be tolerated; therefore, the pneumatics module was restored to normal configuration on Rev 563 by opening HPIV-A. Approximately 28,800 feet of forward looking camera film and 27,700 feet of aft-looking camera film were exposed and stowed in RV-2.

PFA showed overall quality of acquired photography to range from very good to poor but with the majority rated as fair to good. Aft-looking camera performance was still subjectively better than that of the forward-looking. The improved imagery was attributed both to the adjustments and better atmospheric conditions. Forward looking camera OOAA adjustments and snow bias exposure criteria changes were established to further improve image quality.

2.5.3 1208-3 (Twenty-Seven Days Duration)

This segment began with normal operational photography. PFA determined forward-looking camera OOAA adjustment and exposure criteria changes were implemented on Rev 752. An emergency shutdown (ESD) occurred on Rev 980 with with a failure of the forward-looking camera verification interlock signal. Subsequent analysis showed the cause to be a missing builder roller down verification signal. Forward-looking camera verification interlock disable (VIA-DIS) was commanded and operations resumed on Rev 996. No further problems occurred during the remainder of this segment.

Approximately 25,200 feet of forward-looking camera film and 24,300 feet of aft-looking camera film were exposed and stowed in RV 3.

PFA showed overall quality of the acquired photography to range from very good to poor. The poor quality was attributed generally to poor atmospheric conditions, very high sun angles, specular reflections and shadowless imagery. The aft-looking camera continued to be superior to the forward-looking with most of the very good imagery being on the aft-looking camera.

2.5.4 1208-4 (Thirty-Six Days Duration)

Normal photographic operation was obtained throughout this segment except for a malfunction of the aft-looking camera take-up on Rev 1268. Analysis showed the cause to be a failure of the take-up integrator servo to reset at TU brake-off. Operations were resumed on Rev 1300 but compensations for the failure required elimination of nested operations for the remainder of the mission.

Forward and aft-looking camera film depletion occurred on Rev 1694 and Rev 1700 respectively.

Approximately 26,800 feet of forward-looking camera film and approximately 26,600 feet of aft-looking camera film including 2,588 feet of SO-255 (color) and 3036 feet of FE-3916 (infrared color) film, were exposed and stowed in RV 4.

PFA showed results similar to these of Segment 3 with the majority of imagery rated fair to good. The overall image quality of the color and infrared color photography was rated good.

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	Prelaunch SS Pneumatics Leak	HPTV-A closed after SV/BV separation. Opened on Day 15 after leak found tolerable.	2.3, 2.5.1 and 4.1.5
1	ARPA-101 payload failed to eject from the S73-7 Subsat.	No effect on HEXAGON mission. Subsatellite mission aborted.	4.3
15	Aerial recovery hook failed to engage load loop.	RV 1 recovered from water.	5.1
43	Loose wire in Burndy Terminal Block on RV 2.	No mission impact. Discovered in PFA.	5.3
61	ESD caused by failure of A-Camera verification interlock signal.	Disabled interlock on Day 62. Normal operations resumed.	2.5.3 and 4.1.1
64	PCM Remote Unit 4A failed.	Switched to RU 4B. Review indicates no generic problem	3.7.2.1
67	Thrusters 1 and 8 seriously degraded.	Degradation as expected. Switched to RCS-2.	3.3.3
79	TU Integrator servo failed to reset at TU brake-off.	Operations resumed on Day 83 with nested operations eliminated for rest of mission.	2.5.4 and 4.1.1
89	SGLS 2 showed loss in signal strength.	No effect on mission. Each Block II Cubic Transmitter to be reviewed.	3.7.1
92	Noise on IRA roll gyro output.	Transfer control to ACS 2. Attributed to random failure in FEBs.	3.2.6
98	Intermittent transients on PACS HSA.	No mission impact. Attributed to random electronics failure.	3.2.7
106	RV 4 Main Battery vented.	No mission impact. Corrective action began after SV-7 anomaly.	5.3

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; however, the PACS IRA Roll Channel and Horizon Sensor outputs were sporadically abnormal near the end of the flight. These abnormalities are discussed in Paragraphs 3.2.6 and 3.2.7.

3.2.1 BV/SV Separation

BV/SV separation was complted at approximately 549.5 seconds vehicle time (vehicle time started 67.68 seconds prior to liftoff). Master Clear Off, which enables the pitch, roll and yaw integrators to accumulate angle, was at 512.8 seconds and SECO, which terminates BV attitude control, occurred at 537.5 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 (capture).

TABLE 3-1

BOOSTER VEHICLE/SATELLITE VEHICLE (BV/SV) SEPARATION

Axes	RATE AND ATTITUDE AT BV/SV SEPARATION						CAPTURE			
	RATE (deg/sec)		ATTITUDE (DEGREES) H/S at Sep		ATTITUDE (DEGREES) (SECO to Sep)		ATTITUDE		RATE	
	Spec	Actual	Spec	Actual	Spec	Actual (5) H/S/ Integrator	Spec (1) (deg)	Actual (2) (Time in Sec)	Specified (3) (deg/sec)	Actual (4) (Time in Sec)
Pitch	±0.752	-0.128	+13.0 to -21.7	2.80	±3.5	0.25/-0.762	±0.70	(6)	±0.014	(6)
Roll	±0.786	-0.181	±10.6	3.84	±3.5	1.20/0.85	±0.70	(6)	±0.021	(6)
Yaw	±0.752	+0.134	+11.1 to -11.4	---	+4.5 to -3.5	---/1.83	±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 plus 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) Normal performance indicating the pointing requirements are satisfied was observed at a nominal settling time of 520 seconds after the commanded switch to fine mode (657 seconds after separation). The total 1177 seconds is well within the spec of 1500 seconds and no closer study was performed.

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3.2.2 Subsatellite/SV Separation

The Subsatellite/SV separation events of Rev 13 were as follows:

Event	Vehicle Time
Start Neg Yaw Maneuver	70036.2 sec
Stop Yaw	70063.0 sec
Separation	70122.8 sec
Start Pos Yaw Maneuver	70140.0 sec
Stop Yaw	70166.8 sec

The ACS parameters just prior to the instant of separation (70122.8 seconds vehicle time) are presented in Table 3-2.

TABLE 3-2

RATE AND ATTITUDE PARAMETERS AT SUBSATELLITE SEPARATION

PARAMETER	SPECIFIED	ACTUAL
Pitch H/S	±1.0 deg	+0.22 deg
Roll H/S	±1.0 deg	-0.68 deg
Roll Integrator	---	-0.24 deg
Yaw Integrator	---	-0.16 deg
Pitch Integrator	---	+0.24 deg
Yaw Attitude	(-18.9 deg desired)	
Pitch Gyro Rate (1)	±0.1 deg/sec	-0.07 deg/sec
Roll Gyro Rate	±0.1 deg/sec	+0.03 deg/sec
Yaw Gyro Rate	±0.1 deg/sec	0
Maximum rates following separation:		
Pitch Gyro Rate (1)		-0.07 deg/sec
Roll Gyro Rate		+0.21 deg/sec
Yaw Gyro Rate		+0.08 deg/sec

(1) Geocentric Program Rate is connected.

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3.2.4 Mapping Camera Module (MCM) Operations

3.2.4.1 MCM Calibration Maneuvers

The calibration maneuvers on Rev 965 consisted of a negative pitch maneuver of 160.1 degrees, followed by four inertial periods, one for each calibration. The durations of the inertial periods were 160, 100, 120, and 110 seconds for Calibrations 1, 2, 3 and 4 respectively. The desired pitch attitudes at the beginning of each calibration were -160, -149, -142 and -134 degrees 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 fine mode, the horizon sensor was reading +0.18 degree, indicating successful execution of the calibration sequence. Table 3-4 presents the four calibrations. The settling time for Calibration 1 is the time from initiation of the pitch down maneuver to the start of Frame 1. The settling time for Calibrations 2, 3 and 4 is the time from the removal of geocentric rate to the start of Frame 1 in each calibration.

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 157167.2 on Rev 973) are listed in Table 3-5.

3.2.5 Recovery

The pitch down maneuvers preceding RV-1 through RV-4 separations were all within specification and are summarized in Table 3-6. The RV separation performance summary is shown in Table 3-7. It appears that the RV-4 separation impulse may have been in excess of the 166 lb-sec specification limit.

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	140	117121.992	395.2	+0.005	±0.009	-0.001
2	80	117580.450	16.65	+0.004	±0.011	-0.001
3	100	117980.450	16.65	+0.006	-0.012	-0.001
4	320 (1) (80 usable)	118400.450	16.65	+0.006	±0.012	+0.001
Specified	Not to Exceed 300	---	600 Allowed	±0.014	±0.021	±0.014

(1) Long duration to use up film.

TABLE 3-5

RATE AND ATTITUDE PARAMETERS AT RV-5 SEPARATION

Axes	Attitude (degrees)	Source	Rate (degrees/second)
Pitch	-61.4	PDWN	+0.067 (includes Geocentric Program Rate)
Roll	-0.12	H/S	0
Yaw	+0.10	Integrator	0

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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)
1/228	-33.4	-32.7	150	73	-0.705	-0.75 ±0.05	-0.73
2/681	-37.8	-35.8	150	80	-0.705	-0.75 ±0.05	-0.73
3/1118	-37.5	-35.7	150	76	-0.705	-0.75 ±0.05	-0.73
4/1701	-38.7	-36.5	150	79	-0.705	-0.75 ±0.05	-0.72

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 (slug-ft^2)	Pitch Thruster Moment Arm (ft)	ROLL ANGLE	
								Spec (deg)	Meas H/S (deg)
1/228	1.14	4.1	107	-32.7	98.0	146446	27.13	±1.0	-0.10
2/681	1.36	7.1	133	-35.8	98.2	123483	22.01	±1.0	-0.06
3/1118	1.44	4.7	134	-35.7	99.1	94613	17.68	±1.0	-0.06
4/1701	1.40	5.1	172	-36.5	100.0	83840	11.92	±1.0	-0.08

3.2.6 IRA 1021 Anomaly

The ACS performed normally until substantial noise on the roll gyro output of the PACS IRA 1021 was observed on Rev 1487. The noise increased on subsequent revs; therefore, the decision was made to transfer control to RACS and this was done on Rev 1501. Just prior to the transfer, the noise disappeared and did not reoccur during the remainder of the flight.

Examples of the noise are shown in Figures 3-1a and 3-1b and a normal analogue rate record in Figure 3-1c. Figure 3-1b shows the transition from the normal rate of 0.001 to 0.002 degrees/second to a noisy output of approximately 0.015 degrees/second peak to peak. Figure 3-1a shows occasional large noise spikes in addition to noise which is similar to that in Figure 3-1b; the maximum spike observed was 0.1 degree/second peak to peak. Some of the large spikes resulted in error signals which activated a thrust valve as can be seen in Figure 3-1a.

A failure mode analysis of IRA 1021 concluded the most probable cause of the noise to be a random failure in the demodulator amplifier or a gain change amplifier FEB (Functional Electronic Blocks). Problems have been encountered in the past with leaky FETs (Field Effect Transistors), with intermittent FEB pin connections and cracked signal traces on FEBs.

FEBs in the demodulator and gain change amplifiers have been replaced with discrete piece parts on IRA 1026 and subsequent units.

3.2.7 HSA 1007 Anomaly

On Rev 1586 transients were noted on PACS HSA 1007. Intermittent transients were observed on subsequent revs. The vehicle attitude was not affected since control had been transferred to RACS.

Samples of the transients are shown in Figure 3-2. The transient magnitudes vary but exhibit a 5 second exponential decay after the transient peak. Other data indicates transients as large as -1.8 degrees (coarse mode

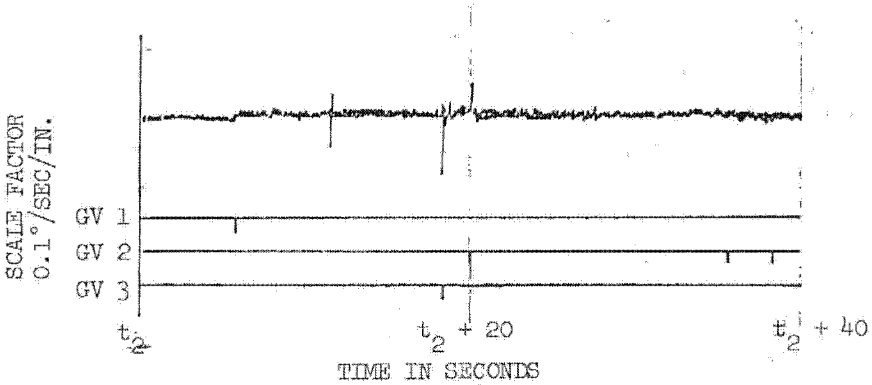


Figure 3-1a. Spikes Causing Gas Valves to Fire

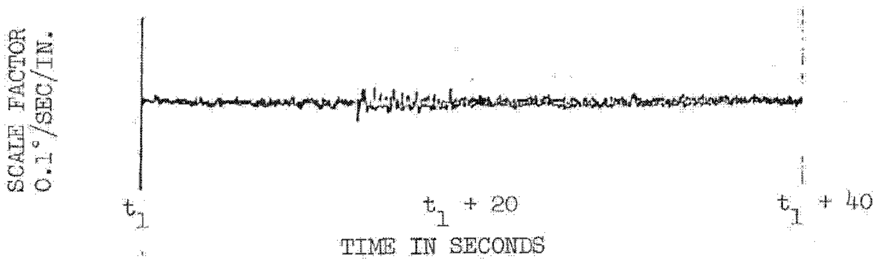


Figure 3-1b. Onset of Noise - Not Sufficient to Fire Gas Valves

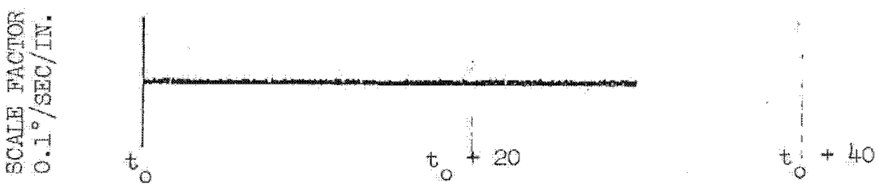
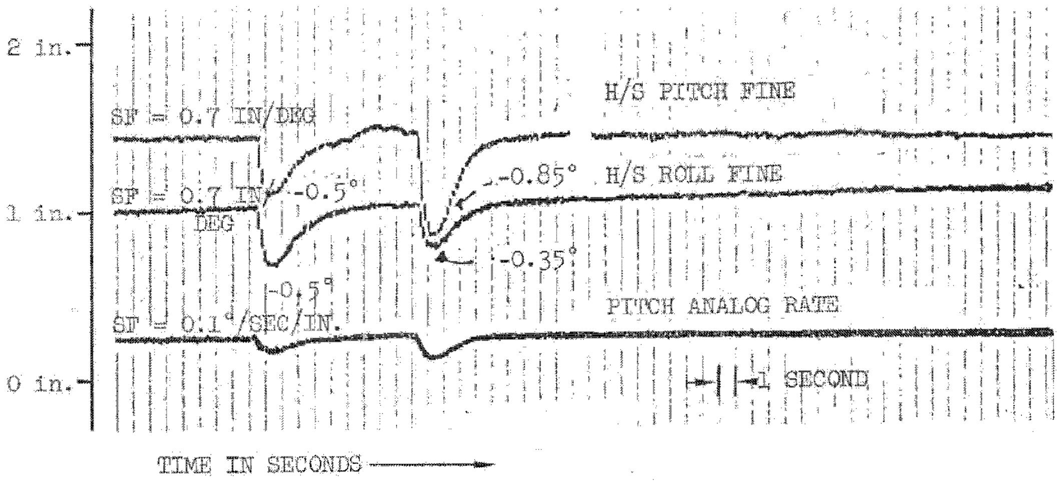


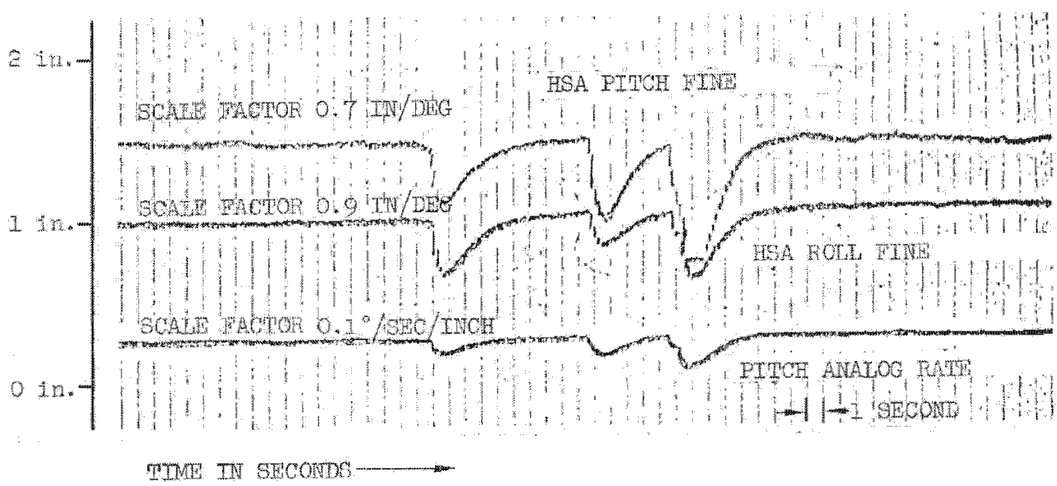
Figure 3-1c. Typical Normal Operation Without Noise

Figure 3-1

Examples of IRA 1021 Roll Rate Noise - Rev 1497



REVOLUTION 1643



REVOLUTION 1740

Figure 3-2
HSA 1007 Transients on Revolution 1643 and 1740

TM) in pitch and -0.85 degrees in roll. Since the pitch HSA signal is fed directly into the pitch rate channel, the transients are also seen on the pitch analogue rate output. These transients are large enough to have fired the thrusters if PACS had been the controlling system.

A failure mode analysis indicated the most probable cause of the transients is either a random failure in the comparator module of the right radiance channel of the HSA Electronics Box or a random failure in the right head electronics which results in a negative spike into the Mixer Box.

3.3 REACTION CONTROL SYSTEM (RCS)

3.3.1 Flight Summary

SV-8 was the second vehicle with a complete set of soft seat valves for the thrusters. Propellant was drawn from RCS Tanks 1 and 2 until Rev 938 when the supply was switched to the OAS Tank. Control was maintained by the Primary RCS through Rev 1082 when the control function was switched to the Standby System (RCS-2) because the thrust on REAs 1 and 8 had seriously deteriorated. The flight was continued with the Standby Thrusters drawing propellant from RCS Tanks 3 and 4 until Rev 1742 when the supply was switched to the OAS Tank.

REA and REM temperatures gave no indication of leakage throughout the flight.

3.3.2 Propellant Consumption

Over the first 938 revs the RCS propellant consumption averaged 3.0 pounds/day on the Primary RCS. The Standby System averaged 4.0 pounds/day over 685 revs that included the [redacted] and Solo activities. (b)(1) (b)(3)

3.3.3 Thruster Performance

RCS-1 thrust levels were determined using individual REA chamber pressures. Figure 3-3 is a history of the normalized thrust. The dashed lines are the ±8 percent run to run tolerance allowed. As can be seen REAs 1 and 8 were seriously degraded while REA 7 was just starting to degrade at Rev 1083.

Table 3-8 shows the thrust values for RCS-2 calculated from gyro rates along with the thrust attained during the qualification tests at fuel supply pressures corresponding to the feed pressures available on the pertinent revs. The total thrust for the pair of thrusters firing was approximately twice the qualification level. The fact that the thrust values for REAs 2 and 7 are consistently low and REAs 3 and 6 are consistently high could indicate a small error in the cg location or an error in the determination of the roll gyro rate rather than the difference in thrust shown.

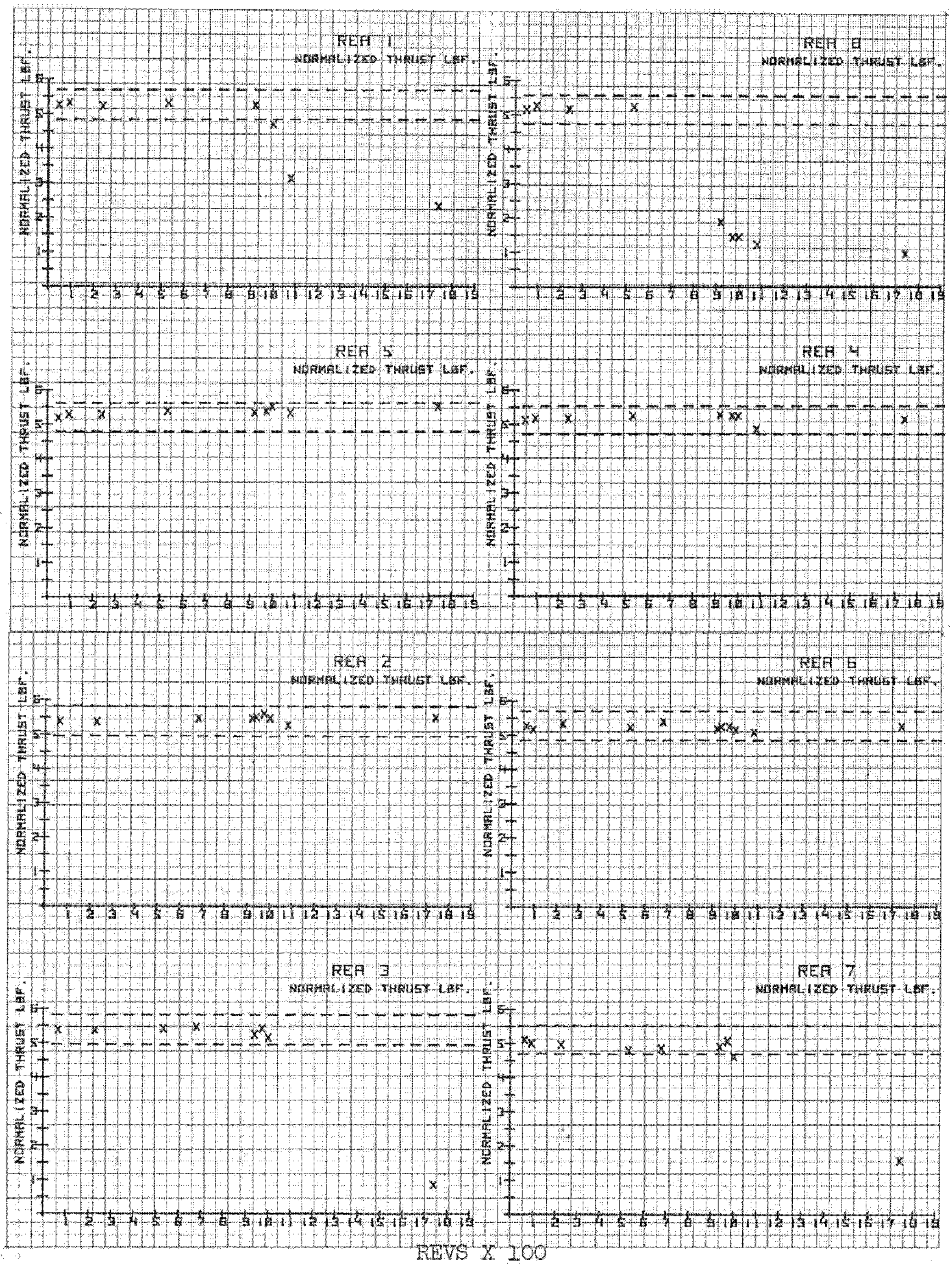


Figure 3-3
Normalized Thrust History

TABLE 3-8

RCS-2 THRUST LEVELS COMPUTED FROM GYRO RATES

Rev	THRUST IN POUNDS								Qualification Level (1)
	REA NUMBER								
	1	2	3	4	5	6	7	8	
1099	5.29	4.35	5.99	4.99	4.36	5.31	5.46	4.98	5.28
1118		4.72	4.80			5.74	4.38		5.03
1218	4.97	3.58	5.24	4.03	4.41	5.71	3.99	4.77	4.49
1408	3.99			3.18	3.22			3.57	4.00
1633		3.41	3.54			3.65	3.90		3.61

(1) Qualification level is the thrust that was obtained during the qualification tests at the fuel supply pressure available on the rev listed.

The estimated total pulse count for RCS-1 through Rev 1083 is shown below:

REA	TOTAL COUNT	AVERAGE COUNT/DAY
1	22500	338
2	8000	120
3	83000	1240
4	5000	34
5	6000	40
6	6600	79
7	117000	1750
8	5000	34

The technique used to estimate the total pulse count is restricted by the sample rate attainable. Past flight tests have shown that on the high duty cycle engines the total count may be accurate within ±30%. The total pulse count on the low duty cycle engines is only accurate enough to establish them as low duty cycle.

3.4 ORBIT ADJUST SYSTEM (OAS)

3.4.1 Orbit Control

The Orbit Adjust System was utilized a total of 46 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 3-9, OAs occurred every three days with an adjustment of perigee location every nine days.

3.4.2 Deboost

The deboost was successfully accomplished with one 231 second firing, followed by a firing to obtain propellant depletion. The depletion was observed to occur at approximately 185 seconds into the burn, close to the predicted time. Engine performance was normal during the deboost and depletion operations.

TABLE 3-9

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	46	13439	19.36	19.79	+2.18
2	94	30404	45.03	44.99	-0.08
3	96	15576	-23.00	-23.16	+0.70
4	143	14966	22.32	22.33	+0.04
5	192	16320	24.45	24.44	-0.04
6	240	26036	42.20	42.21	+0.03
7	242	11758	-18.99	-19.14	+0.80
8	289	13283	21.76	21.66	-0.42
9	337	16234	26.62	26.60	-0.05
10	386	25785	42.34	42.47	+0.30
11	388	11608	-19.20	-19.20	-0.03
12	434	14707	24.32	24.41	+0.34
13	483	15759	26.91	26.25	-2.47
14	532	24018	40.15	40.20	+0.14
15	534	10731	-17.95	-18.01	+0.36
16	580	13188	22.15	22.23	+0.38
17	629	17222	28.84	29.15	+1.05
18	683	11571	21.21	21.39	+0.89
19	726	27774	51.41	51.63	+0.43
20	728	12771	-23.89	-23.87	-0.11
21	774	11721	21.87	21.98	+0.52
22	823	13136	24.54	24.72	+0.74
23	871	15666	29.53	29.60	+0.24
24	920	25151	47.73	47.80	+0.15
25	922	12648	-24.17	-24.16	-0.04

TABLE 3-9 (Continued)

OA Firing Number	Revolution Number	Impulse Delivered (lb-sec)	Planned Δ V (ft/sec)	Achieved Δ V (ft/sec)	Δ V Error (percent)
26	975	13033	25.43	25.60	+0.67
27	1017	14719	28.47	29.03	+1.98
28	1065	11320	22.20	22.43	+1.05
29	1120	20057	43.79	43.86	+0.17
30	1162	21596	47.65	47.69	+0.09
31	1164	14814	-33.15	-32.89	-0.78
32	1211	11396	25.60	25.42	-0.70
33	1260	14556	32.72	32.63	-0.28
34	1282	1768	4.03	3.97	-1.48
35	1313	12052	26.67	27.15	+1.82
36	1357	10751	24.28	24.32	+0.16
37	1406	28112	64.24	63.98	-0.40
38	1408	8177	-19.49	-18.71	-3.96
39	1502	18312	41.90	42.14	+0.58
40	1551	11987	27.78	27.72	-0.22
41	1600	12528	28.80	29.10	+1.06
42	1616	3072	7.18	7.15	-0.32
43	1649	5115	12.02	11.94	-0.70
44	1704	32172	83.00	85.13	+2.57
45	1768	31000*	-87.30	---	---
46	1768	26000*	---	---	---

*Approximate values.

3.5 LIFEBOAT II SYSTEM

On SV-8 the Type 29 Batteries were carried in Bays 3 and 4 and, therefore, had less influence on the Lifeboat Magnetometers in Bay 9. The equivalent attitude errors of the Q, P and R magnetometers were less than 0.5 degrees. The three rate gyros were within 0.05 deg/sec of the rates monitored on the ACS gyros. On Rev 1474 a calibration test was conducted prior to a Solo test wherein Lifeboat II would control the vehicle. The calibration showed the small induced magnetic distortion effects fall within the error allocation for such errors.

No anomalies were noted. Lifeboat Magnetometer prediction checks are shown in Table 3-10.

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 2.8% (average of four legs) after 107 days (1701 revs) in orbit.

3.6.2 Main Bus Voltage

The Main Bus voltage varied from a low of 26.6 to a high of 31.3 volts. The allowable range is 25.5 to 33.0 volts. Low range voltage was obtained during payload operations with a bus load of 62 amps. High voltage data was gathered during charge cycles.

3.6.3 Power Capability and Usage

Power usage ranged from 325 to 387 amp hours/day. The 390 amp hour/day capability was not exceeded during this mission. Excess capability was demonstrated with K2s occurring a minimum of once per day.

All Type 29 Batteries operated in a desirable environment (44°F to 51°F) and performed normally throughout the mission.

TABLE 3-10

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
18.4	S-N, DB	-24	-30.1	212	214.1	Not in Use		-0.08	-0.068				
25°N	S-N, RX	-25	-29.6	Not in Use		Negative Saturation							
163°W	N-S, RX	-24	-29.5	Not in Use		Negative Saturation							
228.4 57°N 147°W	N-S, RX	-39	-44.3	Not in Use		137	135.6	1.14	1.11	-0.13	-0.11	0.05	0.002
1118 56°N 144°W	N-S, RX	-45	-47.4	Not in Use		174	173.9						
1701 59°N 138°W	N-S, RX	-45	-47.7	Not in Use		212 to 206	211.9						

* DB indicates deboost, RX indicates recovery.

** Corrected for an instrumentation bias of -4 milligauss.

3.6.4 Pyro Battery Performance

Pyro Battery 1 stabilized at 49°F to 50°F which minimized self discharge during the mission. Lift off capacity was 11.27 amp-hours. After 107 days and additional usage for instrumentation and self discharge, a residual capacity of 5.84 amp-hours remained. Cell degradation life still available was 11 days.

Pyro Battery 2 followed the same pattern with the exception of 7 days cell wearout life remaining.

3.6.5 Lifeboat Battery Performance

The Lifeboat battery operated normally in a 47°F environment throughout the entire mission. A total of 229 amp-hours remained at the end of 107 mission days from an initial 360 amp-hours at launch. Remaining cell degradation life was 10 days.

3.7 TRACKING, TELEMETRY AND COMMAND (TT & C)

3.7.1 Tracking

Tracking performance was satisfactory throughout the flight. Although there was no affect on the mission, reduced power output from the SGLS 2 Transmitter was reported after a health check on Rev 1426 Hula. Subsequent analysis revealed that the transmitter power level had been reduced as early as Rev 746 Hula. This anomaly is similar to a 25 to 30 dB reduction in transmitter power output experienced on SV-5 SGLS 2 Transmitter during Solo. The SV-5 anomaly was attributed to loss of pressurization in a sealed RF cavity of the transmitter output stages and resulted in adding a new second epoxy seal on the rear of the cavity and in additional leak rate testing of remaining transmitters. Loss of pressurization and resulting corona in the sealed cavity is also the most probable cause of the SV-8 anomaly, but this time the front seal of the cavity is suspected of being the source of the leak.

Once the transmitter low power output was recognized, a series of tests and health checks were made and are summarized in Table 3-11.

TABLE 3-11

SGLS 2 TRANSMITTER POWER TESTS

Rev/Sta	Maximum Elevation (degrees)	Maximum Signal Strength (dBm)	Antenna Diameter (ft)	Results	Type of Test for SGLS 2
10/K	9.1	-93	14		Health - SGLS 2 Only
18/P	6.3	-91	14		Health - SGLS 2 Only
18/H	14.0	-84	46		Health - SGLS 2 Only
163/H	4.8	-84	46		Health - SGLS 2 Only
536/H	8.4	-80	46	Decline in signal strength	Health - SGLS 2 Only
633/H	15.9	-77	46	Decline in signal strength	Health - SGLS 2 Only
746/H	45.8	-84	46	Decline in signal strength	Health - SGLS 2 Only
859/H	7.3	-96	46	Decline in signal strength	Health - SGLS 2 Only
1232/H	4.9	-95	46	Decline in signal strength	Health - SGLS 2 Only
1426/H	18.1	-96	46	Decline in signal strength	Health - SGLS 2 Only
1442/H	83.2	-91	46	Decline in signal strength	Health - SGLS 2 Only
1449/C	8.7	---	60	Decline in signal strength	Health - Switch SGLS 1 to 2
1465/C	20.0	---	60/46	Decline in signal strength	Health - Switch SGLS 1 to 2 to 1
1470/P	10.2	---	14	Decline in signal strength	Health - Switch SGLS 1 to 2 to 1
1539/H	10.2	---	46	Decline in signal strength	Health - Switch SGLS 2 to 1 to 2
1540/P	6.3	-114	14	See Note (1)	Threshold Test
1556/P	6.4	-117	14	No change	Non-coherent test
1571/P	6.8	-112	14	See Note (1)	Completed threshold
1588/H	51.0	---	46	Decline in signal strength	Health - Switch SGLS 2 to 1 to 2
1685/H	10.6	---	60	Decline in signal strength	Health - Switch SGLS 2 to 1 to 2
1754/B	12.8	---	46	Temperature change from 34 to 127°F	SGLS 2 Transmitter on for one rev.
1755/B	6.6	---			

NOTE (1) Threshold Test - The uplink carrier power was reduced to 0.1 watt with no command reject. This test verified that the SGLS 2 Antenna, coaxial cable between the antenna and multicoupler, and part of the multicoupler is good.

Figure 3-4 is a plot of the normalized signal strength measurements made at Hula with the 46 foot antenna throughout the mission.

Corrective action on remaining Block II Cubic Transmitters will be to individually evaluate each as to the need for additional leak rate tests, widening of a marginal gap on an RF Amplifier Board, and resealing and repressurization of the RF cavity.

3.7.2 Telemetry

3.7.2.1 General Performance

Telemetry system performance was satisfactory until Rev 1203 when PCM Remote Unit 4A (RU 4A) (Mid Section) failed to come on as programmed at the start of a Tape Recorder (T/R) record sequence. RU 4B was subsequently programmed on for a T/R record sequence between Pogo and Guam on Rev 1024. Playback revealed satisfactory receipt of normal data, and isolated the problem to RU 4A. Two revs of real time Mid Section data were lost before the telemetry system was configured to use RU 4B; however, this did not curtail or postpone any payload operations. A diagnostic test during the Rev 1037 Kodi pass in which RU 4A and RU 4B were both commanded on allowed an evaluation of the RU 4A power input circuitry via monitor H462, PCM Remote Unit 4A On/Off Monitor and revealed that the power input circuitry (fuse and on/off switch) was operational, but that the RU 4A did not reply to the PCM Master Unit request for data.

All later operations utilized Remote Unit 4B with the exception of Quantic Experiments for which RU 4A was programmed on, since the Mid Section data was not relevant to the Quantic operations. The use of RU 4B was restricted to support of the primary mission so as to reduce the number of on/off cycles and operational time. RU 4A failed to respond on any of the Quantic operations when it was programmed on.

All other operations of the telemetry system were satisfactory throughout the mission.

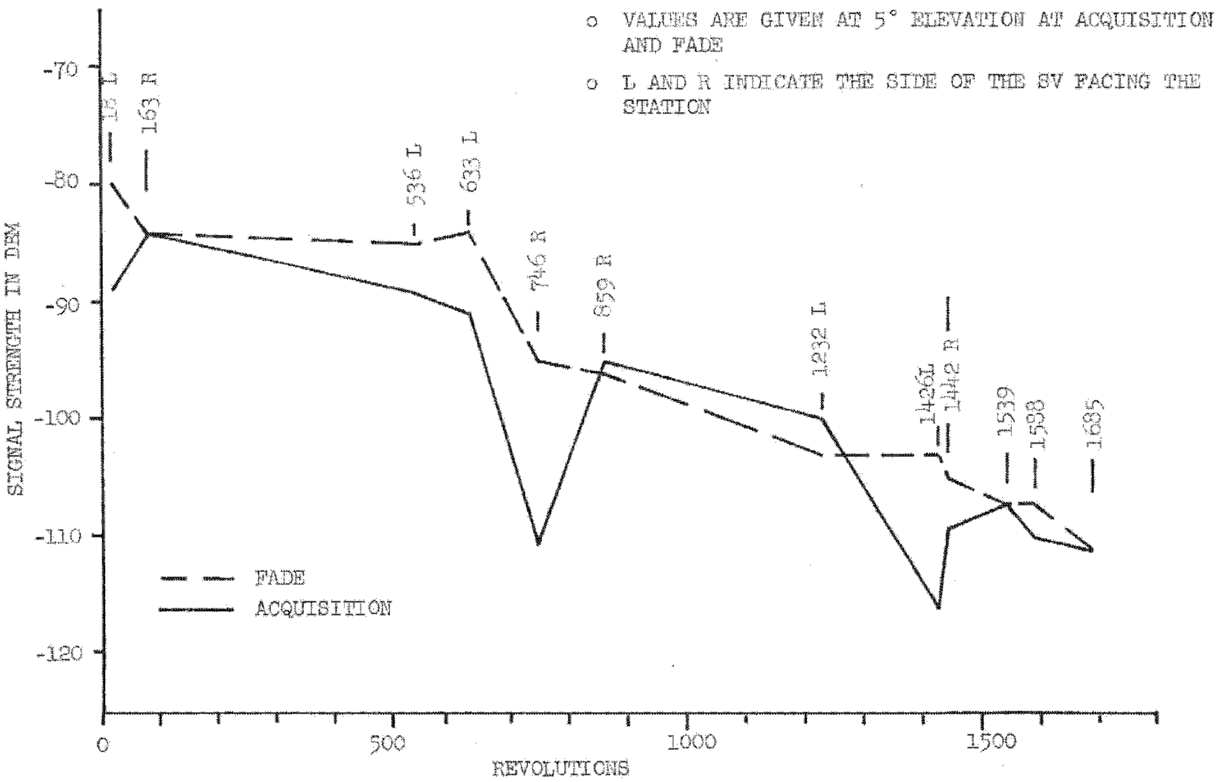


Figure 3-4

SGLS 2 Signal Strength History at Hula, 46 Foot Antenna

All RU failures, both Block I and II, that resulted in no reply or power supply failures and all such RU failures which have occurred at the vendors facility are being reviewed in an effort to isolate the most probable cause of the failure. As of this date, failure data has not revealed evidence of a generic problem.

3.7.2.2 Ascent FM/FM Telemetry

The performance of the ascent FM/FM Telemetry System via the SGLS 1 1.7 MHz FM Subcarrier Link was good except for the failure of one vibration transducer located on the Y Axis at Station 1642. Data from all other transducers was reported as being good.

3.7.3 Instrumentation

Table 3-12 presents the instrumentation anomalies at lift-off.

TABLE 3-12

INSTRUMENTATION ANOMALIES AT LIFT-OFF

Monitor Number	Item	Condition
S152	D Low SH + LS Temp 1	Open, 5.06 TMV
H370 H371	Time Word Converter No. 1 Voltage Time Word Converter No. 2 Voltage	Will read low by about 80 MV
B511	OAS Tank Temp 1	Single samples may read in error
P902	TP Drive 1B Sun Error	Randomly reads "0" TMV
P367 P368	CPA+ CPB+	May fail limits at C-

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

The utilization through Rev 1701 of selected T & T equipment is summarized below:

Primary Equipment	On/Off Cycle	Operating Time (hours)
SGLS 1	1952	195.44
T/R 1	7572	427.80
PCM 1 (1)	8180	571.60

(1) Includes both RU 4A and RU 4B time. RU 4A had approximately 5623 on/off cycles and 327.67 operating hours at the time of failure on Rev 1023.

3.7.5 Command

3.7.5.1 General

The vehicle SGLS command equipment was utilized to receive more than 19 million bits with no vehicle problem indications.

3.7.5.2 GFE Command System

A. Extended Command System (ECS)

The ECS responded satisfactorily in all command modes resulting in the loading of 245,657 SPCs in memory. Of these 104,216 SPCs were output by both PMUs for decoder processing. The remainder were erased prior to their time label matches. The only rejects experienced were due to RTS problems.

The accuracy and stability of the ECS Clock, as computed for each flight segment, are listed in Table 3-13.

TABLE 3-13

ECS CLOCK PERFORMANCE

Segment	Accuracy	Stability (Average 6 hour period)
1208-1	1.11 parts in 10^{7}	5.60 parts in 10^{10}
1208-2	8.10 parts in 10^{8}	2.59 parts in 10^{10}
1208-3	4.78 parts in 10^{8}	9.44 parts in 10^{11}
1208-4	1.97 parts in 10^{8}	1.34 parts in 10^{10}

B. Minimal Command System (MCS)

The MCS responded correctly to all commanding. The MCS was commanded into the Operate Mode on Rev 18.

C. 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.

One command was issued from the Backup Decoder on Rev 18.

D. Command System Usage Summary through Rev 1701

System	Total Operating Time - Hours
ECS	2502.8
MCS	6.0
Remote Decoder	8.3
Backup Decoder	0.05

There were no command system anomalies during the entire mission.

3.8 MASS PROPERTIES

A history of SV mass properties throughout the flight are tabulated in Table 3-14.

3.9 PREFLIGHT WINDS ALOFT LOADS ANALYSIS

Table 3-15 presents a chronological tabulation of the winds aloft computer runs for SV-8. The results are plotted in Figure 3-5.

A series of nine pre-flight winds aloft observations were made using Rawinsonde balloon soundings over a three day span, 8 April to 10 April 1974.

The 8 April 1974 results were within the established critical 100 percent criteria for all monitored parameters and resulted in a go for launch recommendation; however a ground equipment problem developed which caused a 24 hour launch delay.

TABLE 3-14

SV-8 MASS PROPERTIES

Description	Weight (lb)	CENTER OF GRAVITY (inches)			MOMENT OF INERTIA (slug-ft²)			PRODUCT OF INERTIA (slug-ft²)		
		$\bar{X}$	$\bar{Y}$	$\bar{Z}$	I_x	I_y	I_z	I_{xy}	I_{xz}	I_{yz}
At Launch	25582	1972.4	0.93	3.43	7496	211111	211158	-840	-421	31
At Injection	22905	1985.5	1.03	3.88	5486	181850	181801	-779	-150	-34
Sep from Stage 2	22676	1983.3	1.04	3.88	5314	179267	179229	-791	-159	-34
Arrays Deployed 0°	22676	1983.7	1.04	3.88	6514	180329	181553	-788	-150	-48
After Subsat Eject	21892	1989.8	1.02	4.50	6130	175098	176086	-808	372	-49
Prior to Drop 1	21446	1978.0	1.04	5.16	6095	179022	180036	-865	698	-52
After Drop 1	19914	2001.2	1.14	4.21	5889	146446	147601	-734	-634	-47
Prior to Drop 2	18964	1984.3	1.19	5.16	5850	144904	146084	-820	-485	-51
After Drop 2	17431	2005.7	1.30	4.08	5645	123483	124803	-716	-1564	-44
After Drop 5	16417	2005.2	1.38	5.35	5521	108088	109516	-715	-606	-53
Prior to Drop 3	16186	1999.6	1.40	5.71	5487	107785	109190	-746	-532	-47
After Drop 3	14721	2018.8	1.54	4.57	5294	94613	96148	-652	-1305	-39
Prior to Drop 4	13763	2001.0	1.65	5.30	5278	90224	91662	-736	-1050	-264
After Drop 4	12243	2017.1	1.85	3.79	5073	83840	85418	-660	-1631	-255
Prior to Deboost	12082	2014.9	1.87	3.64	5054	82880	84476	-671	-1558	-254
End Deboost	11836	2011.6	1.91	3.71	5053	81507	83105	-687	-1589	-255

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TABLE 3-15

WINDS ALOFT ANALYSIS SUMMARY

Parameters	BALLOON RELEASE TIME									
	8 APRIL 1974			9 APRIL 1974		10 APRIL 1974				
	T-81 (1120Z)	T-60.3 (0800Z)	T-54 (1420Z)	T-36.3 (0800Z)	T-28 (1620Z)	T-12.3 (0800Z)	T-6 (1420Z)	T-4 (1620Z)	T-2.3 (1745Z)	T-0 (2025Z)
SATELLITE VEHICLE										
Bending Moment % Limit Load	56.29	58.27	68.87	75.98	83.09	78.40	82.92	66.01	62.75	63.05
Maximum at:										
Station (in)	1,902.									
Time (sec)	78.00	49.10	57.18	57.25	78.00	58.14	58.58	51.19	26.25	27.48
Altitude (ft)	62,816	25,995	34,986	35,005	62,805	35,987	36,499	27,999	7,297	7,999
BOOSTER VEHICLE										
o SRM Side Force % Allowable	46.67	30.76	39.89	59.76	52.66	58.43	86.72	65.23	61.60	61.34
Maximums at:										
SRM No.	2	1	1	1	1	1	1	1	2	1
Plane	Pitch	Pitch	Pitch	Pitch	Yaw	Pitch	Pitch	Pitch	Pitch	Pitch
Time (sec)	13.92	30.02	29.96	29.98	68.94	29.35	29.50	29.42	29.38	29.38
o TVC Fluid Usage (Control) % Allowable	87.76	64.63	76.30	**132.71**	**125.89**	**115.66**	**109.18**	96.15	91.67	85.43
Maximums at:										
SRM No.	1	1	1	1	1	1	1	2	2	2
Time (sec)	78.00	78.00	78.00	78.00	78.00	78.00	78.00	78.00	73.00	78.00
Expended (lb)	1,861.33	1,370.86	1,618.41	2,814.88	2,670.05	2,453.06	2,315.70	2,039.38	1,800.05	1,811.91
o Alpha · Q Bar % Allowable	37.24	36.47	46.66	61.27	**101.0**	71.82	59.71	49.96	46.45	51.75
Maximums at:										
Time (sec)	43.17	49.10	51.90	36.04	36.76	40.86	38.99	41.08	43.27	42.18
Altitude (ft)	20,008	25,995	29,000	13,923	14,455	17,843	36,999	18,001	20,003	19,002
α · Q Bar (deg-psf)	1,261	1,473.	1,951.	1,912.	3,181.	2,256.	2,532.	1,563.	1,551.	1,672.

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Figure 3-5

Critical Launch Parameter Summary

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The 9 April 1974 results indicated excessive booster TVC fluid usage for control and also one instance of exceeding the maximum Alpha Q Bar constraint. The data indicated an increasing winds aloft trend, consequently the launch was delayed 24 hours.

The 10 April 1974 data initially (T-12.3 and T-6 hours) violated the booster specified TVC fluid usage; however, the succeeding data analysis indicated a decreasing winds aloft trend which became acceptable, ie, less than the 100 percent allowable, at T-4 and again at T-2.6. Consequently, a go for launch recommendation was issued.

An R-17 day preliminary loads data comparison analysis was accomplished on 8 March 1974. Martin Marietta Corporation (MMC) verified the SBAC results as being acceptable by letter (MMC 74-Y-30429) dated 11 March 1974.

3.10 SOLAR ARRAY (SA)

Deployment and erection of the left (-Y) SA is shown in Figure 3-6 and for the right (+Y) SA in Figure 3-7. The arrays were deployed at the first station pass, INDI, and were repositioned from +18° to 0° at KODI on Rev 1 for maximum output at the initial flight beta angle of -2.6°. The arrays were repositioned on Rev 713 to +10.7° on the right and +12.5° on the left when the beta angle reached +7.6°. On Rev 1134, the arrays were repositioned to +18° when the beta angle reached +15.2°.

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 3-16. 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-8 flight experience.

3.11.2 Active Thermal Control (ATC)

The Active Thermal Control reference temperature in the Mid Section remained relatively constant throughout the flight as shown in Figure 3-8.

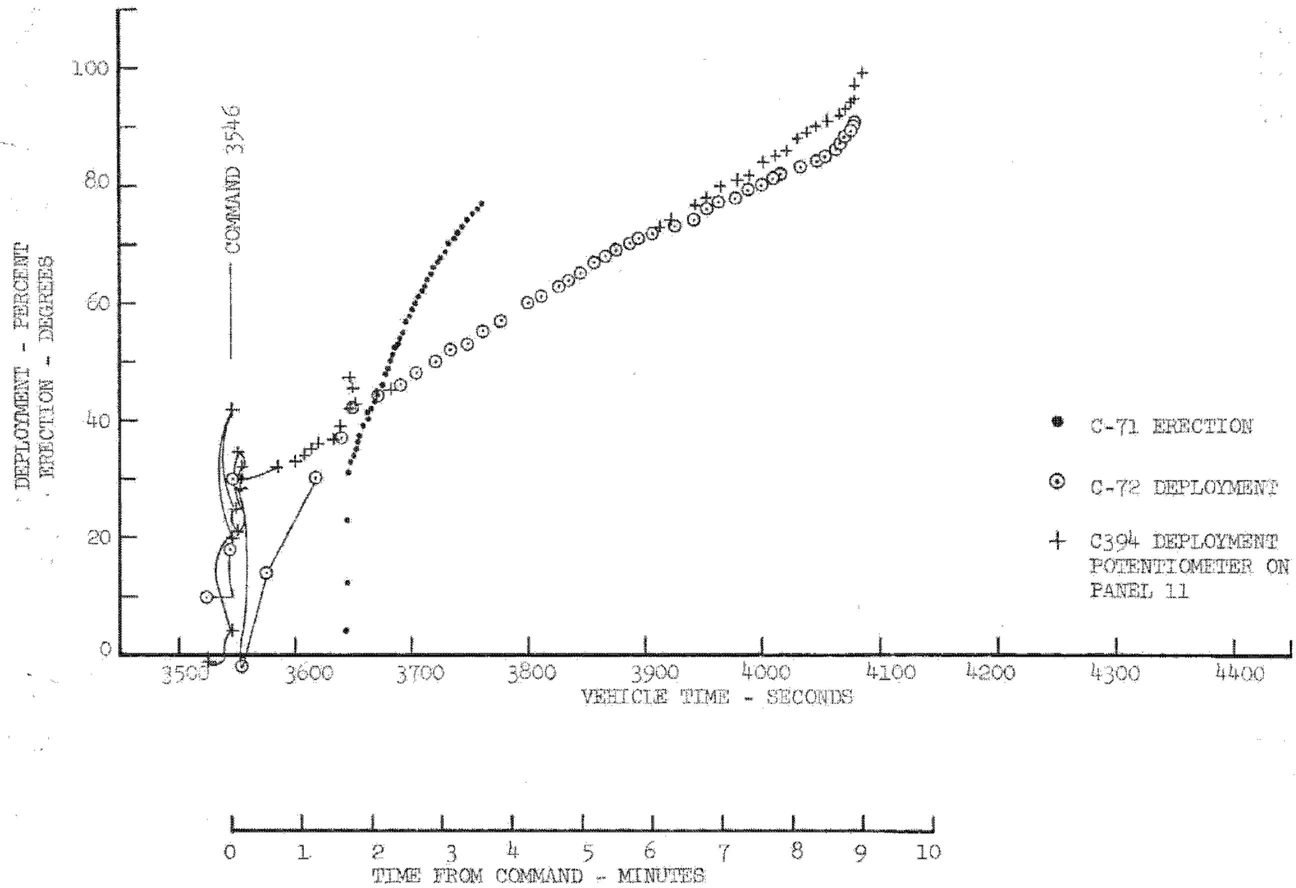


Figure 3-6

Left Solar Array Erection and Deployment Time Histories

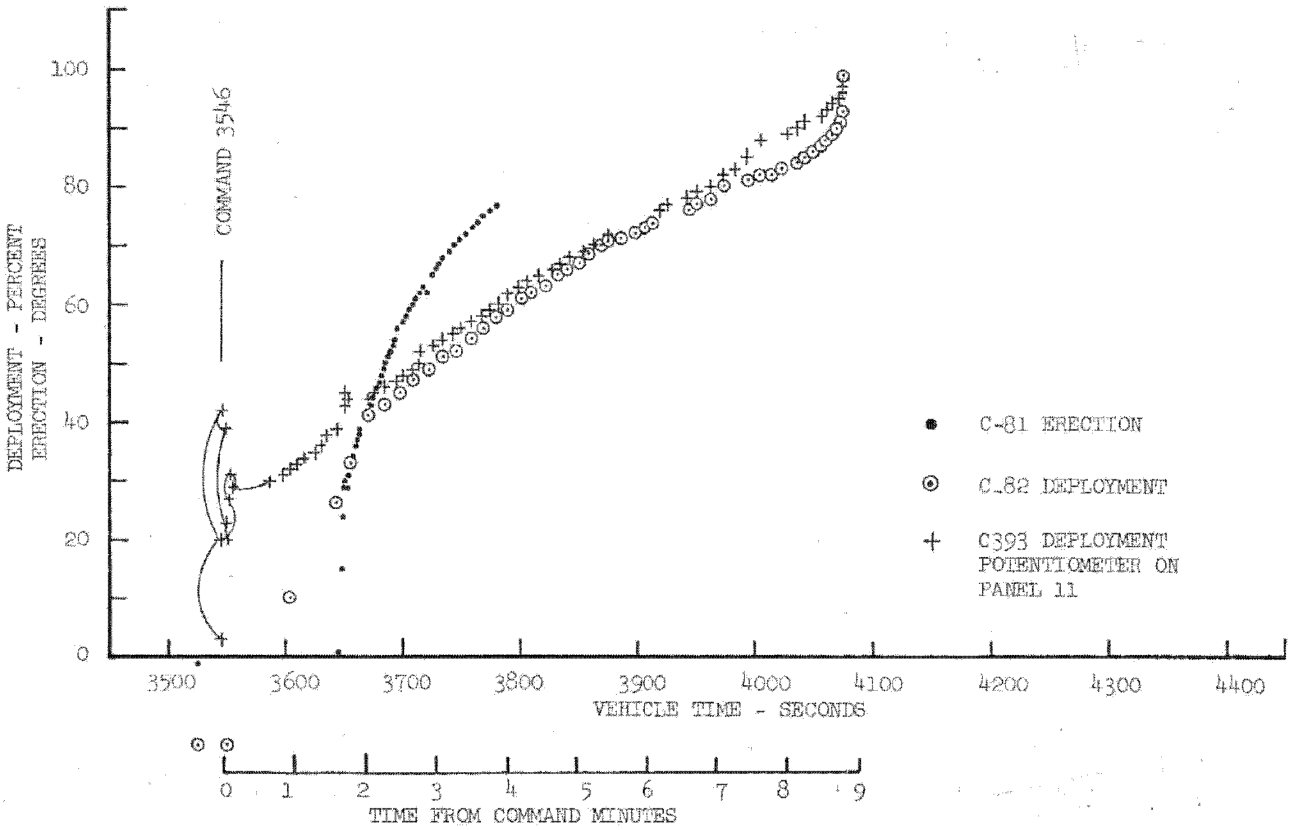


Figure 3-7

Right Solar Array Erection and Deployment Time Histories

TABLE 3-16

THERMAL DATA SUMMARY

Vehicle Section	Parameter	Design Limits	SV-8 Actual*
Mid Section	T_{TCA}	49/91	68
	T_{SU}	47/93	70
	$T_{SU} - T_{TCA}$	5/-4	2
Forward Section	T_{FWD}	47/93	68/74
	$T_{FWD} - T_{TCA}$	±20	0/6
APSA	T_{ENC}	32/69	53
	T_{TU}	30/85	53
	DBS Panel	32/90	60

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_{TU} = Orbit average Satellite Recovery Vehicle takeup temperature.

Temperatures are in °F.

* The actual temperatures shown are taken from the first week of flight after stabilized thermal conditions had been reached. During the remaining portions of the flight these temperatures generally did not change more than 2°F.

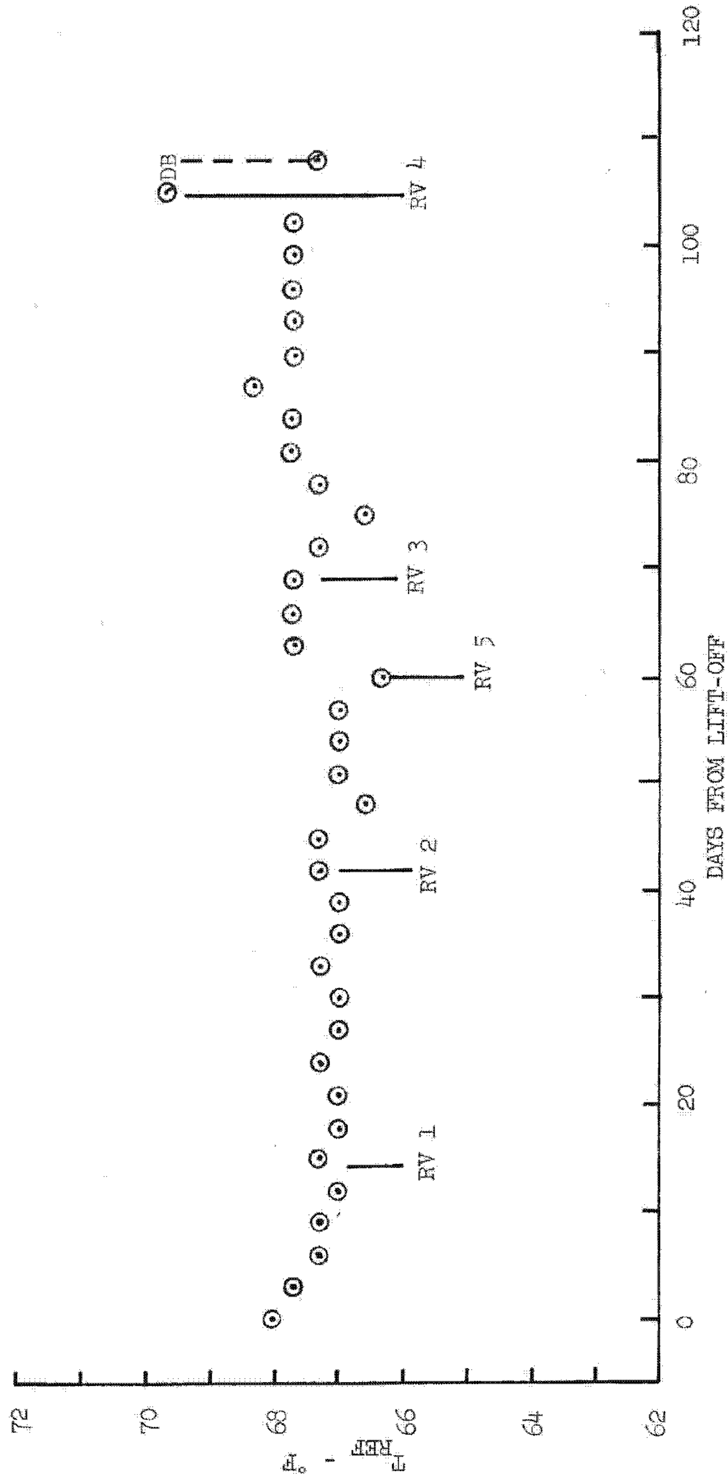


Figure 3-8
SV-8 T_{REF} History

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, due to change in distance from the sun, possibly along with a cooling trend of up to 1°F per week seen on some previous flights, appeared to counteract the effect of beta angle. 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 3-17.

The SV-8 Aft Section was configured to accommodate a beta angle range of -15° to +60° as follows:

- o Batteries were relocated to Bays 3 and 4.
- o More Flexible Optical Solar Reflector (FOSR) was used in the new paint pattern shown in Figure 3-9.
- o RCS Tanks 1, 2, 3 and 4 were all installed.

The FOSR covered doors on Bays 2, 3, 4 and 8 were instrumented and preflight predictions for uncontaminated and worst case contaminations together with the flight data are shown in Figure 3-10. Booster contamination degradation appears similar to that of other flights.

The wide beta range flown on SV-8 provided an opportunity to observe the performance of the new Aft Section thermal design over a variety of environmental conditions. The close correlation of equipment temperature levels with preflight predictions over the beta range -2.6 to +28 degrees substantiates the predicted capabilities of this design.

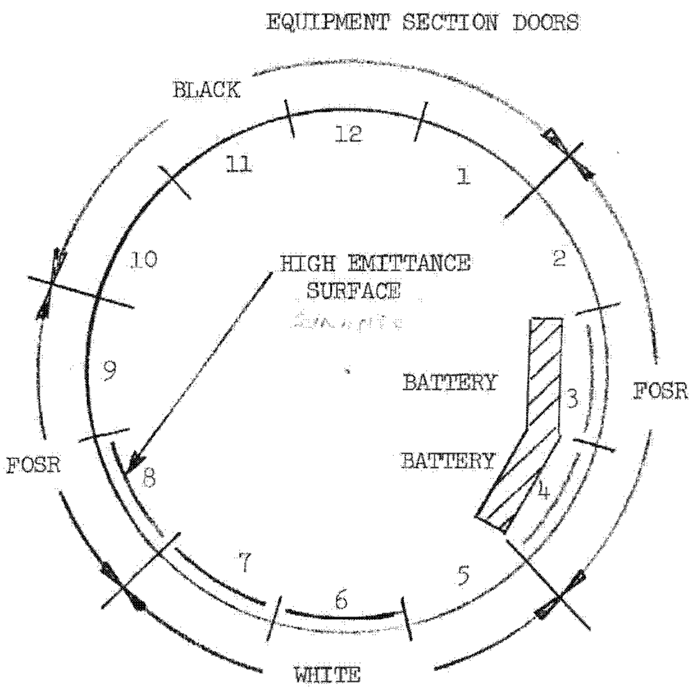
3.11.4 Quantic Experiment

Figure 3-11 shows Horizon Sensor mean temperatures versus Beta angle, while Control Electronics (located in Bay 11) temperatures are presented in Figure 3-12. The Star Tracker heads were not temperature instrumented.

TABLE 3-17

AFT SECTION CRITICAL COMPONENT TEMPERATURES (°F)

Critical Component	Design Limit	Steady State Range
Power Distribution Junction Box	-30/165	74/85
Charge Current Controller 2	-30/170	91/98
Type 29 Batteries, Bay 3	35/70	43/50
Type 30 Battery	30/90	45/47
Type 31 Batteries	40/90	45/49
Type 29 Batteries, Bay 4	35/70	43/49
Positional Drive Assembly	-30/160	55/80
Solar Arrays	-125/225	-76/145
Inertial Reference Assembly	50/135	98/114
Horizon Sensor Assembly Heads	0/130	77/85
Gyros	50/210	147/170
Orbit Adjust Tank	70/100	69/88
Quad Valve	40/200	108/144
Tape Recorders	20/120	72/92
Transmitters	-30/170	72/96
Pulse Code Modulator Master	-30/170	74/102
Programmable Memory Unit A	-40/145	79/95
Programmable Memory Unit B	-40/145	79/104
Clock	-40/153	89/108
Minimal Command System	-40/149	59/80
Reaction Control System Tanks	40/140	67/104
Plumbing Bay 12	35/140	77/95
Plumbing Bay 6	35/140	75/87



OAM-RCM: RCS skins Bays 3 and 4 white; remainder all black/aluminum.

Figure 3-9
Aft Section Paint Pattern

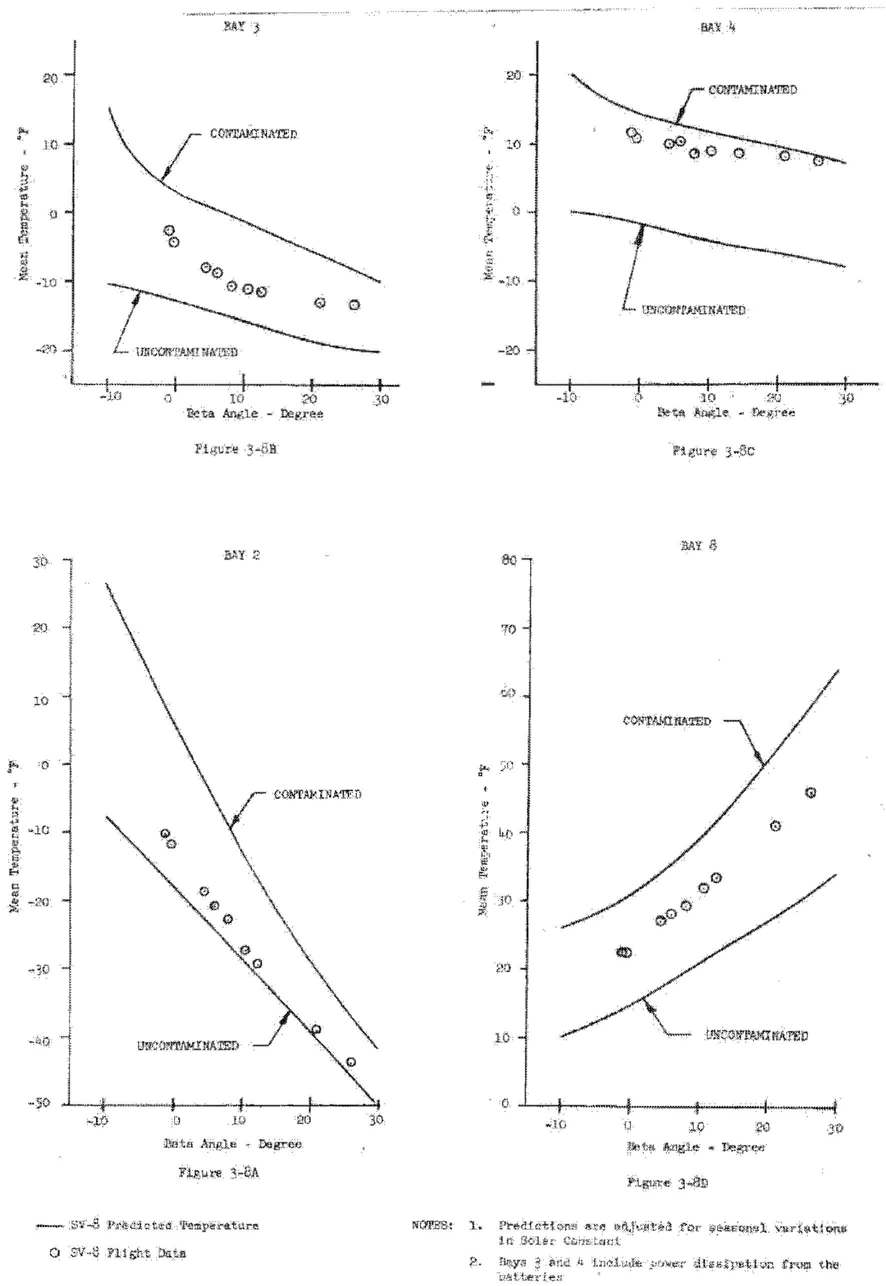


Figure 3-10

Equipment Section Skin Temperature

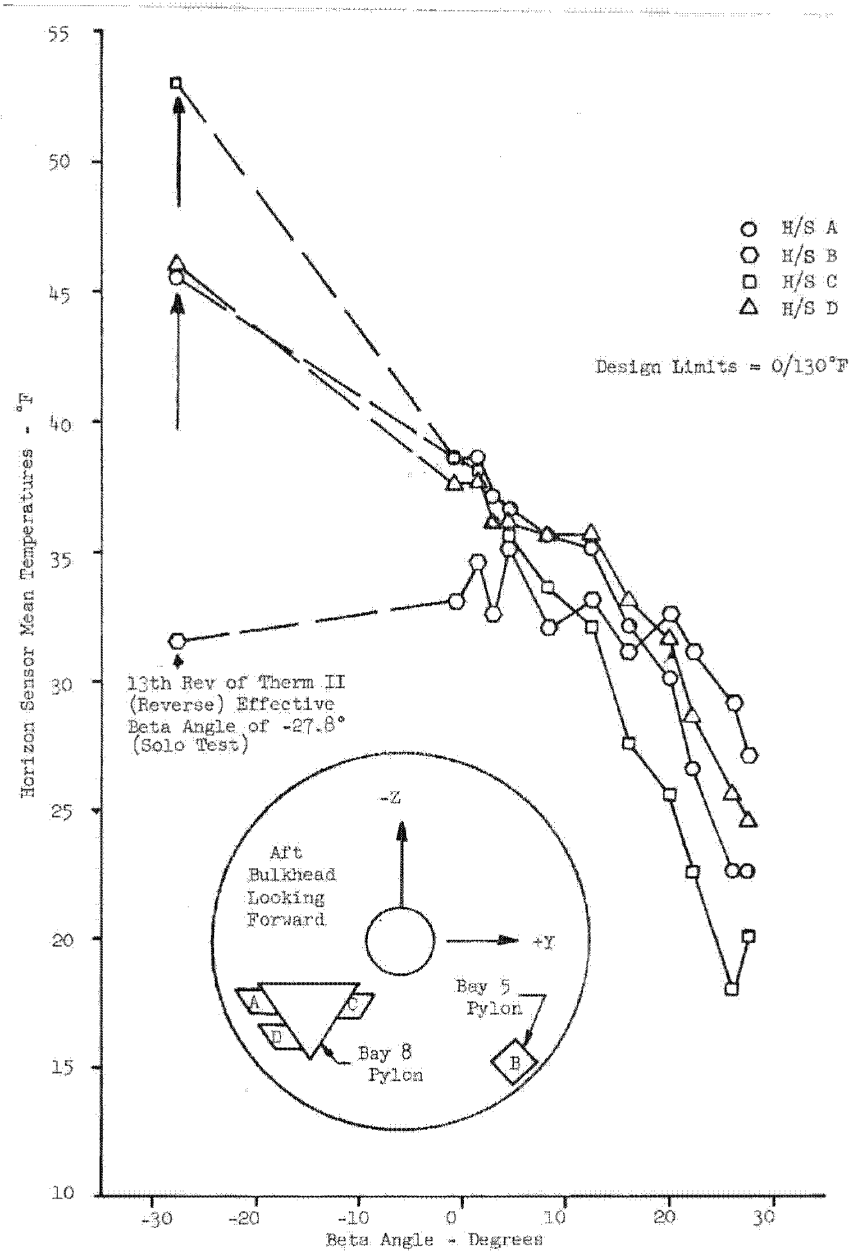


Figure 3-11

Quantic Experiment - Horizon Sensors Mean Temperatures

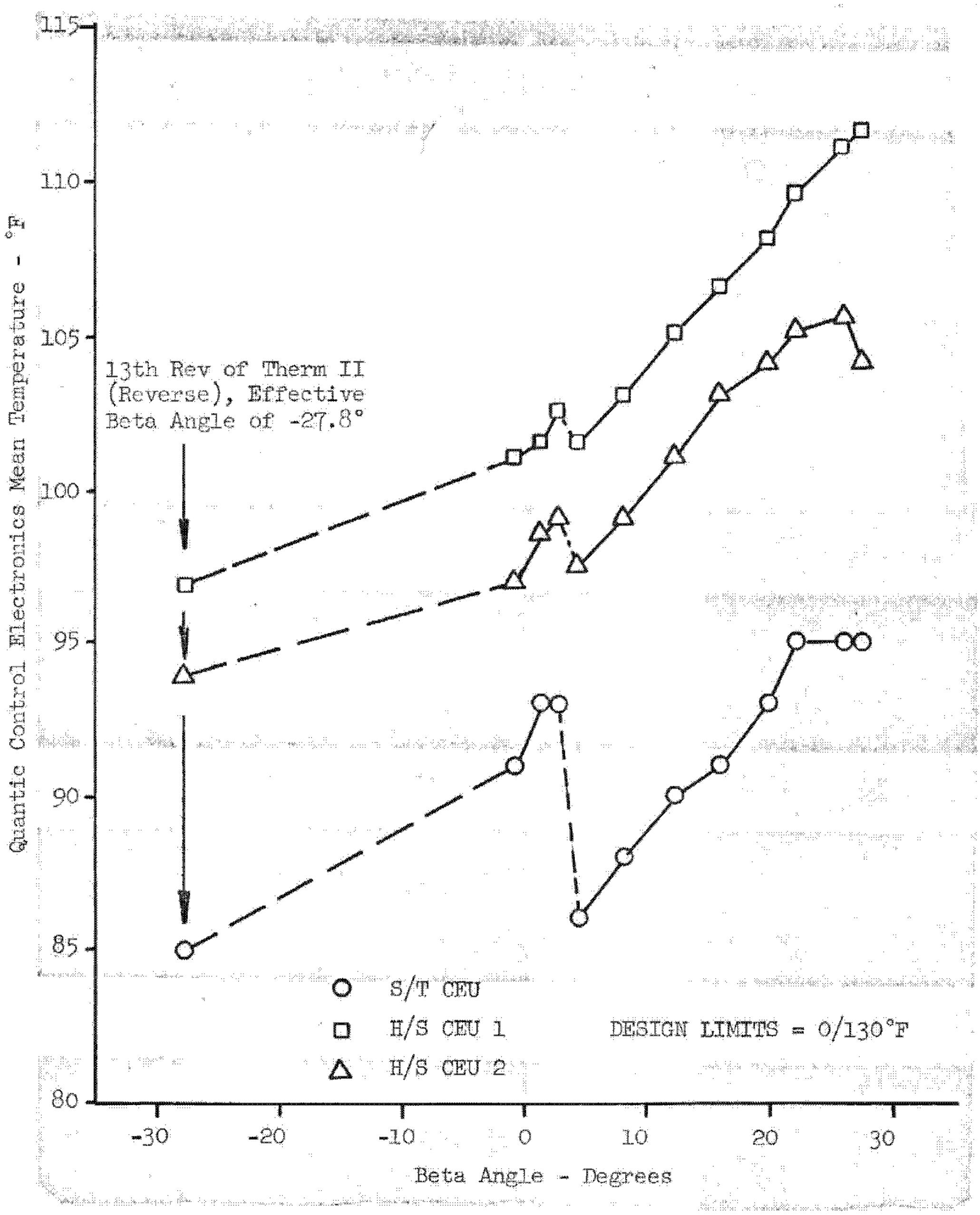


Figure 3-12

Quantic Experiment Control Electronics Mean Temperature

All temperatures remained within the design limits throughout the flight, and flight data showed the expected temperature effects of changing Beta angle.

SECTION IV
PAYLOADS

4.1 SENSOR SUBSYSTEM

4.1.1 Coarse Film Path

Both coarse film paths (supply, loopers, steerers, articulators and takeups) exhibited nominal operation throughout the mission except for a forward-looking camera takeup malfunction in Segment 3, and an aft-looking camera takeup malfunction in Segment 4. In the former case, the cause of failure was determined to be a missing builder-roller down-verification signal, normal operation was resumed by disabling the forward-looking camera verification interlock (VIA-DIS Command employed). The aft-looking camera takeup malfunction was determined to be a failure of the takeup integrator servo to reset at TU brake-off; operation was resumed but with nested operations eliminated.

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. Focus, 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

A small leak was discovered on the A-Side during launch pad testing. Evaluation showed the leak to be small and to be controllable by the high pressure isolation valve (HPIV-A). It was decided to launch with the leak and to close HPIV-A for the duration of the mission immediately after optical bar stowage following SV/BV separation. Additional information on the leak was obtained during Segment 2, which showed that the leak was low pressure and very small. It was determined that this small leakage could be tolerated and HPIV-A was subsequently opened restoring the pneumatics module to its normal configuration for the duration of the mission.

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4.3 SUBSATELLITES

Two Subsatellites were carried into orbit on SV-8. A 432 pound Subsatellite System (ejected weight 378 pounds) 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 18.9°. Separation occurred at 8° North latitude on the descending node. All Subsatellite separation sequence events were within specified limits and the Subsatellite went on to achieve its desired orbit.

A 458 pound S-73-7 Subsatellite (the ARPA Transfer System [ATS]) carried on the +Y Side of the Forward Section (ejected weight was 406 pounds).

Separation of this Subsatellite occurred at Rev 3.4 which coincides with 31° North latitude on a descending node. Again the Subsatellite separation sequence was within desired tolerances and the Subsatellite achieved its intended 435 ±35 nm circular orbit. GFE payload separation from the orbiting ATS did not occur, however, and the absence of telemetry on board the ATS made a complete post flight analysis impossible.

Analysis of available tracking data leads to the conclusion that the most likely cause of the anomaly was the failure of the ARPA Payload Separation System and that the ejection rocket energy caused the ATS to spin up to the final, observed 44 rpm.

4.4 MAPPING CAMERA SUBSYSTEM

The operation and performance of the fourth ST Camera System flown on SV-8 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. Three recoveries terminated in aerial retrieval. RV-1, even though exhibiting a stable fully inflated main chute and target cone, was recovered from the water by Surface Recovery Units. It appears that a very shallow engagement of the retrieval loop and the crown area of the target cone permitted a hook tear through without engaging a load loop.

The outer wraps of film on one stack on RV-3 were torn and loose due to relative motion between the film stack and the RV after pin failure. The second pin in RV-3 was bent and significant RV and takeup damage occurred during the pin removal operations required to permit despooling. 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.

TABLE 5-1

RV RECOVERY SUMMARY

	RV 1	RV 2	RV 3	RV 4
RV Serial Number	36	35	34	33
Recovery Rev Number	228	681	1118	1701
Recovery Date	24 April 1974	22 May 1974	18 June 1974	24 July 1974
Payload Weight (lb) (Measured Weight from Recovered RV)				
Forward	221.0	230.0	199.0	212.4
Aft	221.0	221.0	191.3	226.6
Unbalance Percent	0	3.3	3.1	5.2
SV Orbit (hp X ha/ωp)*	85.52 x 160.97/127.63	85.47 x 158.07/124.8	85.35 x 154.2/119.48	83.53 x 136.48/113.08
SV Pitch Angle (degree)	-33.5	-37.7	-37.5	-37.9
Nominal PIP Latitude (°N)	18.00	18.50	18.00	21.00
Impact Location Error (EPPD Versus Teapot Evaluation)				
Overshoot (nm)	1.2	9.0		
Undershoot (nm)			0.5	0.4
Cross Track (nm)	1.2 E	0.6 W	2.8 E	0.5 E
Recovery (Aerial)				
Altitude (ft)	Water Recovery	7,500	12,800	5,300
Parachute Condition	4 Cone & 1 Main Tears	2 Cone & 1 Main Tear	No Damage	No Damage
Retrieval Pass	Not Applicable	2	1	2
RC/Payload Condition	Good	Good	Good	Good

*hp = Altitude of Perigee (nm), ha = Altitude of Apogee (nm), ωp = Argument of Perigee (deg)

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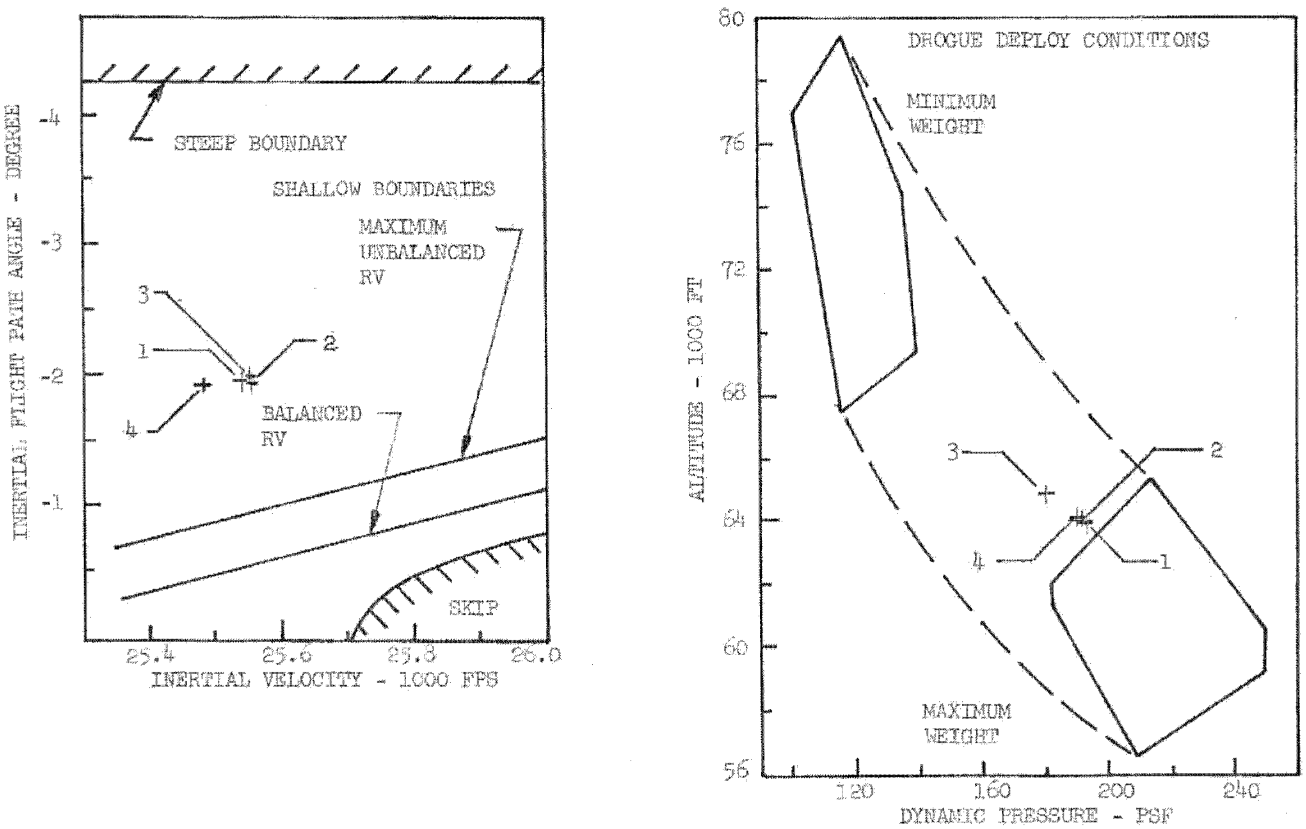

Figure 5-1

SV-8 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} $^{+0°F}_{-5.4°F}$ o Power Usage (Watts/RV) Maximum 18.9 (First day in orbit) Stabilized: 6.3 (Ninth day in orbit) Allowable 20
Trim and Seal	Normal
Electrical Power and Distribution	Normal o All batteries activated. o All voltages at least 24.9 volt open circuit voltage. o RV 4 Main Battery vented small amount of electrolyte overboard. o One of three redundant wires loose on Burndy Terminal Board on RV 2.
Structure	Normal
Pyro Subsystem	Normal
Spin Stabilization	Normal
Retro Motor	Normal
Tracking, Telemetry and Instrumentation	Normal. Low level PCM commutator channels read 4 to 5 counts high on RV 1 and RV 4.
Heat Shield	Normal
Base Thermal Protection	Normal
Sequential	Normal
Recovery	Normal

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5.3 REENTRY VEHICLE SUBSYSTEM PERFORMANCE

Three secondary discrepancies were observed during the post flight examination of the recovered vehicles.

A. Loose Wire

On RV-2 one of the three redundant wires was loose in a Burndy Terminal Block. No anomalous pre-flight test or flight behavior could be attributed to the loose wire. Additional emphasis will be placed on manufacturing and inspection personnel to ensure that all processes and procedures are followed.

B. Batteries

On RV 4, the Main Battery vented a quantity of electrolyte through the overboard vent. This condition may have been aggravated by reduced cell block pressure, with the long time orbit vacuum exposure. Previously initiated corrective action to make more stringent leak test verification of the cell block was not effected on this battery. Main batteries which incorporate this leak test are installed in RVs in Position 4 of SV-9 and SV-10 and all RVs in SV-11 and up.

C. PCM Commutator

Low level channels on RV 1 and RV 4 read 4 to 5 counts high. No data was lost since the flight data measurements can be compensated by taking into account the amount of offset.

5.4 STELLAR TERRAIN RECOVERY (RV-5)

RV 5 (S/N 1804) was successfully recovered on Rev 973. 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 2.53 nm short and 3.65 nm West of the ground track. The capsule was recovered at 13,500 feet on the first pass.

TABLE 5-3

ST-RV (RV-5) RECOVERY SUMMARY

Recovery Rev	973
Date	9 June 1974
Payload Weight (100%)	69.82 lb
SV Recovery Orbit	
Perigee (nm)/Apogee (nm)/Argument Perigee (deg)	85.20/152.58/136.44
SV Pitch Angle (after yaw around) (deg)	-63.3

	PIP	EPPD	Air Catch
Latitude	15° 00'	15° 03'	15° 14'
Longitude	166° 51.6'	166° 55'	166° 41'
Altitude			13,500 ft

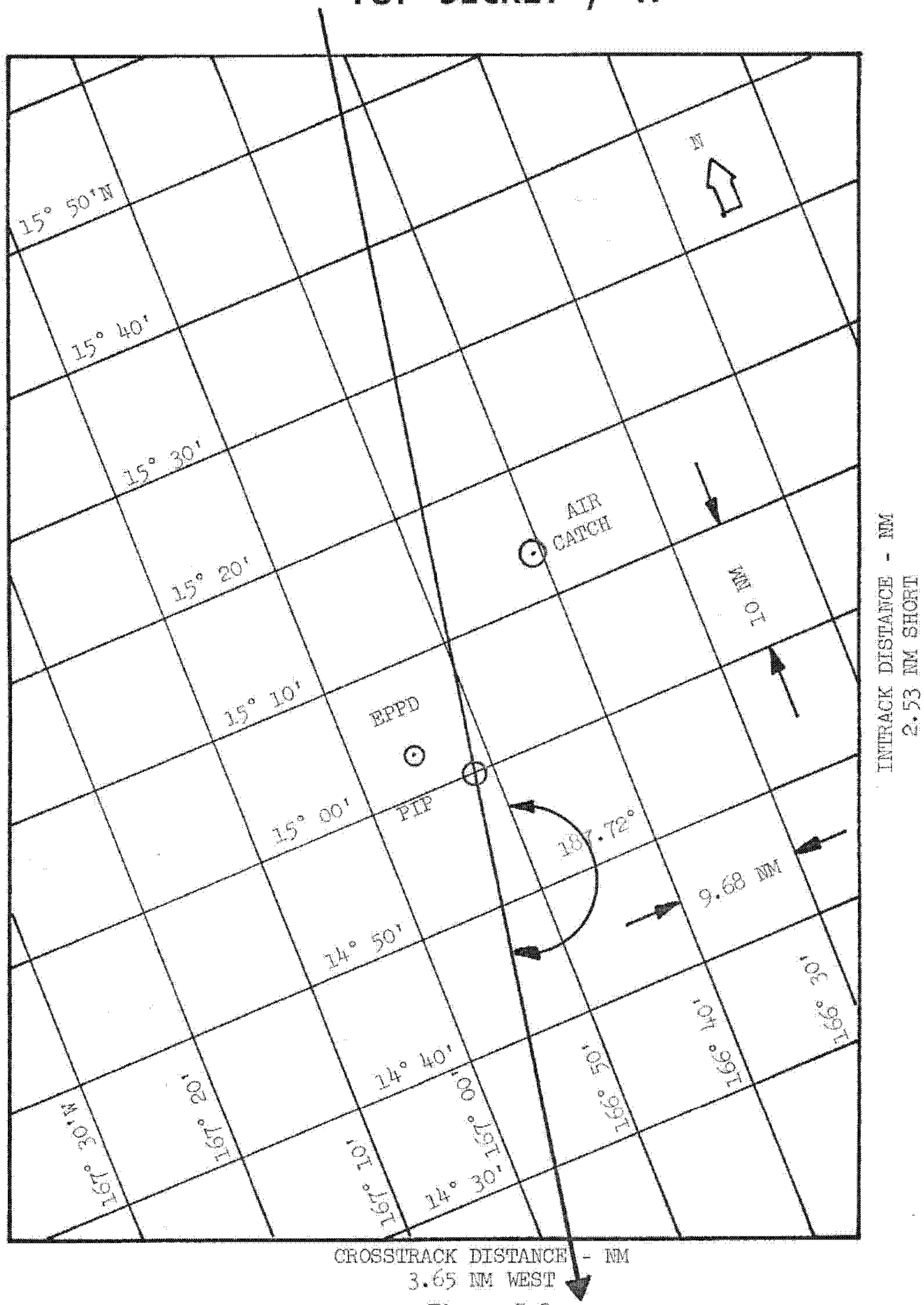

Figure 5-2

ST-RV (RV-5) Recovery Locations

APPENDIX A

GLOSSARY OF TERMS

ACS	Attitude Control System
ARPA	Advanced Research Projects Agency
ATC	Active Thermal Control
ATS	ARPA Transfer System
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
ESD	Emergency Shutdown
FEB	Functional Electronic Blocks
FET	Field Effect Transistor
FOSR	Flexible Optical Solar Reflector
GFE	Government Furnished Equipment
HPIV	High Pressure Isolation Valve
H/S	Horizon Sensor
HSA	Horizon Sensor Assembly
IRA	Inertial Reference Assembly
MCM	Mapping Camera Module
MCS	Minimal Command System
MMC	Martin Marietta Corporation
nm	nautical mile
OA	Orbit Adjust
OAM	Orbit Adjust Module
OAS	Orbit Adjust System
OOAA	On-Orbit Adjust Assembly

APPENDIX A (Continued)

PACS	Primary Attitude Control System
PCM	Pulse Code Modulator
PDWN	Pitch Down
PFA	Post Flight Analysis
PIP	Predicted Impact Point
PMU	Programmable Memory Unit
PST	Pacific Standard Time
RACS	Redundant Attitude Control System
RCM	Reaction Control Module
RCS	Reaction Control System
REA	Reaction Engine Assembly
REM	Reaction Engine Module
Rev	Revolution
RTS	Remote Tracking Station
RU	Remote Unit
RV	Reentry Vehicle
SA	Solar Array
SBA	Satellite Basic Assembly
SBAC	Satellite Basic Assembly Contractor
SECO	Stage II Engine Cut-Off
Sep	Separation
SGLS	Space Ground Link System
Solo	System Engineering Test after Fourth RV Separation
SPC	Stored Program Command
SRM	Solid Rocket Motor
SS	Sensor Subsystem
[redacted]	
SSU	Subsatellite Unit
ST	Stellar Terrain

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APPENDIX A (Continued)

ST/RV	Stellar Terrain/Reentry Vehicle
SU	Supply Unit
SV	Satellite Vehicle
TCA	Two Camera Assembly
T/R	Tape Recorder
TT & C	Telemetry, Tracking and Command
TU	Take-Up
TVC	Thrust Vector Control
VIA-DIS	Verification Interlock Disable