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Performance Evaluation Team Report

JAN 5 1978

MISSION 1204

DIRECTORATE OF SPECIAL PROJECTS
OFFICE OF THE SECRETARY OF THE AIR FORCE

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MISSION 1204

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PUBLICATION REVIEW

This report has been reviewed and is approved.

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FOREWORD

This report was prepared for and by direction of the Director of Special Projects, Office of the Secretary of the Air Force. The report is Volume I of the final mission report for HEXAGON Mission 1204. Volume II is entitled Sensor Subsystem Post Flight Analysis Report, TCS 363502-73.

The report was prepared by the SAFSP HEXAGON Performance Evaluation Team (PET) using reports and data provided by SAFSP, the Technical Advisor (TA) Staff, Post Flight Analysis (PFA) Team, and HEXAGON Satellite Vehicle Integrating Contractor (SVIC).

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AEROSPACE

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SUMMARY

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SECTION I

SUMMARY

1.1 INTRODUCTION

The fourth HEXAGON mission was planned for 60 days in the primary phase, followed by a 15 day Solo operation. The Satellite Vehicle (SV) was placed into an initial orbit of 85 x 156 NM by the Titan III D Booster Vehicle (BV) at 1103 PDT, 10 October 1972. The nominal orbit of 89 x 150 NM was established on the first orbit adjust. Photographic operations commenced the first night and continued throughout the mission. The average ground resolved distance (GRD) was 4.4 feet. The Re-entry Vehicles (RV) were deorbited and aerially recovered on Revs 180, 424, 715, and 1105. Operational photography was terminated on Day 69 with the recovery of RV-4. Solo experiments and lifetime demonstration activities were conducted from Day 69 to Day 91 when the SV was deorbited on Rev 1463.

1.2 AEROSPACE VEHICLE (AV) SYSTEM PERFORMANCE

The Titan III D BV performed satisfactorily, injecting the SV into a nominal orbit. Ascent events were nominal [redacted] The four RVs were separated (b)(1)
from the SV with their film loads on mission Days 12, 27, 45, and 69. All RVs were successfully (b)(3)
recovered in the air. After recovery of RV-2, planning commenced to extend mission life to 70 days. Active mission life was terminated on Day 69. Following an extended Solo operation period, the SV was successfully deboosted on Rev 1463 in the 91st day on orbit. All mission objectives were met.

1.3 SENSOR SUBSYSTEM PERFORMANCE

Operational photography began on Rev 5. The Sensor Subsystem (SS) was operational with constraints of zero rewind and scan center angles of 0° and ±30° to preclude mistracking as had occurred during Mission 1203. During 1204-3 the camera rewind speed was increased to 5 inches/second. Ten thousand feet of color film were exposed through the Forward Camera during 1204-4. The camera system exhibited nominal operation throughout the mission. All film was recovered.

1.4 SATELLITE BASIC ASSEMBLY (SBA) PERFORMANCE

With the exception of the Reaction Control System (RCS) and Attitude Control System (ACS), the performance of the SV subsystems throughout the mission was nominal. All other primary equipment functioned throughout the four mission segments requiring no additional backup equipment. As expected, propellant leakage on RCS-1 gradually increased until transfer to RCS-2 was necessary on Day 26. A yaw bias developed on ACS-2 on Rev 564. A transfer to the redundant ACS was made on Rev 582.

MISSION OVERVIEW

SECTION II

MISSION OVERVIEW

2.1 PREFLIGHT PLANNING

The prime concern during preflight planning was how to manage an anticipated thruster valve leakage problem causing thruster degradation. Four basic preflight decisions were made to minimize the anticipated thruster problem:

A. Insure that the fuel loaded into the vehicle was as clean as possible.

B. Lift-off to be with the RCS tanks 1 and 2 filled with hydrazine and with the RCS tanks 3 and 4 empty and capped.

C. Minimize vehicle activity to delay onset of thruster degradation.

D. Transfer to the secondary RCS only after the primary RCS starts to degrade. At transfer, the secondary RCS system would be supplied with propellant directly from the orbit adjust (OA) tank, thus bypassing tanks 3 and 4.

2.2 CONSTRAINTS

The major constraints applied to Mission 1204 were defined to provide an acceptable thermal environment in the presence of contamination and to minimize vehicle activity to delay the onset of thruster degradation. These constraints included:

A. Solar (Beta) angle to be within +32° to +24° for the 60 day prime mission phase.

B. Orbit adjust to occur on a 4 day cycle with all OAs positive except for a positive and negative OA combination at mid-mission.

C. Perigee altitude to remain above 88 NM.

D. Vehicle maneuver activity to be minimized with all maneuvers separated by at least two revs.

E. No practice maneuvers.

Additional constraints were made on the camera system during on-orbit operation, see paragraph 4.1.

2.3 LAUNCH BASE

The Titan III D BV arrived at the SLC-4 East, VAFB launch site and began its prelaunch readiness cycle. The SV was delivered to the launch pad on 21 September 1972 and mated to the BV. The prelaunch cycle was normal and the vehicle was launched 10 October at 1103 PDT at the opening of the launch window.

2.4 ASCENT

The BV successfully injected the SV into an 85 x 156.6 NM orbit. A small ascent guidance problem resulting in a flight path angle error gave the following deviations from the planned orbit parameters:

Apogee Altitude (NM)	+6.830
Perigee Altitude (NM)	-3.998
Period (second)	+5.504
Eccentricity	.001454
Argument of Perigee	+20.123

The 4 NM error in perigee and 20° argument of perigee rotation of the orbit was due to a slight path angle error of injection caused by a combination of atmospheric noise and closure time of the pitch relay in the missile borne guidance equipment (MBGE). Additional experiments were conducted to measure the contamination environment during launch and ascent. Results were similar to 1203 and confirm the presence of contamination during the ascent phase.

2.5 ORBIT AND RECOVERY

2.5.1 1204-1 (Eleven Days Duration)

After verification of SV stability on Rev 1, the solar array deployment was initiated. The right hand solar array was slow in erecting; however, complete deployment was verified at the POGO Rev 2 contact. Normal vehicle health checks were completed by Day 1. The first operational command message was generated and loaded on Rev 5 and payload operations began the first night.

[redacted] All camera (b)(1)
operations throughout RV-1 demonstrated nominal characteristics; however, the use of a back-up film (b)(3)
transport (BUFT) off command to prevent rewinds for non-nested camera operations caused improper optical bar (OB) stows and consequently degradation in early rev photography.

Approximately 28,300 feet of film per camera were exposed and stored in RV-1. Overall quality ranged from Good to Very Poor in both cameras. The Poor quality data occurred early in the mission as a result of the improper OB stows.

Late acquisition of the space ground link system (SGLS) signal was noted on early rev remote tracking station (RTS) contacts. After analysis, the problem was isolated to 30 to 60° elevation passes and procedures to avoid the obscured area were implemented.

RV-1 was successfully re-entered and aerially recovered on Rev 180 (Day 12). The RV was loaded to 97.4% of capacity and the film load was balanced. Post recovery inspection revealed that one of eight heavy parachute load lines was broken in the aerial retrieval.

2.5.2 1204-2 (Sixteen Days Duration)

As a result of the 1204-1 PFA evaluation, focus adjustments were made to provide optimum focus position through the RV-2 segment. Mission photography resumed after RV-1 recovery and continued nominally throughout this segment. Overall photographic quality showed improvement over 1204-1. The modified timing of the BUFT off command prevented improper OB stows and the resulting poor quality experienced earlier in 1204-1.

Thruster leakage was detected on Day 12 and became aggravated by OAs 4 thru 7. After OA 5 on Day 18, leakage increased very rapidly requiring propellant transfers on Rev 359 and Rev 398. RCS use rate had increased to approximately 6 lbs/rev on Day 26, with thruster performance and control capability becoming marginal.

An orderly transfer from the primary ACS and RCS systems was accomplished to the secondary ACS and RCS systems on Rev 413. In this configuration, the secondary RCS operates directly from the orbit adjust system (OAS) tank with the redundant attitude control system (RACS) controlling.

On Rev 423 (Day 27), RV-2 was successfully re-entered and aerially recovered. The RV was loaded to 99.7% of capacity and was balanced. Post recovery inspection of the parachute revealed that three of the eight 4,000 pound heavy load lines were broken. In addition, the core of pyro battery number 2 was swollen and electrolyte leakage spotted a small portion of the payload.

2.5.3 1204-3 (Eighteen Days Duration)

Normal mission photography resumed after RV-2 recovery and continued until Rev 474 when the Forward Camera experienced an emergency shutdown (ESD). Analysis of SS data following the ESD showed film tension had returned to normal but until an engineering test could be run on the Aft Camera, operations continued in a monoscopic mode with the Forward Camera. The SS engineering test proved successful and stereo operations resumed on Rev 508. At this time, the zero rewind constraint on non-nested operations was lifted to improve film tracking. Excessive yaw and roll rates were detected on Rev 564 and SS operations were once again halted. Mission photography resumed on Rev 588 and continued uninterrupted until RV-3 recovery. Approximately 27,006' of Forward Camera film and approximately 27,665' of Aft Camera film were loaded in RV-3. Overall photographic quality showed an average performance level that was slightly less than RV-2.

Following the excessive SV yaw and roll rates detected on Rev 564, the redundant ACS (RACS) was commanded into the course attitude mode of control to protect against vehicle tumble. In this mode, the bias disappeared; and on Rev 572 the vehicle was returned to the fine mode. The rate bias again appeared and several experiments were conducted to isolate the malfunction. These experiments verified a -.02 degree/second yaw bias drift of the RACS inertial reference assembly (IRA) and a transfer to the primary ACS for control was accomplished successfully on Rev 582. The performance of RACS remained nominal throughout this mission segment with no signs of degradation. RV-3 was successfully re-entered and nominally recovered on Rev 715 with the RV loaded to 93.74% of capacity and 2.17% unbalanced.

2.5.4 1204-4 (Twenty-Four Days Duration)

Ground rules were established for segment 4 of the flight to execute both mono and stereo operations, such that all film would be used by the termination of the mission. Photographic operations continued normally until a Rev 778 manual operation (MOP) where film did not transport. A series of engineering tests established the failure in the verification interlock (VI) circuitry. Operations continued normally with VI-A disabled and using subsystem command and control (SCC) II. The Forward Camera was transferred to color film on Rev 889 and a temporary zero degree scan center constraint was implemented at this time to allow evaluation of color film tracking performance. On Rev 960, the scan center constraint was removed and the basic preflight constraint of ±15° and ±30° centers prevailed throughout the remainder of the RV-4 segment. Normal mission operations continued until an ACS yaw bias error was detected on Rev 1088. This resulted in a yaw and roll attitude error. Payload velocity settings were modified to compensate for the attitude error. The SV attitude error disappeared just prior to the last operation. At the termination of photographic operations (signaled by an ESD indicating end of film) the Forward Camera film supply was used completely and the Aft Camera left 40 feet of film on the Supply unused. Overall quality of both cameras with black and white photography was slightly poorer than that of RV-3; the Forward Camera had slightly better quality than the Aft. Quality of the color photography was similar to the black and white with some samples of excellent quality. Quality throughout this phase was degraded by poor weather, cloud cover, and low sun angles. Approximately 23,250' of film was loaded in RV-4 from the Forward Camera and approximately 25,640' from the Aft.

Performance analysis of the RCS system was conducted during this segment with special pitch and yaw tests and OB rotations after each OA. The thruster performance data from the first payload operation of each day was also preserved for analysis. Thruster performance during this period showed some degradation, with the thrust level of Numbers 2, 4, and 7 decaying to approximately 50% of nominal, but the thrusters stabilized at this level and did not degrade significantly more during the remainder of the operations. An ACS yaw error was detected on Rev 1088 resulting in a yaw attitude error of -3.5° and a roll error of -1.0°. This condition disappeared on Rev 1102.

A normal RV re-entry and aerial recovery was accomplished on Rev 1105. The RV payload was 89.78% of capacity with 3.91% imbalance. Post recovery examination of the capsule showed extensive damage to the last two operations from Potassium Hydroxide (KOH) electrolyte from Pyro Battery No. 2 and a number of epoxy debris fragments from the RV panel were found embedded in the film stack.

2.6 SOLO TESTING

The Solo phase extended from Rev 1106 to deorbit on Rev 1463. Solo objectives were:

A. Demonstrate 75 day life of the satellite basic assembly (SBA) and subsystems. This was later extended to 90 days.

B. Perform a list of Solo experiments.

The Solo experiment chronology is shown in Table 2-1. Solo test results are not available to date; however, equipment failures experienced during Solo are reported in appropriate sections of this report. A report covering Solo results will be published by LMSC approximately 60 days after vehicle deboost.

TABLE 2-1

SOLO TEST CHRONOLOGY

Load Rev/Sta	Test Description
1055	Lifeboat II Tank Heater Characteristics
1096-1097	Redundant Subsystem Performance Verification
1110-1112	Power Depletion Simulation for SV-5
1111-1124	Optical Bar Mis-stow Thermal Gradient Tests
1118-1121	RCS-1 Performance Evaluation
1121-1123	ACS-2 Temperature Diagnostic
1125	ACS-2 High Rate Diagnostic
1126	ACS-2 Variable Rate Diagnostic
1127	Optical Bar Start/Stop Effect in Course Deadbands
1128	RCS-1 Performance Evaluation
1128-1137	Battery Efficiency Test
1129/KODI	PCM 1 Master Multiplexor Failure
1134	Switch to PCM 2
1139-1142	EDAP Deep Discharge Characteristics
1152	ACS-2 Variable Rate Diagnostic (Modified Fine Deadbands)
1158-1356	RCS-2 Performance Degradation Evaluation
1161-1188	Battery Heat Dissipation Tests
1183-1191	ACS-2 Modified Variable Rate Diagnostic
1192/COOK	Solar Array Dynamics
1200/COOK	PCM-1 Multiplexor Diagnostic Test
1209/COOK	ECS Command Loading Test
1210/HULA	ECS/Lifeboat Execute Test
1215/BOSS	ECS-A Command Loading Test
1216/COOK	ECS Command Loading Test
1227/HULA	ECS Commanding Test
1266/KODI	MCS Commanding Test
1267-1290	EDAP Slow Discharge Test
1267-1277	Horizon Sensor Output Mapping
1332	RV-5 Simulated Separation Demonstration
1335, 1337, 1339	ACS Diagnostic

TABLE 2-1 (CONT'D)

Load Rev/Sta	Test Description
1348-1356	Command Loading Diagnostic
1365-1368	SGLS-1/SGLS-2 Half Pass Switch
1385-1392	Commanding Tests, Reduced Power Testing
1389-1441	Modified Solar Array Dynamics Testing
1413	ECS Commanding With Vehicle Pitched Down
1447	ECS Secure Word Commanding Test
1448-1452	Special Bit Pattern Command Test
1459	OA to 66 NM Perigee
1460	Switch Bus to Lifeboat Battery

2.7 COMMAND LOAD SUMMARY

The software configuration used to support this mission was 'TUNITY MOD 1A and system software was Model 13.1E. A nominal two rev load cycle was used during the initial 62 days of the mission, a one rev load cycle was used during the remainder of the mission. A total of 644 command messages were generated during the flight, of which 603 were loaded into the vehicle.

2.8 ANOMALY SUMMARY

Significant anomalies and malfunctions are listed chronologically in Table 2-2. The list includes a description of the anomaly, the mission consequences, and in some cases the changes indicated for subsequent vehicles. Detailed discussion of these anomalies can be found in this report.

TABLE 2-2

SUMMARY OF ANOMALIES

Day	Description	Impact
1	Right solar array deployment delayed.	No effect on mission. Cage mechanism revised for SV-5.
1	Delayed station acquisition.	No impact on basic mission. Modulation indexes will be adjusted on SV-5 to increase downlink real time performance.
1	Improper OB stowage.	Poor photography. Corrected by Day 6 by modifying software.
11	PACS pitch rate noise.	No effect on mission. Disappeared on Day 22. Attributed to components used in IRA pitch channel. Component upgrading in process.
12	RV-1 one of eight heavy lead lines broken.	No effect on mission. Slack members abrade loaded lines. Adequate strength remains.

TABLE 2-2 (CONT'D)

Day	Descriptions	Impact
12	RCS-1 REM leak.	No effect on mission. Primary REM valves started to leak as expected. Switched to RCS-2 on Day 26 and completed flight using OAS propellant.
27	RV-2 retro truss separation delayed.	No impact on RV performance. Redesign from shear pins to controlled separation plane on all future RVs.
27	Pyro battery 2 failure.	Electrolyte spotted outer wraps of stack. Redesigned vents on all future RV batteries.
27	Three of eight heavy lead lines broken.	Same as RV-1.
30	ESD on Forward Camera. Looper carriage contacted travel limit switch.	P/L operations resumed on Day 32. Drag of roller or capstan near input of fine film drive.
35	RACS yaw bias offset.	Transfer RACS to PACS on Day 36. P/L results on Rev 567 were poor. No P/L operations Rev 570 to Rev 582. Attributed to short in torquer circuit inside gyro. Testing augmented. Design changes under study.
48	Forward Camera interlock source signal failure.	Verification circuitry disabled to complete mission.
58	PACS FCEA failure.	No impact on basic mission. Not discovered until vehicle tumbled after RV-4. Attributed to inoperative hybrid integrated buffer switch.
68	PACS yaw bias offset.	Offset disappeared on Rev 1102. P/L operations continued with SS Vy and OOAA adjustments. Similar to RACS yaw bias offset.
68	ESD on Forward Camera film depletion.	Normal with supply depletion as confirmed by flight data. ESD disabled and end of film stowed.
69	RV-4 relay panel epoxy cover failure.	Pieces retrieved in both RV-4 and RV-3, some interleaved in film stacks. Under investigation. Interim fix by taping epoxy on relay panels.

SATELLITE BASIC ASSEMBLY

SUBSYSTEMS

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SECTION III

SATELLITE BASIC ASSEMBLY SUBSYSTEMS

3.1 ATTITUDE CONTROL SYSTEM (ACS)

The SV-4 ACS experienced four anomalies during the mission. These four anomalies are discussed in paragraphs 3.1.5 thru 3.1.8. Because of previous gyro restart failures, the SV-4 vehicle thermal design was modified (addition of heat straps to the IRA baseplates and a change in external paint pattern) to allow simultaneous operation of both IRA assemblies (PACS and RACS). Both systems were operated until Rev 582, when an orderly transfer was made from RACS to PACS control because of a yaw bias problem on the redundant system. At that time the RACS gyros were shut down. Subsequent restarts of both systems during the primary mission and during Solo resulted in no restart problems. Simultaneous ACS operation is also planned for SV-5.

3.1.1 Booster Vehicle/Satellite Vehicle (BV/SV) Separation

BV/SV separation was completed at approximately 545.5 seconds vehicle time. (Vehicle time starts 67 seconds before lift-off.) Master clear off (MCLR), which enables the pitch, roll, and yaw integrators to accumulate angle, was at 513.4 seconds and Stage II engine shutoff (SECO), which terminates BV attitude control, occurred at 533.6 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. Also, the times in which the SV attitudes and rates came back within the specified limits following BV/SV separation (capture) are shown in Table 3-1.

[redacted] (b)(1)
(b)(3)

TABLE 3-1

BOOSTER VEHICLE/SATELLITE VEHICLE (BV/SV) SEPARATION

	RATE AND ATTITUDE AT BV/SV SEPARATION						CAPTURE			
	RATE (degrees/second)		ATTITUDE (degrees) HS @ SEPARATION		ATTITUDE (degrees) Δ(SECO - SEPARATION)		ATTITUDE		RATE	
	Specified (deg/sec)	Actual (seconds)	Specified (degrees)	Actual (seconds)	Specified (degrees)	Actual HS/Integrator	Specified[1] (degrees)	Actual[2] (seconds)	Specified[3] (deg/sec)	Actual[4] (seconds)
Pitch	±0.752	-.159	-21.7 to +13.0	1.3	±3.5	-.50/ -.74[5]	±0.70	661.3	±0.014	99
Roll	±0.786	-.241	±10.6	1.0	±3.0	.52/ .60	±0.70	661.3 +520	±0.021	661.3
Yaw	±0.752	.175	-11.4 to +11.1	-	+4.5 to -3.5	-/1.82	±0.64	661.3 +520	±0.014	661.3

NOTES:
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.

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The maximum SV rates observed following the yaw separation impulse are listed in Table 3-3.

TABLE 3-3

MAXIMUM SV RATES AFTER SEPARATION

SV Rates	Rate (degrees/sec)
Pitch Gyro[1]	-.072
Roll Gyro	-.080
Yaw Gyro	-.044

With the Subsatellite located on the vehicle +Y side and along the -Z axis at Z = -11.47 inches, a negative roll rate such as shown above was expected.

3.1.3 Payload Operations

Satellite Vehicle rate and attitude requirements during payload operations were met with the exception of those listed in Table 3-4. The yaw gyro rate bias (discussed in paragraph 3.1.5) resulted in roll and yaw attitudes outside the pointing requirements of .70° and .64° respectively. The yaw gyro rates shown in Table 3-4 are the bias rates of the anomalous controlling IRA and are not the actual vehicle rates. Rate performance was met at all times.

TABLE 3-4

PAYLOAD OPERATIONS DURING YAW GYRO BIAS PERIODS

Rev	Film Transport +Vehicle Time (seconds)	Film Transport -Vehicle Time (seconds)	Yaw Gyro Rate (degrees/sec)	Approximate Yaw Attitude (degrees)	Roll Attitude (degrees)
567.4	504104.0	504127.2	-.008	2.2	.50
1093.4	789090.6	789147.0	.012	-3.3	-.85
1094.4	794467.8	794492.8	.011	-3.0	-.80
1095.4	799733.8	799770.4	.012	-3.3	-.85
1096.3	804976.2	805078.6	.012	-3.3	-.85
1097.4	810550.2	810566.8	.013	-3.6	-.90
1100.6	827613.2	827635.2	.013	-3.6	-.90
1101.4	831868.8	831941.0	.013	-3.6	-.90
1101.4	832055.4	832083.2	.013	-3.6	-.90
1102.4	837271.8	837288.4	0	0	0

[1] Geocentric program rate of -.0687 degrees/second was included.

Since no yaw attitude sensor is available, the yaw attitude is estimated from the following relationship:

$$\psi_\epsilon = -\frac{H_\phi}{\omega_o H_\psi}\,\omega_{z_g}$$

where:

ψ_ϵ = Yaw attitude

H_ϕ = Roll H/S to Roll gain = .0055

ω_o = Orbital rate = .0012 rad/sec

H_ψ = Roll H/S to yaw gain = .0167

ω_{z_g} = Yaw Gyro Rate

3.1.4 Recovery

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

Following separation of RV-4, vehicle control was lost. Details of the failure are discussed in paragraph 3.1.6. Subsequent tumbling capture details are covered in paragraph 3.1.7. Separation of RV-4 from the satellite vehicle was completed before loss of control.

3.1.5 IRA Bias Anomalies

The yaw rate bias offset was seen on IRA 1012 (PACS) starting at Rev 1088 and on IRA 1009 (RACS) starting at Rev 564. The probable cause was a high impedance short in the torquer circuit. Because the bias rate was determined during Solo to be rate and acceleration sensitive, the most probable short location is within the gyro. Three possible origins for this short are:

A. Distortion of the gyro torquer flex leads by overheating due to excessive current.

B. Distortion of flex leads by gas bubbles inside the flex lead cavity.

C. Poor connection of a gyro gimbal snout lead to the snout.

In a ground test, a short having resistance values within the range required to cause the flight offset was induced by injecting a bubble into the gyro flex lead cavity to force the flex leads against the gyro header. Changes in the bubble can be associated with motion and changes in temperature. The IRA 1009 anomaly occurred first and did not change with subsequent temperature changes; the amount of offset can be seen to vary during vehicle maneuvers. The IRA 1012 operated at a constant temperature until the RACS was turned off and then for a long period at a lower temperature before its anomaly appeared. On IRA 1012 the offset occurs after vehicle maneuvers and then disappears after a period of time (in the first

TABLE 3-5

PITCH DOWN PERFORMANCE PRECEDING RECOVERY VEHICLE SEPARATION

	Pitch Down Angle		Maneuvering Time to ≤ 0.1 Deg/Sec		Pitch Down Coast Rate		
RV/Rev	Desired ±3.0 Deg (degrees)	Actual (PDWN) (degrees)	Spec (seconds)	Actual (seconds)	Command Rate (deg/sec)	Coast Rate Expected (deg/sec)	Coast Rate Actual (deg/sec)
1/180.3	-37.9	-37.5	150	68	-.705	-.75± .05	-.70
2/423.3	-40.9	-41.2	150	85	-.705	-.75± .05	-.70
3/715.3	-41.7	-40.8	150	70	-.705	-.75± .05	-.73
4/1105.3	-41.5	-39.8	150	70	-.705	-.75± .05	-.74

TABLE 3-6

SUMMARY OF RE-ENTRY VEHICLE/SATELLITE VEHICLE SEPARATION PERFORMANCE

RV/Rev	Peak Pitch Rate (deg/sec)	Max. Pitch Integrator Angle (degrees)	Induced Impulse By Rev (lbs/sec)	Pitch Down Prior to Sep (degrees)	Pitch Up Following RV Sep to Removal of Maneuver Command (degrees)	Pitch Inertia (After Sep) (slug-ft^2)	Pitch Thruster Moment Arm (feet)	Roll Spec (degrees)	Angle Meas H/S (degrees)
1/180.3	2.24	14.1	133	-37.5	99.2	99040	29.0	±1.0	.16
2/423.3	2.31	16.2	125	-41.2	99.4	76222	24.5	±1.0	.30
3/715.3	2.39	17.7	127	-40.8	99.3	61358	20.1	±1.0	.20
4/1105.3	[1]	[1]	[1]	-39.8	[1]	51730	15.4	±1.0	[1]

NOTES: [1]See discussion paragraph 3.1.6.

occurrence it lasted from 1088 through 1101). The IRA 1012 offset was made to recur in Solo Experiment ACS 200 by cooling for several revs (shutdown PACS) and then reheating.

The following tests have been implemented to detect similar problems on future IRAs:

A. Compare PACS and RACS channels using differential analyzers and recorders to detect small offset differences. This test also to be run at the launch base.

B. Tumble test to check for bubbles or any foreign matter in the gyros.

The following are being investigated as possible improvements:

A. Seal gyro bellows to offset altitude changes.

B. Insulate the flex lead cavity and the gimbal snout by anodizing.

C. Revise fill procedures to improve bubble and foreign matter exclusion.

3.1.6 PACS Flight Control Electronics Assembly (FCEA) Failure

Following separation of RV-4, control of vehicle attitude was lost. The pitch rate following separation exceeded 5.6 degrees/second before the failure detector responded and closed Isolation Valve No. 4. Analysis showed that the FCEA pitch analog rate channel was not responding to sensed positive rates. Thus when a pitch up maneuver was commanded, the thrusters remained full on during the period of the maneuver (approximately one minute on post recovery pitch up). Review of the data from earlier pitch maneuvers indicated that the failure occurred between Rev 940 and Rev 973. There was no loss of control during these earlier pitch maneuvers since they were negative (pitch down) with an inertial return to horizontal. No positive (pitch up) maneuver had been executed since RV-3 (Rev 715).

The cause of this anomaly has been diagnosed as an inoperative hybrid integrated circuit buffer switch on the rate processor subassembly in FCEA 1015. Corrective action has previously been implemented on the buffer switch to improve design reliability. The failure rate on this switch lot is not abnormal; therefore, no other corrective action is contemplated.

3.1.7 ACS Performance During Tumbling Capture (M1C2)

The pitch rate channel failure (FCEA) discussed in paragraph 3.1.6 resulted in the SV tumbling following RV-4 separation (Rev 1105.3). The tumble and subsequent capture occurred on the same system (M1V2). The significant tumble and capture times are given in Table 3-7.

TABLE 3-7

TUMBLE AND CAPTURE TIMES

Event	Vehicle Time (seconds)
RV 4 Sep	13947.0
FDU Enable	14037.0
FDU Closed IV 4	14074.6
ACS 2	14677.6
RCS IV 4 Open	14702.8
ACS 2 Exc	14707.8
H/S Search	14827.8
MCLR	19676.8

The failure detector unit (FDU) closed the No. 4 isolation valve (IV) 37.6 seconds after the FDU enable. Although the pitch rate was in excess of 5.56 degrees/second prior to FDU enable, the SV rates at the time of IV 4 closing were (1) pitch - 2.02 degrees/second; (2) roll - .03 degrees/second; and (3) yaw - 0 degrees/second.

Following separation, the SV crossed the zero degree pitch attitude (nose forward) at 13962 seconds vehicle time, and again (after one complete revolution) at 14188.6 seconds vehicle time. This was observed using pitch HS data.

Following initialization of the capture sequence (14707.8 seconds vehicle time) the SV captured and was stable in the nose forward attitude on Rev 1106.2P at 18617 seconds vehicle time, which was the first following real time pass.

Bipolar noise spikes as great as .03 degrees/second peak-to-peak were observed on the PACS pitch rate output. The same type of noise spikes have occurred during module testing and noise of this magnitude would have been cause for removal of the IRA. At most, this noise may have caused some extra pulses on the RCS but was not sufficient to affect the system operation. The noise is attributed to components used in the gain change amplifier/demodulator area in the IRA pitch channel. Suspect components are being replaced.

3.2 REACTION CONTROL SYSTEM

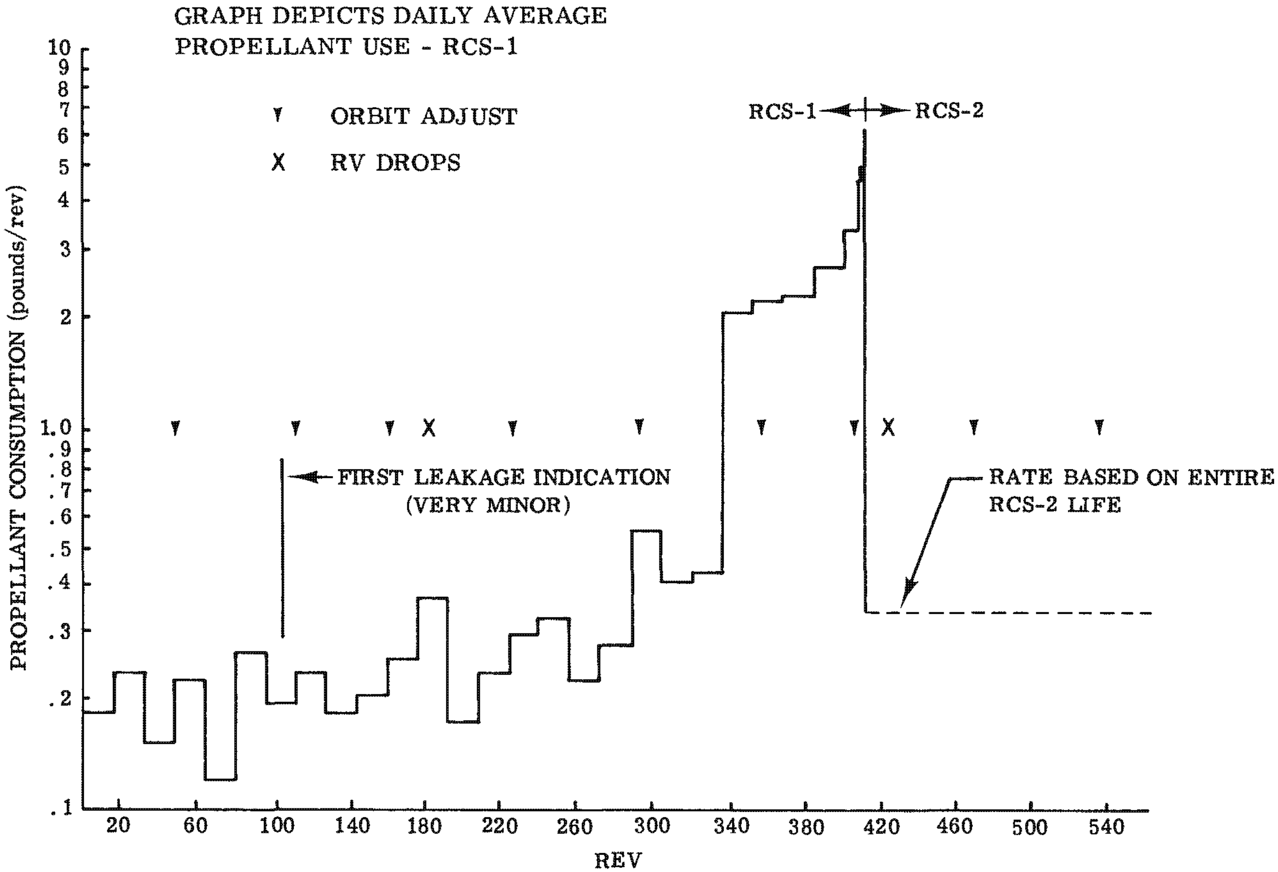
3.2.1 Flight Summary

The history of RCS propellant consumption is shown in Figure 3-1. Satisfactory vehicle attitude and rate control was provided by the RCS at all times during the 91 day flight. Leaks developed in the primary system (RCS-1) as expected and control was switched to the standby system (RCS-2) on Day 26. No leaks were detected in RCS-2 for the remainder of the flight, and although some degradation of thrust

FIGURE 3-1

REACTION CONTROL SYSTEM PROPELLANT CONSUMPTION

(Daily Average Consumption)



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was observed, the degree of degradation expected based upon ground tests did not occur.

3.2.2 Propellant Consumption

Management of propellant consumption in view of expected RCS valve leakage was very successful for SV-4. Total propellant consumption for the 91 day mission was computed to be 870 pounds. The primary (RCS-1) system was operated from RCS tanks 1 and 2 for the first 413 revs, at which time leakage exceeded tolerable limits and transfer was made to RCS-2. RCS-2 was fed directly from the main OAS tank. RCS tanks 3 and 4, which normally feed RCS-2, were cut out of the system prior to flight, thereby eliminating the source of the valve leakage, i.e., the non-volatile residue (NVR) from the tank expulsion diaphragms. As a result, no leakage occurred in the RCS-2 thruster valves for the remainder of the flight. A similar plan will be utilized on SV-5 except that tanks 1 and 2, which are associated with the primary (RCS-1) system, will be cut out instead of tanks 3 and 4. This was done to permit better insight into long term thruster performance in the presence of no leakage (RCS-1 has thrust chamber pressure and temperature instrumentation, while RCS-2 does not). In summary, the plan is to obtain the maximum life on the system wherein leaks are expected and then transfer to the remaining system at the point leakage exceeds tolerable limits.

3.2.3 Thruster Performance Degradation

No thruster performance degradation was observed on RCS-1 at the time transfer was made to RCS-2 on Rev 413. Thrust levels were as expected. Thruster ground tests conducted subsequent to qualification revealed that thrust degradation occurs for certain pulse duty cycles (pulses/day) when the total pulse count reaches approximately 60-70,000. For this reason, thrust levels of RCS-2 thrusters were closely watched as the mission progressed toward the time when the pulse count on some thrusters approached this number. Because of the lack of instrumentation on RCS-2, a series of "mini-pitch" and "mini-yaw" vehicle maneuvers were accomplished periodically to determine the steady state thrust of the pitch and yaw thrusters. Mini-pitch consisted of pitching the vehicle down 14° and then flying 3.5 minutes prior to reconnecting the horizon sensors which allowed the vehicle to reattain the proper flight attitude without additional commanding. The mini-yaw was a 14° yaw and then a commanded return to the zero yaw position. Utilizing the rates measured from these maneuvers and the known vehicle mass properties, the thrust levels were computed. The thrust degradation at 60-70,000 pulses seen in ground test did not materialize in flight. However, only one thruster significantly exceeded the ground test life in terms of pulses at the time of mission termination. The dominant parameter appears to be duty cycle as contrasted to pulse count. The duty cycle for the thruster on RCS-2 was lower than that causing degradation in ground test, and was most probably the reason for the lack of thrust degradation on orbit.

Investigations are continuing into the thruster degradation problem. On SV-5, there will be a better opportunity to evaluate the long term thruster performance. Thrust chamber pressure and temperature will be available for thrust determination since RCS-1 will feed directly from the OAS tank and will be the long term control system.

3.2.4 Conclusions

Elimination of the present RCS tankage, which have EPT10 rubber diaphragms installed, stops the gross valve leakage problem experienced on previous flights. Thrust degradation was not accelerated by operation at duty cycles (and temperatures) lower than ground test. The degradation observed in operation on the ground was not confirmed in orbit operation.

3.3 ORBIT ADJUST SYSTEM (OAS)

3.3.1 Orbit Control

The OAS was utilized eighteen times during the active mission for drag makeup, perigee location control and ground trace control. The OA firings were all normal and the engine performance was well within specifications. The OAS was utilized six times during the Solo phase of the mission for drag makeup. Three additional firings were used to place the vehicle in an orbit with a 65 NM perigee for four revs before deboost. System performance is summarized in Table 3-9.

3.3.2 Deboost

The final firing of the OA engine was for the deboost on Rev 1463. The firing duration was 486.2 seconds to achieve a planned negative velocity increment of 222 feet/second.

3.4 LIFEBOAT II SYSTEM

3.4.1 Health Checks

The Lifeboat data that was examined is summarized in Table 3-10. The magnetometer sensor data presented in Table 3-8 indicates equivalent attitude errors.

TABLE 3-8

MAGNETOMETER SENSOR DATA

(degrees)

Magnetometer	Attitude Error
Q	-.9 to .2
P	-.5 to .2
R	-.3 to 1.0

The rates measured on the three Lifeboat II rate gyros were within .10 degrees/second of the rates measured on the ACS gyros.

3.4.2 Inertial RV Recovery Experiment

The vehicle was inertially flown so as to be in an RV recovery attitude with a pitch down of -40.7° at KODI at system time 75785.6 on Rev 813. Based on the observed position of the vehicle after the ACS programmed pitch up, the actual pitch down at separation was .22° greater than desired.

TABLE 3-9

ORBITAL ADJUST SYSTEM PERFORMANCE

OA Firing No.	Rev Number	Impulse Delivered (lbs/sec)	Planned ΔV (feet/sec)	Achieved ΔV* (feet/sec)	Error ΔV (percent)
1	47	9424	15.57	15.56	-.06
2	111	17623	29.00	29.14	.48
3	160	13904	22.97	23.14	.74
4	225	21498	39.23	39.16	-.19
5	290	20422	37.42	37.42	0
6	355	17603	32.25	32.43	.55
7	406	16233	30.10	30.06	-.13
8	468	18896	38.97	39.04	.18
9	533	12280	25.35	25.50	.59
10	598	23516	48.86	49.08	.45
11	600	6499	-13.68	-13.62	-.46
12	663	17796	37.23	37.36	.35
13	728	13810	32.46	32.51	.15
14	792	14820	35.16	35.05	-.32
15	857	15096	35.86	35.90	.11
16	922	9641	22.86	23.03	.74
17	987	12462	29.76	29.91	.50
18	1052	13418	32.36	32.37	.03
19	1109	10020	24.54	24.35	-.82
20	1158	9849	27.00	27.11	.41
21	1199	14491	40.44	40.71	.69
22	1264	7904	22.50	22.40	-.44
23	1297	11047	31.42	31.42	0
24	1378	10874	31.24	31.13	-.35
25	1457	19202	55.26	55.38	.22
26	1458	17744	51.24	51.54	.58
27	1459	15218	-44.45	-44.49	.09
Deboost	1463	75549	-222.19	-	-

*Determined from the tracking ephemeris data.

[redacted]

TABLE 3-10

LIFEBOAT II OPERATION

		Q Magnetometer (milligauss)		R Magnetometer (milligauss)		P Magnetometer (milligauss)		Y-Axis Gyro (degrees/second)	
Rev	Mode	Observed	Theoretical	Observed	Theoretical	Observed	Theoretical	Observed	Theoretical
18.4	SN-DB	-23	-19.9	Saturated	Saturated	Not in use		-.05	-.069
		-23	-20.0	206	208.6	Not in use			
	SN-RV	-29	-21.2	Not in use	-	Negative Saturation	Negative Saturation		
		-26	-19.9	Not in use	-	Negative Saturation	Negative Saturation		
		-23	-19.8	Not in use	-	Negative Saturation	Negative Saturation		
	NS-RV	-23	-21.2	Not in use	-	Negative Saturation	Negative Saturation		
180.3	NS-RV	-26	-24.7	Not in use	-	Saturated	Saturated		
	NS-DB	-29	-24.7	Saturated	Saturated	Not in use	-	2.11	2.15
	NS-RV	-29	-26.8	Not in use	-	221	215.2	.27	.227
423.3	NS-DB	-38	-35.0	Saturated	Saturated	Not in use	-		
	NS-RV	-38	-35.4	Not in use	-	218	224	2.28	2.31
	NS-RV	-38	-36.4	Not in use	-	218	222.3		
715.3	NS-DB	-38	-34.4	Saturated	Saturated	Not in use	-		
	NS-RV	-38	-35.2	Not in use	-	218	216	2.37	2.39
1105.3	NS-DB	-20	-17.1	Saturated	Saturated	Not in use	-		
	NS-RV	-20	-17.4	Not in use	-	221	225	3.07	3.10
								X Axis Gyro	
1210.2								.13	.04
								.06	.07
								Z Axis Gyro	
								.39	.48
								.62	.61

Program DGMAP (which predicts the magnetic lines of flux of the earth) using the inertial stabilization option, was used to predict the theoretical P, Q, and R magnetometer readings to be expected at POGO and KODI for the predicted inertial position of the vehicle. The predicted and observed sensor readings and the predicted and observed pitch angles are shown in Figure 3-2. It can be seen that the R sensor reading at POGO established the attitude to be about 1° less than the true position of the vehicle if the .22° error is added to the difference on the graph. The P sensor is above the saturation level of the telemetry (TM) at POGO. At KODI the R sensor saturates the TM but the P sensor established the pitch attitude to be about .4° less than the actual.

The Q sensor (sensing yaw) was within 3 milligauss of the predicted values from DGMAP, which is equivalent to an error of about .33° (these values were not plotted). These values are well within the ±3° tolerance on attitude for RV ejection and will provide a reliable check on the position if inertial flight is used for a recovery.

3.4.3 Functional Health Check

To insure that the Lifeboat II system could perform a deboost if required, a functional health check was conducted on Rev 1210. The ACS thruster power was removed and Lifeboat II was allowed to control the vehicle for 30 seconds. The Lifeboat II performed as expected, pitching down approximately 20° and yawing approximately 14° before being reset.

3.5 ELECTRICAL DISTRIBUTION AND POWER SYSTEM (EDAP)

3.5.1 Solar Arrays

Solar arrays were extended on Rev 1. The left array started deployment on command and required 640 seconds to complete deployment and erection. The right array did not start to deploy until 1680 seconds after receiving the command and then required approximately 750 seconds to complete deployment and erection. Power output from each leg exceeded the specification value. Degradation from the initial output to end of the fourth segment was 2%. This is well below the 5% allocated.

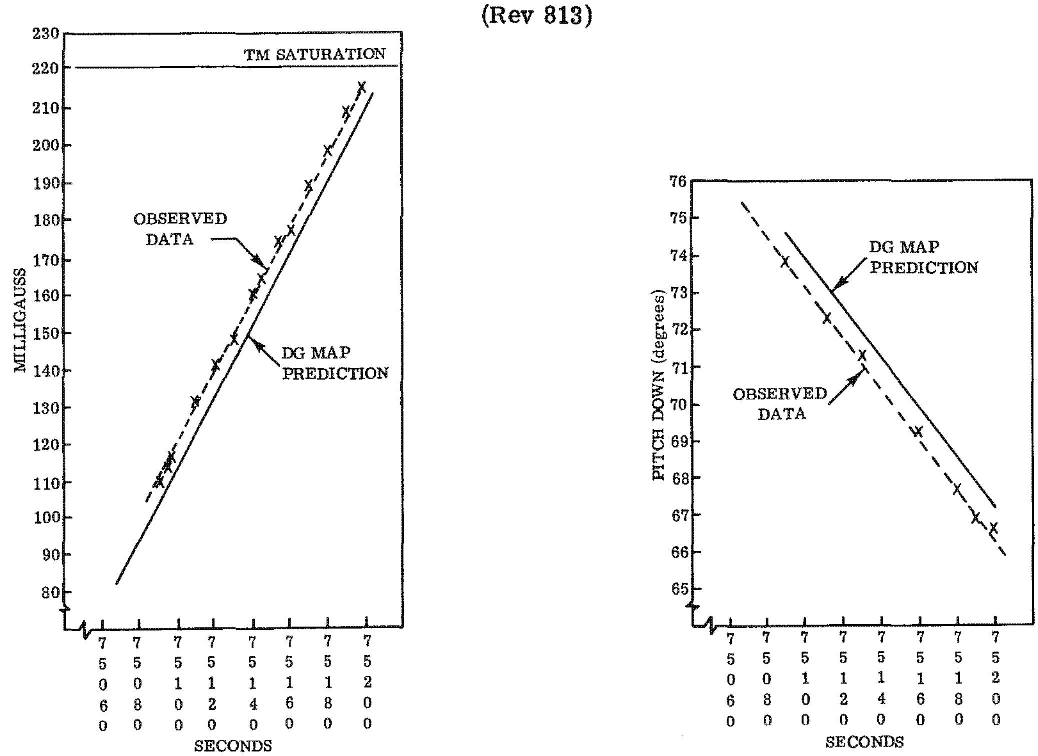
3.5.2 Main Bus Voltage

The main bus voltage varied from a low of 27.0 to a high of 31.8 volts. The allowable range was 25.5 to 33.0 volts. Low voltage data was obtained in the dark with a bus load of 70 amps. High voltage data was gathered during charge cycles.

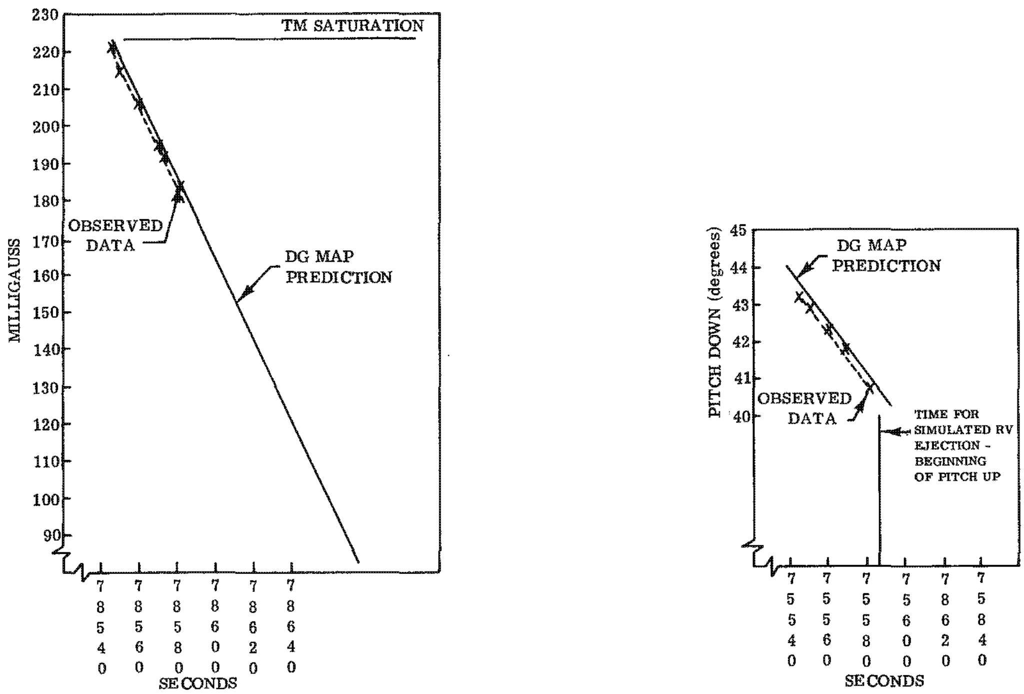
3.5.3 Power Capability and Usage

Power usage ranged from 217 to 352 amp hours/day. This is well below the 400 amp hour/day capability. Excess capacity was demonstrated by K2 charge relay cut offs occurring on Rev 7 and essentially every rev thereafter, except those with heavy payload operations. All Type 29 batteries operated in a desirable environment (42° F to 51° F) and performed normally throughout the mission. Pyro Battery 1 stabilized at 49° F, thus minimizing self discharge during recovery events. Twenty-three days

VEHICLE ATTITUDE AS DETERMINED BY LIFEBOAT II DURING INERTIAL RV RECOVERY EXPERIMENT (Rev 813)



R SENSOR IN N-S RV RECOVERY MODE OVER POGO (P SENSOR SATURATES TM)



P SENSOR IN S-N DEBOOST MODE OVER KODI (R SENSOR SATURATES TM)

FIGURE 3-2

after launch the battery left the peroxide region indicating a computed 3 amp hours had been removed, leaving 10 amp hours for continued use. After seventy days the calculated capacity was 6.3 amp hours. Cell degradation life still available was 46 days. Pyro Battery 2 followed the same pattern with the exception of leaving the peroxide operating region on Day 18. The Lifeboat Battery operated normally in a 49°F environment throughout the entire mission. A total of 97 amp hours remained at the end of the fourth segment from an initial 354 amp hours at launch. Remaining cell degradation life was 56 days.

3.6 TRACKING, TELEMETRY AND COMMAND (TT&C) SYSTEMS

3.6.1 Tracking

Tracking reductions were normally accomplished every 4 revs and were within the prescribed mission requirements.

3.6.2 Telemetry (TM)

TM system performance was satisfactory throughout the primary mission. During Solo (Rev 1129) the pulse code modulation (PCM), side 1, master multiplexor unit failed. A summary of usage through Rev 1463 (deboost) is given in Table 3-11.

TABLE 3-11

SUMMARY OF TM USAGE THROUGH REV 1463

Space Ground Link System (SGLS)	Side 1	Side 2
Number of ON/OFF cycles	1,404	145
Operational Time (minutes)	8,899	1,228
Pulse Code Modulation (PCM)		
Total Operational Time (minutes)	20,310	6,323
Number of ON/OFF Cycles	8,183	2,938
Tape Recorder	No. 1	No. 2
Number of ON/OFF cycles Record	8,708	865
Record Time (minutes)	15,966	1,036
Reproduce Time (minutes)	2,984	215
ON/OFF cycles Reproduce	1,024	78

3.6.3 Space Ground Link (SGLS) Performance

Down Link Signal Strength Fluctuations occurred, similar to SV-1, 2, and 3. The methods for predicting these dropouts resulted in excellent station pass planning with essentially no data loss.

SV-4 also experienced delayed acquisition on high elevation passes predominantly during passes with a maximum elevation of 38° $\pm 10°$ and on the left side of the vehicle. A comparison of output power, modulation loss on the SGLS subcarriers, and antenna patterns taken from SV-3 and 4 flight data show the only differences between vehicles are:

A. SV-4 was .5 db lower in output power than SV-3.

B. A modulation index analysis shows the modulation loss on SV-4 is .6 db greater than SV-3. This accounts for a total of 1.1 db difference. This difference could account for a delayed acquisition of up to 3 degrees.

The vehicle SGLS command equipment was utilized to receive approximately 11.2 million bits, with significantly more rejecting of commands than was experienced on previous vehicles, reference paragraph 3.6.8.

3.6.4 Instrumentation

Instrumentation successfully supported all mission requirements, with no instrumentation problems during the flight. SV-4 was launched with two known anomalies given in Table 3-12.

TABLE 3-12

INSTRUMENTATION ANOMALIES AT LIFT-OFF

ID No.	Description	Status
H012	PCM No. Bit Rate Indicator	Can Indicate 128 KB Instead of 48 KB
Z529	MS Temp Monitor	Invalid Readings

3.6.5 Command System

A summary of command system usage through deboost is presented in Table 3-13.

TABLE 3-13

SUMMARY OF COMMAND SYSTEM USAGE

System	Total Operating Time (hours)
Extended Command System	2,195
Minimal Command System	6
Remote Decoder	4.5
Backup Decoder	3.5

The extended command system (ECS) responded satisfactorily in all command modes resulting in the loading of 139,367 stored program commands (SPC) in memory; of these 139,367 SPCs loaded, 78,081 were output by both programmable memory units (PMU) for decoder processing. The remainder were erased prior to their time label matches. The accuracy of the clock throughout flight has been determined to be .23 parts in 10^6. The clock oscillator frequency changed .12 Hz in 91 days.

Both sides of the Remote Decoder were utilized for each of the four recoveries. Performance of both sides was determined to be acceptable through analysis of telemetry data.

The minimal command system (MCS) responded correctly to all commands. An MCS health test was performed successfully on Revs 18-20. The MCS was not used during the remainder of the primary mission.

The MCS was operated on two occasions during the Solo portion of this flight. The first MCS operational period was from 1200 POGO - 1201 POGO. This was a health check of the MCS done as part of the Solo objective to verify operation of all redundant systems. The second period was from Rev 1459 POGO - 1460 POGO. This operation was a test of the MCS capability to change the switched backup system (BUS) to the Lifeboat Battery. This test was conducted without any problems. The MCS responded properly to all commanding attempted during this segment. The backup decoder (BUD) was on for the period of the two MCS operations. There was no attempt to execute commands from the BUD during the flight.

3.6.6 375 MHz Receiver

The 375 MHz Receiver was powered during the entire mission with no anomalies.

3.6.7 Data Interface Unit

The data interface unit performed satisfactorily throughout the flight.

3.6.8 Command Reject Problem

SV-4 had significantly more rejects of commands during loading than any previous vehicle. The rejects were still of such a low frequency that they did not interfere with the primary mission. During Solo Transponder Two tests were conducted. On Rev 1149B and 1162 HULA, the stations were unable to load the messages scheduled for those stations. Both station passes were at approximately 6° elevation and the station was looking at the same side of the vehicle. This problem with SGLS-2 never recurred after 1162 HULA. On Rev 1209 KODI (SGLS-1), the attempt to load a message was hampered by many rejects (including four consecutive rejects of the same word) but the message load was eventually completed.

There were no more commanding problems until Rev 1267 KODI where there were 29 rejects in loading four message blocks. These problems continued with only KODI (Rev 1283, 1299, 1315) until Rev 1315 HULA. The same message was loaded without a single reject at POGO on Revs 1284, 1300, and 1316. Throughout the remainder of Solo, several test messages were loaded at every available opportunity. The data obtained from all the command loading exercises during Solo can be summarized in a few statements.

A. A significant number of rejects occurred only at KODI and HULA on SGLS-1.

B. HULA did not have problems when using their 60' antenna.

C. When an SGLS-1/SGLS-2 switch was made midpass at HULA, rejects almost disappeared.

The reject problem cannot be absolutely separated from the ECS since the bit read and S pulse on SGLS-1 are single point inputs to the command interface unit. However, it is highly unlikely that the ECS would have a failure that only occurred at KODI and HULA (and only on the 46' antenna at HULA). Investigation will continue into possible causes of these rejects utilizing some composite video tapes that were made at the tracking stations during commanding problems and another SGLS/ECS combination.

3.7 THERMAL CONTROL

3.7.1 Forward and Mid-sections

The Forward and Mid-section structural temperature control is summarized in Table 3-14. The data indicates that the Forward and Mid-section thermal design provides good control of payload temperature levels. No design changes are forthcoming as a result of flight performance.

TABLE 3-14

FORWARD AND MID-SECTION TEMPERATURES FOLLOWING INITIAL TRANSIENT

(° F)

Parameter	Design Limits	SV-4 Actuals
$\overline{T}_{FWD}$	47/93	80/83
$\overline{T}_{TCA}$	48/92	74/76
$\overline{T}_{FWD} - \overline{T}_{TCA}$	< 20	5/7
$\overline{T}_{SU}$	49/91	76/78
$\overline{T}_{SU} - \overline{T}_{TCA}$	6/-4	1/2

NOTE: The following are definitions of these parameters:

1. $\overline{T}_{FWD}$ - Average radiation temperature of the Forward-section derived from the average bulkhead temperature.
2. $\overline{T}_{TCA}$ - Average radiation temperature of the Forward compartment structure in the Mid-section.
3. $\overline{T}_{SU}$ - Average radiation temperature of the Aft compartment structure in the Mid-section.

3.7.2 Active Thermal Control

The active thermal control system performed normally throughout the primary mission. T_{REF} which represents the average Mid-section film path temperature was usually between 73° F and 75° F.

The RV heater zones which are actively controlled relative to T_{REF} were generally within 1° F of T_{REF} indicating adequate performance of the active thermal control system.

3.7.3 Aft-section

Acceptable Aft-section temperature control was maintained throughout the flight. All equipment remained within design temperature as indicated by the summary in Table 3-15.

Equipment and structure temperatures indicated contamination degradation to external vehicle thermal control surfaces similar to that noted on all other flights. The amount of degradation appears to be less than occurred on SV-1, about the same as SV-2, and greater than occurred on SV-3. The amount of degradation was within the bounds of preflight analysis, as is indicated by Figure 3-3, which compares actual door temperatures with the preflight predictions.

To provide capability for dual ACS operation on SV-4, a paint pattern change was made to the Bay 7 door and heat straps were added between the attitude reference module (ARM) and the Bay 6 door. It is estimated that these changes decreased IRA operating temperatures by approximately 30°F, resulting in satisfactory IRA temperatures during dual operation. Flight data indicates that the heat strap design performed as predicted based upon preflight analysis.

3.7.4 Contamination Experiments

3.7.4.1 Description

Additional contamination experiments were flown on SV-4 to measure two distinct contamination environments. The first group of experiments, installed on Bay 12 of the Aft-section as shown in Figure 3-4, were designed to measure the ascent contamination in terms of mass deposit and the effect of this mass deposit on the surface properties of flexible optical solar reflector (FOSR). Note that the blowoff shield was not used, as on SV-2 and SV-3, to isolate the effects of the ground lift-off cloud. Note also that the quartz crystal microbalances (QCM) have a mass rate channel added for this flight as was done for SV-3 to help interpret the readings of the mass channel. The second set of experiments were installed on the Aft bulkhead at Station 2203. These devices were designed to monitor the mass deposit produced by the RV retro motors and to assess the effect of the mass deposit on the surface properties of white silicone paint and FOSR.

3.7.4.2 Results

3.7.4.2.1 Aft-section - Bay 12

Orbital temperature data indicated the apparent solar absorptivity (α) values given in Table 3-16.

TABLE 3-15

AFT-SECTION CRITICAL COMPONENT TEMPERATURES

(° F)

Critical Component	Design Limits	SV-4 Actuals[1]
— EDAP —		
PDJB	-30/170	58/61
CCCs	-30/170	82/94
Batteries, Bay 3	35/70	41/45
Batteries, Bay 1	35/70	45/50
PDAs	-30/160	81/100
Solar Arrays	-125/225	-72/162
— ACS —		
IRA	50/130	93/113
HSA Heads	0/130	64/86
FCEA	-30/160	93/107
— OAS —		
Tank	65/100	79/91
Quad Valve	35/200	113/122[2]
Catalyst Bed	-	125/155[2]
— T&T —		
Tape Recorders	20/130	80/102
Transmitters	-30/170	80/103
PCM Master	-30/170	94/115
PCM Remote, Bay 2	-30/170	54/64
PCM Remote, Bay 10	-30/170	107/114
— COMMAND —		
PMU-A	-40/145	98/101
PMU-B	-40/145	111/114
Clock	-40/153	113/115
MCS	-40/149	98/103
— RCS —		
Tanks	40/140	72/104
Plumbing, Bay 12	35/140	81/94

NOTES: [1]Steady-state
[2]OA not firing

EQUIPMENT DOOR TEMPERATURES

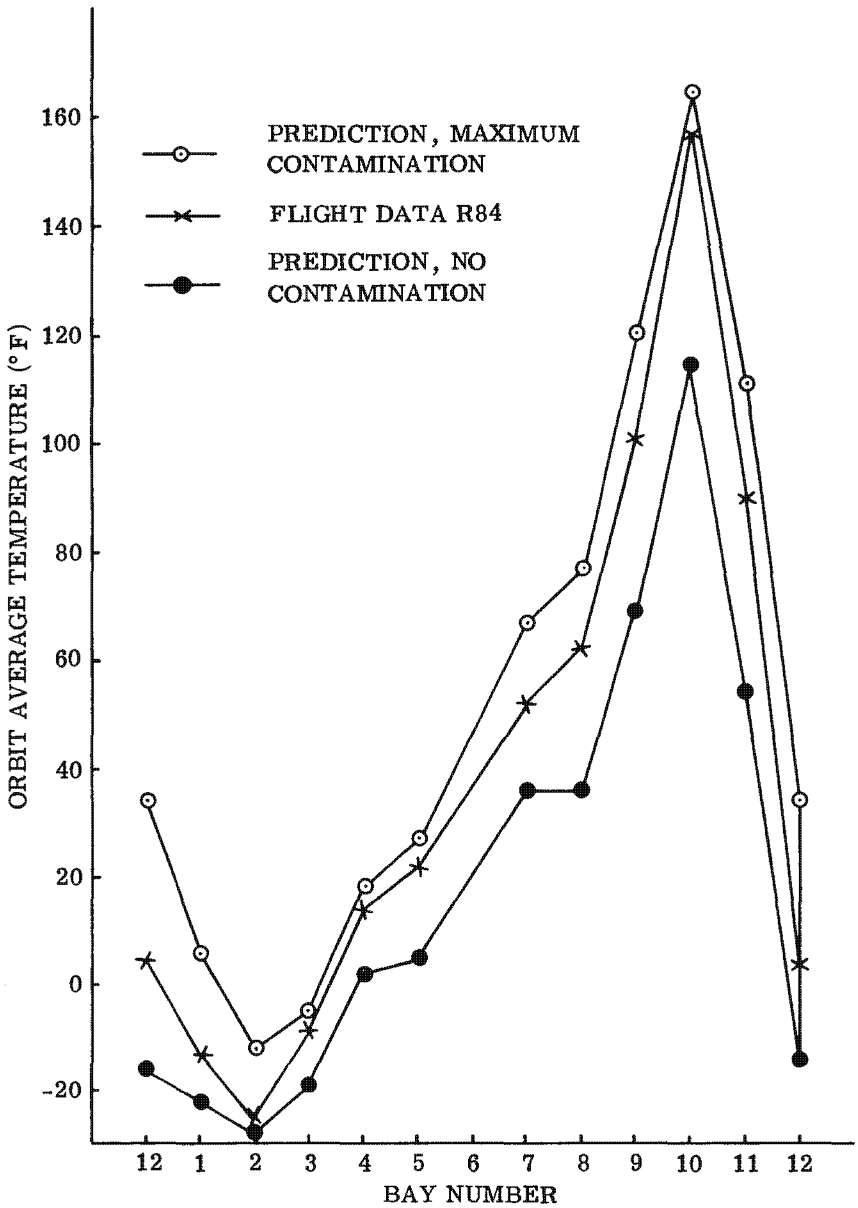


FIGURE 3-3

CONTAMINATION EXPERIMENTS

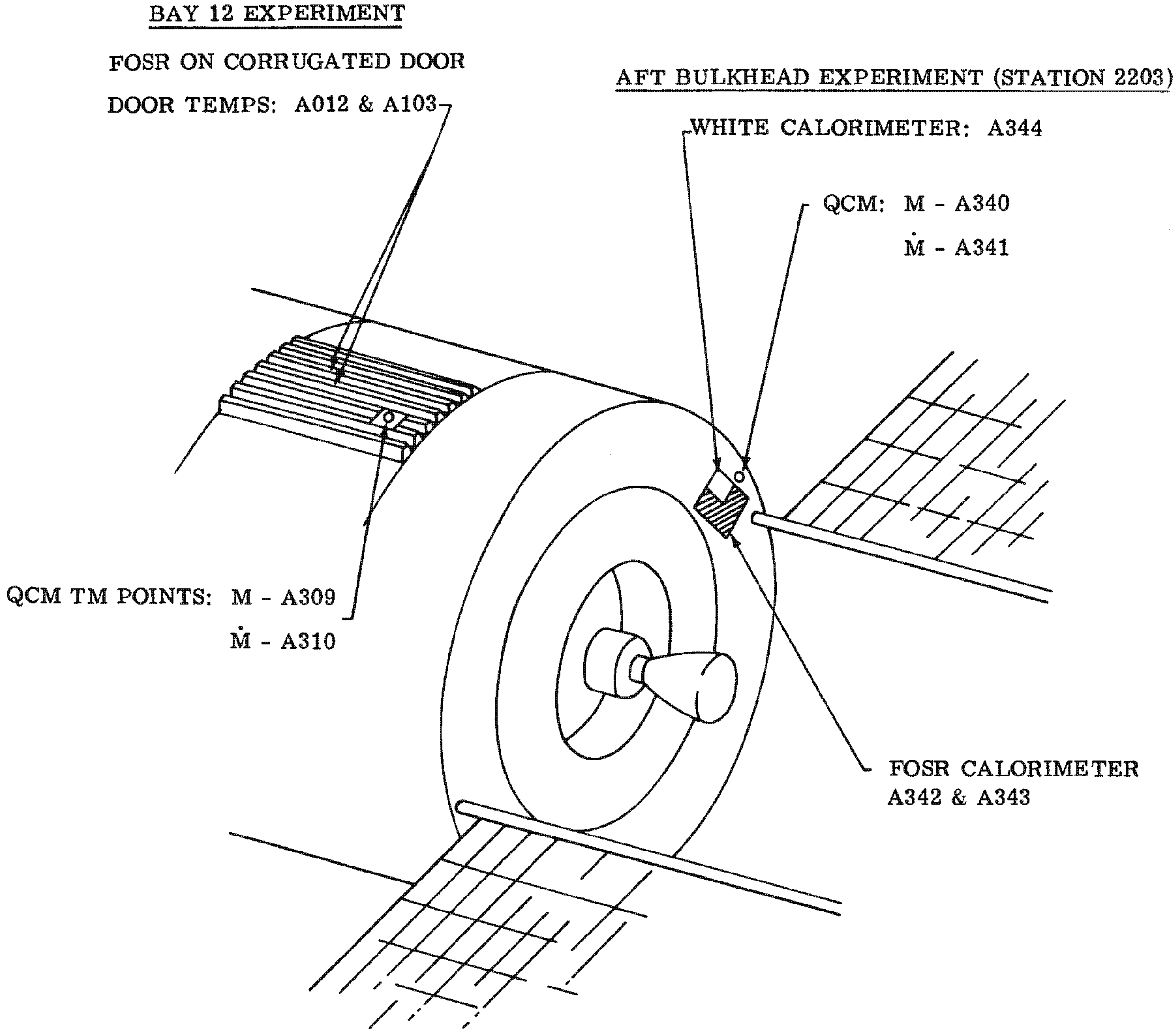


FIGURE 3-4

TABLE 3-16

APPARENT SOLAR ABSORPTIVITY

Surface	Preflight α	Orbital Data α	$\Delta\alpha$
White paint Bay 8	.22	.54	.32
White paint Bay 7	.22	.49	.27
FOSR Bay 12	.13	.29	.16

3.7.4.2.2 Aft Bulkhead

The ascent contamination caused negligible change in the mass deposition level measured by the Station 2203 QCM as shown in Figure 3-5. This was as expected, since the solid rocket motor (SRM) staging event causes only a momentary inflow of external gases into the Aft-section cavity via the vent ports. The first RV retro caused the mass deposit to rise from near lift-off levels to the telemetry limit of 5 TMV where it remained until the second RV retro. The extrapolated mass deposit due to the first RV was 3.8×10^{-5} g/cm^2. The second RV retro caused the QCM to saturate and the crystal stopped oscillating as is expected when the mass deposit becomes excessive. The FOSR and white silicone calorimeter panels indicated an increase in α of .10 and .13 respectively from early flight to after the RV-3 retro event. After correction for ultraviolet degradation, a total change in α of .06 for both calorimeters is associated with the three contamination events.

3.7.4.3 Conclusions

3.7.4.3.1 Aft-section - Bay 12

The ascent contamination exhibited the same general characteristics of deposit times and magnitude as did that of SV-2 and SV-3, and the SRM staging event again caused significant contamination. The total deposit again approached TM saturation and was slightly higher than SV-2 or SV-3. Again, the decrease in indicated mass deposit between 60 and 100 seconds after lift-off is caused by temperature gradients induced in the QCM during ascent aeroheating and does not indicate an actual mass loss. Orbital temperature data indicates that the FOSR tape successfully survived the ascent environments and experienced only about half the contamination degradation that areas of white paint experienced.

3.7.4.3.2 Aft Bulkhead

The mass deposition due to RV retro plume impingement generally agreed with analytical predictions (3.8×10^{-5} g/cm^2 measured per RV versus 2.8×10^{-5} g/cm^2 predicted). The somewhat unexpected result was that this mass deposit caused an average change in α of .02 per RV for white silicone paint whereas the same deposit on the Aft-section during SRM staging caused a .30 change in α. This result can be tentatively explained by theorizing that the mean particle size of the contaminating material produced during SRM staging (aluminum oxide particles plus miscellaneous solids from the BV

QUALITY CRYSTAL MICROBALANCE DATA

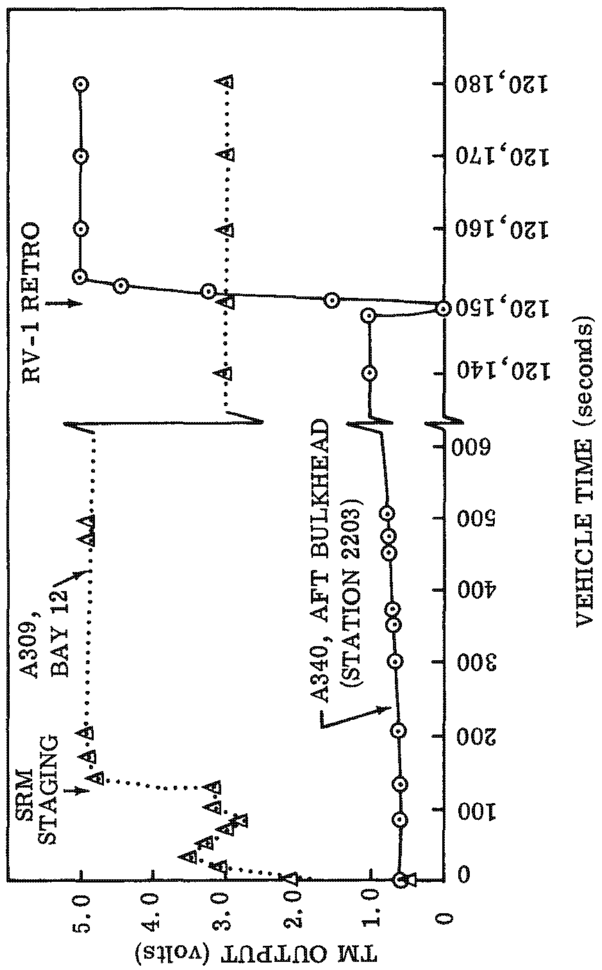


FIGURE 3-5

insulator pad) is significantly smaller than the particles produced by the RV retro motors (almost 100% aluminum oxide). A given mass of small particles will cover more surface area, and hence will cause larger changes in α than the same mass of large particles.

A specific objective of the Aft bulkhead experiment was to measure the contamination environment expected on the Quantic experiment scheduled for SV-8. The Quantic equipment installation was designed with shields to protect against the contaminants and the results reported above confirm that this protection is indeed mandatory.

3.7.4.4 Action for Subsequent Vehicles

FOSR is now considered to be fully qualified for flight and its extensive use in lieu of white silicone paint is being implemented for Aft-section external surfaces on SV-5 and up. No further contamination experiments are planned.

3.8 SOLAR ARRAY ERECTION ANOMALY

The erections and deployment time histories for both the left (-Y) and right (+Y) solar arrays are shown in Figure 3-6. Since the arrays were deployed and erected in the proper position for the flight beta angle, no positioning was necessary and none was performed during the basic mission. Positioning to the standard positions of $\pm 18°$ and 0° were accomplished on both arrays during the Solo mission.

The times for deployment and erection of the left array were similar to those for previous vehicles. The right array indicated that some motion was experienced by the deployment potentiometer at the time of the command but then no motion occurred for 28 minutes. The right array then deployed and erected in a normal manner. The delay was attributed to the rubber pads added to the restraining cage arms (on SV-4 only) sticking to the white silicone paint on the outer surface of the right array. Heat and plastic deformation of these pads eventually permitted the cage arms to disengage, and the subsequent motion was normal. The ability to prevent the cage from releasing the array was reproduced in the laboratory.

The conclusion that the right array had not been released by the cage mechanism was deduced from the additional curves shown in Figure 3-6. The electrical charging current for Rev 1 is shown, and also a curve derived by multiplying the current generated on Rev 2 by the solar array area exposed if the right array followed the postulated deployment. It is interesting to note that the erection motion has to occur before the arrays are in a position to produce energy. Also the temperature of panels 1, 5, and 10 of both arrays are shown and the beginning of deployment can be seen in the rapid rise of the panel 10 sensor. Because of the time delay, the sun angle on the right panel 10 was similar to that on the left panel 10 at the time of deployment, the left being illuminated near sunrise and the right near sunset.

A thorough review of the solar array was conducted and the following changes for SV-5 and subsequent vehicles have been implemented:

A. The rubber pads at the restraining cage arm tips were removed.

~~TOP SECRET~~-HEXAGON

TIME HISTORY OF SOLAR ARRAY PARAMETERS DURING DEPLOYMENT

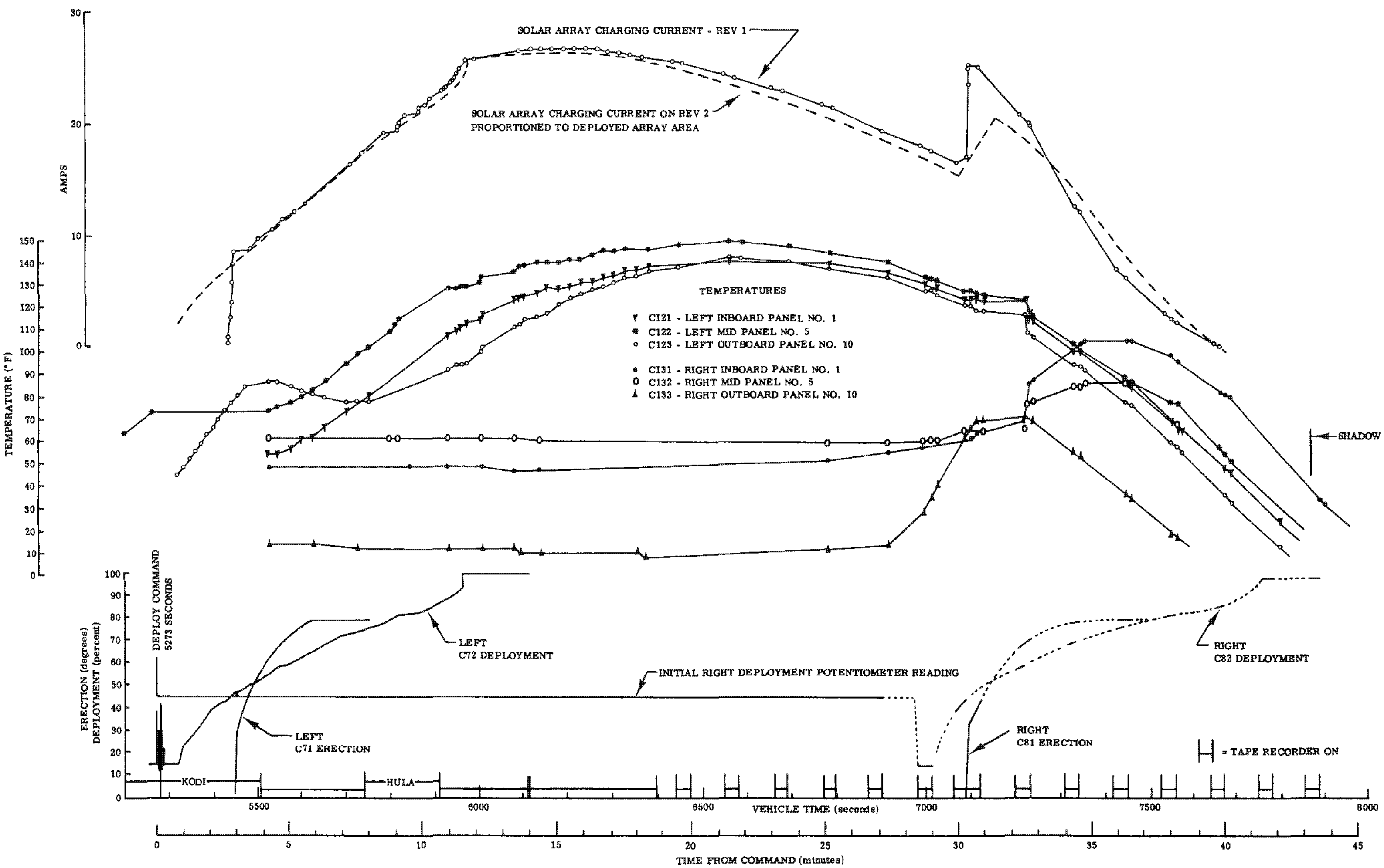


FIGURE 3-6

~~TOP SECRET~~-HEXAGON

B. Teflon blocks have been added to the tips of the two arms which disengage first in the sequence. This will support the solar leaves during ascent (this was the purpose of the rubber pads added on SV-4) but adhesion to the white silicone paint is minimized and more force is available to move these arms.

C. Teflon surfaces are being provided wherever silicone rubber contacted metal or other surfaces that could produce forces working against the solar array motion.

D. A spring has been added between panels 10 and 11 on the right array to prevent a theoretically possible overcenter interference.

E. Instrumentation has been added to aid in analyzing the solar array motion in flight.

F. Silicone rubber protection of the electrical cabling has been removed which should result in a faster deployment.

IV

PAYLOADS

SECTION IV

PAYLOADS

4.1 SENSOR SUBSYSTEM

4.1.1 Camera Operations and Performance

Mission 1204 provided the best overall image quality relative to previous HEXAGON missions. This was due primarily to the very small percent of photography acquired beyond ±45° of scan in combination with the relatively low altitude of 1204, versus 1201 and 1203. The importance of this observation is SO-255 Color Film, the first time such film has been flown with the HEXAGON Camera. The color film was transported with no problems and the resultant photography is of fair to good quality. The details of the evaluation of the SO-255 portion of Mission 1204 will be covered in a separate report. The only significant detractors to the success of Mission 1204 were the rewind and scan angle constraints imposed in the attempt to preclude film folds similar to those that occurred on Mission 1203 (see the 1203 PFA Report). These constraints were:

A. No 120° scan angle acquisitions.

B. No rewinds other than 5 ips.

C. No 30° scans at ±45° scan centers.

The rewind constraint most significantly affected the efficiency of film utilization.

Mission 1204 provided the best overall image quality relative to previous HEXAGON missions. This was due primarily to the very small percent of photography acquired beyond ±45° of scan in combination with the relatively low altitude of 1204, versus 1201 and 1203. The importance of this observation is realized fully when it is noted that the SV-4 Cameras were the lowest quality on-orbit (as measured by VEM) of the four camera systems flown to date.

Table 4-1 represents the percentage of United States Intelligence Board (USIB) targets against which 90% clear photographic coverage has been obtained through Mission 1204 to satisfy three month surveillance of target clusters, six month area search, and twelve month area search requirements.

TABLE 4-1

RESPONSE TO USIB REQUIREMENTS

Area	3 Months	6 Months	12 Months
China	88	91	96
Eastern Europe	78	88	70
Mongolia	N/A	95	98
North Korea	93	94	N/A
North Vietnam	60	62	N/A

TABLE 4-1 (CONT'D)

Area	3 Months	6 Months	12 Months
Mid East	100	99	98
USSR	71	86	85

4.1.2 Camera Data

The camera data for 1204 is summarized in Table 4-2.

TABLE 4-2

CAMERA STATISTICS

Parameter	Forward Camera	Aft Camera
Camera Designation	A	B
Film	1414/SO-255	1414
Focal Length (inches)	59.9916	59.9950
Equivalent Filter Type	Clear	W-12
Initial Focus Setting (μ)	98	80
Supply Footage (feet)	97,617/10,000	110,837
Supply Spool No.	5031	5016
Supply Film Weight (lbs)	768.2/105.7	872.3
Optical Set Nos.	024	012
Initial Pneumatics (lbs)	33.98	
Remaining Pneumatics (lbs)	2.5	

4.1.3 Focus

Mission 1204 was the best focused of the missions launched to date. The Forward Camera required an adjustment of 8 microns from launch nominal while the Aft Camera required no adjustment from launch nominal.

The improvement in flight focus settings is believed to be due to the new procedures employed in the focus data collection at the SVIC facility. New procedures and equipment were installed in Chamber A-2 which allows for pitching the vehicle and collecting resolution and focus data at any number of field angles with the same collimator.

During 1204-1, the SSC recommended, based on lateral separation focus sensor (LSFS) telemetry, that the Forward Camera platen be retreated by 11 microns and that the Aft Camera platen be advanced by 12 microns. Based on this recommendation the West Coast Project Office (WCPO) reviewed the focus data in the Flight Readiness Report, and decided to change focus by +4 microns on the Forward Camera and -4 microns on the Aft Camera. These changes were implemented on Rev 119. Analysis of both the thru focus engineering and operational photography at BRIDGEHEAD, during the PFA, indicated that another 4 micron platen retreat was necessary on the Forward Camera, and that a nullifying 4 micron platen retreat was desirable on the Aft Camera. Subsequent thru focus analysis on Mission 1204-2 and 1204-3 verified these final focus positions as being optimum. The Aft Camera initial adjustment of -4 microns was not, however, detrimental to imagery, since the image quality difference between the 80 micron platen position and the 76 micron position was slight. If focus had been adjusted during 1204-1 per the LSFS indications, the Forward Camera would have been set to within 3 microns of photographically determined optimum, while the Aft Camera would have been 12 microns out of focus. There is no question, at this point in time, that the LSFS should be ignored in making on-orbit focus decisions.

Line targets were deployed to test their usefulness for flight focus setting decisions, and were found to be useless for that purpose. The major problem is that adequate sample sizes simply cannot be obtained. Lines will not be further deployed for this purpose.

The new vehicle pitch procedure discussed above had indicated that the platens of both cameras were not optimally set, particularly on the Forward Camera. Whether the indicated platen error was indeed real, or there was some unknown problem with the pitch technique was not determinable in the Flight Readiness effort, and was left for PFA assessment. VEM and line analyses tend to indicate that the vehicle pitch produced field curvature (Platen Tilt) data is correct, that the platens were mistilted, and that this procedure is valid for system focus evaluations.

4.1.4 Photographic Image Quality

The photographic performance of both Mission 1204 cameras was acceptable. While the mean performance (considering all field/scan angles) between the two cameras was essentially the same, there were differences in performance worth noting. This analysis indicates that the resolution performance of the camera is scan mode dependent. A performance anomaly, referred to as "McDonald's Arches" is characterized by a decreasing level of performance at both beginning of scan and end of scan, with peak performance occurring at scan center, regardless of center location in scan length. This phenomenon is most noticeable on the Forward Camera, but examples of it exist on the Aft Camera as well. The resolution loss that has been measured can be as high as 80 lines/mm between the center of scan and the ends. This is particularly prevalent with the 90° scan lengths. However, losses of a similar magnitude do occur with some 60° scan lengths.

The cause of the above problem is not known, and it is currently under intensive investigation. Indeed, it is not clear that there is only one problem. The individual scan mode data indicates that, in many cases, the peak performance (at center of scan) is equal regardless of the level of fall off existing at the beginning and end of scan. This is not always the case, however, and some scan modes show overall depressed levels of performance. When one pools all the VEM major axis data into a single data set, a peaking in performance at nadir and a fall off at the beginning and end of scan is noted. This would not be expected if all scan modes produced the same resolution at scan center. Further analysis is required to understand this set of performance anomalies.

Analysis of the combined VEM data indicates that the mean area weighted averaged 2:1 contrast GRD for Mission 1204 was: 2.8' between ±30° of scan, 4.0' for acquisitions beyond ±30° of scan, and 9.0' for acquisitions beyond ±45° of scan. Based on the 2:1 contrast VEM data and grand area weighted average, resolution is estimated to have been 4.4'. This is the best of the four missions to date, comparing with values of 5.1' for 1201, 5.0' for 1202, and 6.9' for 1203. The major reasons for this superior overall quality are: (a) the constraints resulted in virtually no photography beyond ±45° of scan, and (b) the mission was flown at a reasonably low altitude.

4.1.5 Exposure

The exposure analysis performed on the black and white portion of the mission indicated a nominally exposed record requiring no correction to the basic recommendation. Biased snow surround scenes were lower in density than previous missions, but were above 1.0 density, thus requiring no alteration to the snow bias. Similarly, biased sand surround scenes in the Middle East were sampled and found to be at the nominal 1.0 density on the average. Uncorrected acquisitions of both types of surround also were investigated and indicated that the current biases, had they been applied, would have been appropriate. A new desert polygon in the Soviet Union was added during 1204-3 (on Op 381), but the bias was limited to -.10 log E because of dissimilarities between it and the Middle East polygons.

4.1.6 Operational Anomalies

Mission 1204 was relatively free of serious camera induced anomalies. There were, however, some problems worth highlighting. A relay panel in RV-4 disintegrated sending large numbers and sizes of epoxy resin chips into the system. Portions of these were retrieved in RV-2 and RV-3 as well as in RV-4. This material was found in the TU film stacks, and most likely was the cause of a large film tear that occurred in RV-3. This tear was nearly catastrophic. The RV-4 pyro battery exploded during re-entry, spilling potassium hydroxide over film and TU components. This severely damaged approximately 150' of both Forward and Aft Camera film and caused varying degrees of minor damage to several hundred feet of Aft Camera film.

4.1.7 Command and Control

Sensor subsystem performance was nominal with respect to command and control throughout the mission. All instrumentation was operational throughout the mission.

4.1.8 Exploitation Suitability

The overall interpretation suitability of Mission 1204 is Fair to Good being superior to 1203 and similar to 1202. Mission 1204 is believed to be better than 1203 because photo acquisitions were limited to scan angles below 45° for most operations.

Analysis showed good correlation between PI ratings, photographic scale (obliquity) and the GRD of intelligence targets. Most targets rated as good on Mission 1204 were of a scale of 130,000:1 or larger and a GRD of 4' or better. For a mission with an altitude of 90 NM this scale and GRD is achieved only at scan angles less than 30-35 degrees. Most targets with poor suitability ratings were acquired at scan angles greater than ±30 degrees.

MIP ratings for Mission 1204 ranged from 142 to 151 with an average of 148. The averages of Mission 1202 and 1203 are 150 and 135 respectively. This comparison of MIP averages is in agreement with previous statements that Mission 1204 is better than 1203 and similar to 1202.

4.1.9 Processing and Reproduction

Defilming was accomplished without major difficulty for RV-1 and RV-3. But loose, creased and tangled film was encountered in RV-2 and RV-4. It was necessary to disassemble the Aft builder roller on RV-2 to free film that was jammed under the assembly. The exploded pyro battery noted above dumped debris and electrolyte in the RV and on the film, degrading considerable imagery at the tail of both rolls.

Three high priority operations from 1204-4 Aft were extracted during defilming for a special forced speed process. These operations were underexposed relative to the normal process since the sun angle was very low and a larger slit was not available. The special process compensates for this under-exposure to sun angles as low as 1.2 degrees. It should only be used for priority targets, however, since locating the target on the film during defilming can be very time consuming.

NPIC analysis showed that there is no significant difference between BRIDGEHEAD and AFSPPF second generation DPs in terms of resolution transfer, granularity, acutance or tone reproduction. These results are consistent with 1203. The results for the second generation DNs from BRIDGEHEAD and the third generation DPs from AFSPPF show improvements in quality over past mission reproductions.

High contrast duplicate positives of selected low contrast original negative parts were provided to the PIs instead of normal contrast. The high contrast copies were preferred for both search and target readout in answering the essential elements of information (EEI).

4.2 SURVIVABILITY SYSTEMS

(b)(1)
(b)(3)

[redacted]

(b)(1)
(b)(3)

[redacted]

4.4 STELLAR TERRAIN SUBSYSTEM

There was no Stellar Terrain Subsystem flown on Mission 1204.

RE-ENTRY VEHICLE

SUMMARY

SECTION V

RE-ENTRY VEHICLE SUMMARY

5.1 SUMMARY

The recovery statistics are shown in Table 5-1. Performance of the re-entry vehicle (RV) subsystems is summarized in Table 5-2. Data indicates that all RV events (on-orbit, re-entry, and recovery) occurred as planned (except for delayed separation of the propulsion truss on RV-2) and the RV flights followed the predicted trajectories. The delayed truss separation had no subsequent effect on the flight.

The payloads on RV-1 and RV-2 were recovered in good condition. On RV-3 and RV-4 the fragments of an epoxy protective cover from a relay panel in RV-4 were found within the wraps and within RV-3 and RV-4. After the recovery of RV-4 the battery case failed, contaminating the loose outer wraps of the stock. The outer wraps were loose on RV-1, RV-2, and RV-4 due to payload rotation after aerial retrieval induced shearing of the core pins. Aerial retrieval loads exceeding the core pin strength were anticipated.

5.2 RE-ENTRY 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 exoatmospheric coast and retrograde phase of the re-entry trajectory.

5.3 RE-ENTRY VEHICLE SUBSYSTEM PERFORMANCE

Review of the re-entry vehicle subsystem indicates four anomalous conditions.

A. Delayed retro truss separation on RV-2.

B. Relay panel epoxy cover failure on RV-4.

C. Pyro battery case failure on RV-2 and RV-4.

D. Heavy load lines on main parachute broken on RV-1 and RV-2.

Fragments of the truss assembly on RV-2 provided positive evidence that truss separation did not occur normally even though the attaching bolts were fractured at the proper time. Available evidence indicates that the pyro system fractured all bolts but that friction or interference of the separating elements delayed separation. Truss attachment details have been redesigned to eliminate the shear pins and provide bolts with controlled separation planes. This redesign, which precludes recurrence of this anomaly, will be flown on all future flight vehicles.

Pressure buildup caused case failures of pyro batteries in RV-2 and RV-4. The failure on RV-2 was typical for a battery with a blocked vent. Leakage of gas products and some electrolyte occurred at the electrical connector seal at the case wall. On RV-4, the pressure buildup appears to have been of an explosive nature, causing the metal case to rupture. Failure analysis has not been completed to date.

~~TOP SECRET~~-HEXAGON

TABLE 5-1
RV RECOVERY SUMMARY

	RV-1	RV-2	RV-3	RV-4
RV Serial No.	20	19	18	17
Recovery Rev No.	180	423	715	1105
Recovery Date (1972)	21 Oct 1972	5 Nov 1972	23 Nov 1972	17 Dec 1972
Payload Weight (lbs) (Measured weight from recovered RV)				
Forward	223.1	229.0	211.5	213.3
Aft	224.8	229.0	216.5	203.7
Unbalance Percent	.7	.0	2.2	3.9
SV Orbit (hp x ha/ω p)[1]	89.2 x 148.6/145.9	87.9 x 157.3/131.6	90.9 x 145.2/132.2	91.7 x 144.7/119.6
SV Pitch Angle (degrees)	-37.9	-41.0	-41.8	-41.5
Nominal PIP Latitude	24.5°N	18.0°N	18.0°N	23.5°N
Impact Location Error (EPPD versus Teapot Eval)				
Overshoot (NM)	0.1	10.4	7.0	2.0
Cross-Track (NM)	1.2E	3.0W	2.8E	3.9W
Recovery (Aerial)				
Altitude (feet)	12,500	6,800	10,500	11,000
Parachute Condition	No Damage	No Damage	Minor Damage	No Damage
Retrieval Pass	1	3	1	2
Recovery Capsule Payload Condition	Good	Good	Good	RC Condition Good, Approximately 681 Feet of Payload Damaged by Battery Electrolyte.

[1] hp = Altitude of Perigee (NM), ha = Altitude of Apogee (NM), ωp = Arg of Perigee (degrees).

~~TOP SECRET~~-HEXAGON

TABLE 5-2

RE-ENTRY VEHICLE SUBSYSTEM PERFORMANCE SUMMARY

RV Subsystem/Function	Performance Assessment
On-Orbit Thermal Protection	Normal. T_{PL} Container = $T_{ref}{}^{+2°F}_{-5°F}$ Power Usage (Watts/RV) Maximum = 18 (First Day in Orbit). Stabilized = 6 (Sixth Day in Orbit). Allowable = 20.
Trim and Seal	Normal.
Electrical Power & Distribution	Normal during life of mission. All Batteries Activated. All Voltages 27.2 Volts. RV-2 Pyro Battery 2 leaked electrolyte. RV-4 Pyro Battery 2 had a seam rupture occurred onboard retrieval aircraft.
Sequential Subsystem	Normal on RV-1, RV-3, and RV-4. TM and postflight test verified RV-2 primary and redundant systems functioned properly for retro truss separation. Inspection indicates physical separation did not occur properly.
Pyro Subsystems	Normal. All primary and redundant pyrotechnics in each RV were verified by post flight inspection to have functioned properly.
Spin Stabilization	Normal.
Retro Motor	Normal.
Tracking, Telemetry, Instrumentation	Normal.
Heat Shield	Normal.
Base Thermal Protection	Normal.
Structure	Normal.
Recovery System	Normal. Heavy suspension lines were broken during tow period on RV-1 and RV-2.

In the interim, redesigned interval vents have been included in all future flight vehicle batteries (both pyro and main). This will eliminate pressure buildup and minimize the probability of case failure. Analyses are continuing to determine the cause of the failure.

Parachute behavior was essentially similar to that observed on prior flights except for the heavy suspension lines. During the aerial retrieval one of the eight heavy suspension lines was broken on RV-1 and three were broken on RV-2. The breaks are the result of abrasive action of other slack members during the tow period prior to boarding. This phenomenon is self limiting in that as lines are broken fewer slack members are present to cause abrasion. There will always be a sufficient number of active lines (strength) to successfully board the recovery capsule. Operational procedures, such as minimum tow time, aircraft velocity, and mild aircraft maneuvers are emphasized to minimize the damage due to abrasive action.

OPERATIONAL SUPPORT

VI

Approved for Release: 2020/12/01 C05131459

SECTION VI

OPERATIONAL SUPPORT

6.1 SOFTWARE

The software configuration used to support Mission 1204 was 'TUNITY MOD 1B with 'STAGEN, 'TELPRO, 'TAPOUT, AND 'TWURT modules providing greater flexibility. AOES/System II configuration was 13.1E SST Corrector Tape CT 13.E2. A nominal two rev load cycle was used during the mission through Day 62. Beginning on Day 63, a one rev load cycle for the payload revs was implemented. A total of 644 command messages were generated, of which 603 were loaded into the vehicle. Four 'TUNITY software problems which were considered flight critical were corrected during the mission. The flight critical software problem reports (SPR) are summarized below.

6.1.1 SPR M1B-4099 ('TOREP)

The 'TOREP output was scrambled on the transmission tape. The problem was determined to be flight critical because the user could not make use of the transmission tape. A change was made to 'TOREP correcting the problem and was incorporated on the Flight Aux Master.

6.1.2 SPR M1B-4116 ('TFIELD)

'TFIELD computed an incorrect value for slit on Rev 405. The values should have been .222 for Slit A and .303 for Slit B, but the values assembled for both sides were .080, Step 0. A change was made to 'TFIELD correcting the problem and was incorporated on the Flight Aux Master.

6.1.3 SPR M1B-4124 ('TSTAGEN)

An error in 'TSTAGEN caused an item to be passed to 'TOTEM to indicate that one station had a 25 minute duration. [redacted] The 'TSTAGEN assembled (b)(1)
ON/OFF commands were correct. A change was made to 'TSTAGEN correcting the problem and was (b)(3)
incorporated on the Flight Aux Master. An add on message had to be generated and the [redacted] manually inserted at the proper times.

6.1.4 SPR M1B-4120 ('TCATCH)

'TBALL assembled an operation on Rev 501 which was far below 'COPTI efficiency criteria. A change was made to 'TCATCH correcting the problem and was incorporated on the Flight Aux Master.

6.2 SATELLITE CONTROL FACILITY (SCF)

The performance of the Satellite Control Facility (SCF) in support of the fourth HEXAGON mission was commendable. Equipment and operational problems were encountered but were solved without impact on the mission. Command message generation and transmission, as well as down link TM reception and processing were satisfactory to support the operation. An SV-5 software exercise was conducted in parallel with mission operation during 1204-4.

6.2.1 Readiness

A one week exercise using 'TUNITY MOD 1B and MODEL 13-1E was conducted in parallel with mission operations during 1203-4. A 34 rev dress rehearsal was begun on 3 October 1972 and was successfully concluded on 5 October 1972.

6.2.2 Orbit Operation

One dedicated CDC 3800 computer was used throughout the operation; a second computer was used for 192 hours of operation. The computer usage rate was 1.117 computer hours per day. Table 6-1 provides a breakout by Remote Tracking Station of the anomalies that occurred during this operation. The paragraphs that follow discuss these anomalies.

TABLE 6-1
TRACKING STATION ANOMALIES
(occurrences)

Equipment	GTS	HTS	KTS	IOS	NHS	OL 5	VTS	STC
1230 mTc	0	1	2	0	0	1	0	--
CDC 160A	4	3	4	3	1	8	2	25
1200-bps dataline	0	3	3	0	3	20	2	--
Microwave system	--	--	--	--	--	--	--	9

A. HTS, Rev 34. The station could not achieve range-lock during pass. Fix: The station made repeated attempts to acquire range-lock, but it was unsuccessful. Postpass equipment checks indicated that the ranging system was operational. Impact: No ranging data was acquired.

B. VTS, Rev 925. Autotracking of the vehicle was delayed for 80 seconds because of an antenna operator error. Fix: After discovering his error, the operator was able to establish autotracking. Impact: The delay prevented the transmitting of the desired tape recorder commands, and some tape recorded data was lost.

C. KTS, Rev 91. A "no prepass disk" alarm occurred while the station was preparing to receive a command message. The backup prepass disk had to be used to receive the command message and to support the pass. After the pass, another "no prepass disk" alarm occurred while the computer was being initialized for a postpass playback; however, the station had no remaining prepass disk available because it had failed to duplicate the backup prepass disk before it was used. Fix: A new prepass was transmitted within three hours. Investigation of the problem by the resident software representative revealed that the combined effect of (1) an intermittent hardware fault in Disk Drive 2, and (2) a software deficiency caused the prepass disks to be erased. Octal correctors were generated and sent to all RTSs as a part of Corrector Set 3. Impact: The postpass playback was delayed for 4 hours.

D. KTS, Rev 1048. The 1230 m Tc computer could not be accessed by the SOC-II keyboards.

Fix: No real time action was possible, but the station was able to provide tracking and telemetry support. The problem disappeared during troubleshooting efforts and did not recur. Impact: The station had no commanding capability. The desired tape recorder commands could not be sent until Rev 1050 at GTS. Tape recorded data reception was delayed for two hours and fifty minutes.

E. VTS, Rev 154 and Subsequent Revs. During many passes the real time data transmitted via the microwave link to the STC was unusable. Fix: Western Union (WU) personnel performed a checkout of the microwave system and found that some of the equipment at the STC and at VTS was defective; this equipment was repaired. A procedure to checkout the microwave system during prepass checkout was developed and implemented. Also, WU will now perform a monthly check on the system. Prior to this time, maintenance was apparently performed only when an outage had been filed. Impact: Data transmission was delayed.

F. GTS, Rev 142. Reject alarms occurred after the first commands were transmitted. Fix: Ground-station personnel discovered that an incorrect command bit rate had been selected on the baseband assembly unit (1K vice 2K); the correct bit rate was then selected. Impact: Implementation of the command plan was delayed about 160 seconds.

G. OL 5, Rev 193. Autotracking was delayed about 83 seconds and telemetry data processing was delayed about 43 seconds because Cable P31010 became disconnected from the antenna console at ETA - 10 seconds (this cable routes the hold function signals that maintain the slave and autotrack modes when one of these modes has been selected). Fix: The cable was reconnected as quickly as possible. Impact: Tape recorded vehicle data could not be obtained until Rev 194 at OL 5, 90 minutes later.

H. HTS, Rev 383. SGLS commanding could not be accomplished because S-pulses were not being transmitted to the SGLS antenna. Fix: After the pass the problem was isolated to a bad contact on the S-pulse amplifier in Data Transceiver B; the contact was repaired. Impact: The required commanding had to be accomplished at the backup station (KTS) on Rev 383.

I. OL 5, Rev 475. The AOC was not selecting the leading zero in the command address (a four-digit entry must be made before the command can be transmitted), and Block 9233 could not be transmitted. Fix: The command and the address were reselected several times before the problem was identified and corrected. Impact: Required commanding was delayed for 121 seconds.

J. OL 5, Rev 765. The SGLS transmitter and 1230 m Tc computer could not be operated because station power, which is supplied by the host base, dropped to 100v. Fix: The station could only track the vehicle and record telemetry data. Impact: Required commanding had to be accomplished at the backup station (GTS) on Rev 766.

K. OL 5, Rev 1136. At ETA -2 minutes, the SGLS 14 transmitter's exciter failed. Fix: The station provided telemetry and passive ranging support; the exciter was repaired after the pass. Impact: The required commanding had to be accomplished at the backup station (HTS) on Rev 1137.

L. KTS experienced many reject alarms on Rev 1209 and subsequent revs. Reject alarms were occasionally experienced at other RTSs. Fix: Extensive station tests were performed, and detailed analyses were made of the vehicle command system telemetry data; however, the cause(s) of the problem could not be determined. The investigation of the problem is being continued. Impact: Some Solo phase tests were delayed.

6.2.3 Recovery Operations

6.2.3.1 Recovery 1

The first capsule was air recovered by NYLON 2 on Rev 180. Visual contact with the capsule was first reported by NYLON 3 and NYLON 4 at 2037Z when the aircraft were at 24,000 feet. At the time of recovery, the parachute was oscillating from 20° to 30°, and the cone collapsed slightly each time the parachute reached the limit of an excursion.

6.2.3.2 Recovery 2

The pilot of PINUP 2 intentionally pulled off from the first two recovery attempts because the capsule's parachute was unstable; the capsule was recovered on the third attempt. Visual contact with the capsule was reported by PINUP 3 at 2009Z at 25,000 feet, and by PINUP 2 at 2010Z at 23,000 feet. Both pilots described the parachute oscillation as violent. During its descent from 25,000 to 20,000 feet, the parachute oscillated from 40° to 50°; the system was rotating and breathing during the descent. At times, the breathing became extreme and the cone would sink into the main parachute deeper than the geodetic line opening. Neither the oscillations nor the breathing occurred in a cyclic fashion, so it was extremely difficult to predict the parachute's movement. The rate of descent varied from 1500 to 2000 fpm, and the variation appeared to be a function of the breathing. The movement of the system decreased as the capsule descended; however, the system never stabilized, and sudden, unpredictable 30° oscillations occurred throughout the entire descent. At the instant of contact, crew members heard a very loud popping sound; one of the heavy load lines was found broken after the system had been brought aboard the recovery craft.

6.2.3.3 Recovery 3

The third capsule was air recovered by FLESH 2. Visual contact with the capsule was reported by FLESH 3 at 2014Z when the aircraft was at 24,000 feet, and by FLESH 2 at 2015Z, when that aircraft was at 22,000 feet. The cone had a vertical split below the geodetic lines, and the parachute oscillated a maximum of 40° (30° average) until the capsule descended to 14,000 feet. The oscillations decreased somewhat during the remainder of the descent; however, the cone was pumping slightly on the recovery pass. Numerous lightweight suspension lines (skirt-to-swivel) were found broken. A heatshield search was implemented in accordance with the SPO's request. The heatshield was sighted at 2242Z and retrieved at 2253Z.

6.2.3.4 Recovery 4

The fourth capsule was successfully recovered by DELAY 2 during the second recovery attempt. Visual contact with the capsule was reported by DELAY 3 at 2114Z, when that aircraft was at 25,000 feet, and by DELAY 2 at 2118Z, when it was at 23,000 feet. The pilot of DELAY 2 made an intentional pull-off because the parachute was oscillating up to 25 degrees. The pilot had observed the action of the parachutes of the three previously recovered capsules and reported that this parachute appeared to be more stable than the others. The search for the heatshield was begun at 2150Z and terminated at 2350Z; however, the search was unsuccessful because of poor visibility that was caused by haze, strong surface winds, and moderately rough seas.

6.2.4 Command Message Generation

The two rev load cycle philosophy was employed during the first 62 days of operation. Under this concept, a base station pass (SP) message is generated twice a day and an add on message containing [redacted] (b)(1) (b)(3) latest weather data. The one rev load cycle concept will be used on all subsequent flights. There were 644 command messages generated during the primary mission; of which 4% were rejected. The message rejections were primarily caused by violations of hardware constraints and changes in payload selections.

APPENDIX A - REFERENCES

APPENDIX B - GLOSSARY

PERFORMANCE EVALUATION TEAM
REPORT NO. 1204/73

APPENDIX A

REFERENCES

1. HEXAGON Program Preliminary Post Flight Report Flight No. 4. Technical advisory report, BIF-107W-71002-73, 12 January 1973. (~~S~~/H)

2. Flight Test Engineering Analysis Report for the HEXAGON Program Satellite Vehicle No. 4 BIF-003W/2-068875-73, February 1973, LMSC Integrating Contractor. (~~TS~~/H)

3. Satellite Control Facility Operations Evaluation, January 1973. (~~S~~/H)

APPENDIX B

GLOSSARY OF TERMS

ACC Mode	Software Option that Finds Areas of Intelligence Value for a Rev Span.
ACS	Attitude Control System.
AFSPPF	Air Force Special Projects Production Facility.
Aft	Aft-looking Camera, Camera B.
AIM	Aerial Image Modulation.
ANDING	Logic Operation With Two or More Inputs, All of Which Must Be True For The Output To Be True.
ANOVA	Analysis of Variance.
AOB	Air Order of Battle.
AOES/System II	General Purpose Satellite Flight Support Software at STC.
AOR	Angle of Reflectance.
aprx	Approximately.
ARM	Attitude Reference Module.
AS	Aft-section.
ASE	Articulator Summed Error.
ATCS	Active Thermal Control System.
AUGIE	Acronym for Data Compression Technique Used for RTS to STC Data Transmission.
Aux Master	Auxiliary Master Tape. Contains Flight Support Software at STC.
AV	Aerospace Vehicle.
BBRT	Bird Buffer Retrieval Tape. Records at STC From Transmissions From RTS.
BFE	Best Fit Ephemeris.
BPI	Band of Peak Information.
BRIDGEHEAD/BH	Primary Film Processing and Immediate Post Flight Evaluation Facility.
BUD	Backup Decoder for MCS.
BUFT	Backup Film Transport.
BUS	Backup System.
BV	Booster Vehicle.
CATS	Camera-Target-Sun Angle.
C-	Camera Power-off Command.
C+	Camera Power-on Command.
CCC	Charge Current Controller.
CEI	Contract End Item.

APPENDIX B (CONT'D)

CG	Center of Gravity.
Chamber A	Photographic Vacuum Test Chamber Located at East Coast SSC Facility.
Chamber A-2	Photographic Vacuum Test Chamber Located at SVIC Facility.
CL	Centerline.
c/mm	Cycles Per Millimeter.
COMIREX	USIB Committee on Imagery Requirements and Exploitation.
CORN	Controlled Range Network.
CORREL	On-orbit Adjust Assembly Calibration Test Program.
CP	Corrector Plate.
cps	Cycles Per Second.
CRYSPER	On-orbit Performance Prediction Program Combining Target Acquisition, Atmospheric, Illumination, and Camera Performance Models.
CRYSTAL BALL	Photometric Atmospheric Model Computer Program. Used to Calculate Exposure for Orbital Acquisitions.
CV	Constant Velocity.
CW	Continuous Wave.
DCSE	Drive Capstan Summed Error.
DFC	Defenses/Security.
DGMAP	Computer Program for magnetic force field predictions.
DIM	Dynamic Image Motion.
DIU	Data Interface Unit.
D log E Curve	Sensitometric Response of Film to Light. Plot or Density to Log of Exposure.
DMAAC	Defense Mapping Agency Aerospace Center.
DMATC	Defense Mapping Agency Topographic Center.
DN	Duplicate Negative.
DP	Duplicate Positive.
DRAP	Pulse Code Modulation TM Data Retrieval and Analysis Program.
ECS	Extended Command System.
EDAP	Electrical Distribution and Power.
EEI	Essential Elements of Information.
ELC	Electronic.
EM	Electromechanical.
EMI	Electromagnetic Interference.

APPENDIX B (CONT'D)

E-MODEL	Engineering Model of the Sensor Subsystem.
EOD	Electro-Optical Department.
ESD	Emergency Shutdown.
ESO	Emergency Shutdown Override.
EXSUBCOM	Exploitation Subcommittee of COMIREX.
EXTRFPLS	Focal Plane Position Transducer and LSFS Reading Extractor Program.
FAFNIR	Program that Locates CORN Deployed Targets and Edge Catalog Targets.
FAK	Forward Assembly Kit.
FBS	Film to Bar Synchronization.
FCEA	Flight Control Electronics Assembly.
FDU	Failure Detection Unit.
FFL	Flange Focal Length.
FIDAP	Flash Image Displacement Analysis Program.
FOCMO	Thru Focus Motion Plot and Line Indicated Focus Program.
Forward/Fwd	Forward-looking Camera, Camera A.
FOSR	Flexible Optical Solar Reflector.
FP	Focal Plane.
FPA	Flight Profile Addendum.
FP-A	Focal Plane - Forward Camera.
FP-B	Focal Plane - Aft Camera.
FPP	Focal Plane Position.
fps	Feet Per Second.
FS	Forward-section.
FST	Flight Support Team.
FT	Film Transport.
FTF	Field Test Force.
FTFD	Field Test Force Director.
g	Acceleration Due to Gravity.
GAWA	Grand Area Weighted Average.
GMT/Z	Greenwich Mean Time.
GOB	Ground Order of Battle.
GRD	Ground Resolved Distance.

APPENDIX B (CONT'D)

HBT	Horizontal Baseline Test.
HFLIP	Data Strip and Print Program.
HOPE	Operational Performance Estimated Report. Summarizes Key Performance Related TM Data for Mission Engineering Operations.
HS	Horizon Sensor.
HSA	Horizon Sensor Assembly.
HWT	Hardwire Tester.
Hz	Cycles Per Second (Hertz).
IAS	Imagery Analysis Service.
ICD	Interface Control Document.
ID	Input Drive Capstan.
IMC	Image Motion Compensation.
IOR	Interop Runout.
ips	Inch(es) Per Second.
IR	Infrared.
IRA	Inertial Reference Assembly.
IV	Isolation Valve.
KALEIDOSCOPE/ KSCOPE	Radiometric Acquisition Model. Used to Calculate Basic Exposure Time Versus Solar Altitude, Haze Level, and Target Spectral Reflectance Characteristics.
LBS	Lifeboat System.
LMODE	Off-Line Program That Extracts Camera Data From BBRT for MPR Generation.
LP	Log Periodic Target.
LSFS	Lateral Separation Focus Sensor.
LT	Line Target.
MAA	Mission Analysis Area.
MACFACT	Mission Accomplishment Factor Program. Used to Process Key Performance Related Electromechanical Data.
MBGE	Missile Borne Guidance Equipment.
MC	Metering Capstan.
MCLR	Master Clear Off.
MCM	Mapping Camera Module.
MCRECON	TM Cross-Track Smear Estimate Program. Processes the Metering Capstan Summed Error Signal to Produce an Estimate of Film Motion and Absolute Smear Levels.

APPENDIX B (CONT'D)

MCS	Minimal Command System.
mcs	Meter Candle Seconds.
MCSE	Metering Capstan Summed Error.
MES	Mission Evaluation Score.
MEV	Million Electron Volts.
MFA	Measurement Filter Assembly.
MI	Measure of Interpretability Rating Technique.
MIP	Mission Information Potential.
MIPOLPER	Program which Combines the Optical Transfer Function Program with the Performance Prediction Program.
MMTF	Monochromatic Modulation Transfer Function.
MOD	Modification.
MONO	Monoscopic Operation.
MOP	Manual Operation.
MPR	Mission Performance Report.
MS	Mid-section.
MTF	Modulation Transfer Function.
MTF/AIM	Intersection of the Modulation Transfer Function and Aerial Image Modulation Curves.
MWC	Midwest Contractor.
NBS	National Bureau of Standards.
NCVU	Negative Constant Velocity Unit.
NEC	Northeast Contractor.
NISC	Naval Intelligence Support Center.
NM	Nautical Miles.
NOB	Naval Order of Battle.
NPIC	National Photographic Interpretation Center.
NSPC	Normal Stored Program Command.
NVR	Non-volatile Residue.
OA	Orbit Adjust.
OAK	NPIC Publication That Lists First Phase Exploitation Results.
OAS	Orbit Adjust System.
OB	Optical Bar.

APPENDIX B (CONT'D)

OD	Output Drive Capstan.
OFK	Orbital Fixed Knowns.
ON	Original Negative.
OP/Op	Camera System Operation.
OPD	Optical Path Differences.
OTD	Optical Technology Division of SSC.
OTF	Optical Transfer Function.
O^2A^2/OOAA	On-orbit Adjust Assembly.
P	X-axis Magnetometer Output.
PACS	Primary Attitude Control System.
PBF	Plane of Best Focus.
PCA	Point of Closest Approach.
PCM	Pulse Code Modulation.
PDA	Positional Drive Assembly (Solar Array).
PDJB	Power Distribution J-Box.
PDS	Pneumatics Distribution System.
PDWN	Pitch Down.
PERFORM	Camera Resolution Performance Prediction Program.
PERSAP	TM Resolution Performance Prediction Program. Estimates From Metering Capstan Telemetry and Measured Optical Performance.
PFA	Post Flight Analysis.
PFALINES	Post Flight Analysis Line Program. Computes 2:1 Resolution Performance and Estimates Image Motion Amplitudes.
PGR	Pitch Gyro Rate.
PI	Photointerpreter.
PIP	Predicted Impact Point.
PL	Proximate Line Target. The Line Target Which Has Been Deployed Closest to the Tribar Target.
P/L	Payload.
PME	Photo Mode Summed Error.
P-mode	Photographic Mode.
PMTF	Polychromatic Modulation Transfer Function.
PMU	Programmable Memory Unit.

APPENDIX B (CONT'D)

PN NEG (-)	Pneumatics-off.
PN PLUS (+)	Pneumatics-on.
PN/PNU	Pneumatics.
ppm	Pulse Per Minute.
PRF	Pulse Repetition Frequency.
PSA	Power Spectrum Analysis.
PSD	Power Spectral Density.
psi	Pounds Per Square Inch.
psia	Pounds Per Square Inch Absolute.
PVA	Pitch Vehicle Attitude.
PW	Pulse Width.
Q	Y-axis Magnetometer Output.
QCM	Quartz Crystal Microbalances.
R	Z-axis Magnetometer Output.
RACS	Redundant Attitude Control System.
rad/sec	Radians Per Second.
RCS	Reaction Control System.
REA	Reaction Engine Assembly (Thruster).
REM	Reaction Engine Module.
REV	Orbital Revolution.
RGR	Roll Gyro Rate.
RMS	Root Mean Square.
RPS	Reserve Power System.
RTS	Remote Tracking Station.
RV	Re-entry Recovery Vehicle.
RVA	Roll Vehicle Attitude.
RVTS	Re-entry Vehicle Test Station.
RWC	Rewind Constant.
RWV	Rewind Velocity.
SAL	Scan Angle Length.
SBA	Satellite Basic Assembly.
SBAC	Satellite Basic Assembly Contractor.

APPENDIX B (CONT'D)

SBAMS	Satellite Basic Assembly Mid-section.
SC	Scan Center.
SCC	Subsystem Command and Control.
SCF	Satellite Control Facilitv.
SDV-3	Satellite Development Vehicle.
SE	Solar Elevation.
SECO	Stage II Engine Shutoff.
SE Select	Software Option Which Selects Optimum Camera Op Sequence for a Rev Span.
Seq	Sequence.
SETS	Sensor Subsystem Engineering and Technical Support Staff.
SGLS	Space Ground Link System.
SLC-4E	Space Launch Complex-4 East.
SO Tape	System Output Tape at STC.
SOC	Satellite Operation Center.
SOF	Start of Frame.
Solo	System Engineering Test after Fourth RV Separation.
SPC	Stored Program Command.
SPEC	Specification.
SPL	Sound Pressure Level.
SPR	Software Problem Report.
SRM	Solid Rocket Motor.
SS	Sensor Subsystem.
SSC	Sensor Subsystem Contractor.
SSTC	Sensor Subsystem Test Console.
STC	Satellite Test Center.
SU	Supply Unit.
SURVEY	Quick-look Time and Data Characteristics Program.
SV	Satellite Vehicle.
SVACS	Satellite Vehicle Attitude Control System.
SVIC	Satellite Vehicle Integrating Contractor.
SVT	Satellite Vehicle Time.
SWT	Slit Width Tests.

APPENDIX B (CONT'D)

SYNCER	FIDAP Subroutine for Determining Film Synchronization Error.
TCA	Two-Camera Assembly.
TCS	Thermal Control System.
TCT	Test Control Team.
TM	Telemetry.
TMOTION	Estimate of Image Smear Program for Laboratory Tests.
TOBACC	Time for OB Velocity Command.
TTC	Tracking, Telemetry, and Command.
TU	Take-up Unit.
TUA	Take-up Assembly.
TUNITY	Computer Program for HEXAGON mission support at the STC.
TVC	Thrust Vector Control.
USIB	United States Intelligence Board.
UTB	Ultra Thin Base Film.
VAFB	Vandenberg Air Force Base.
VBE	Variable Block Erase.
VBT	Vertical Baseline Test.
VCO	Voltage Control Oscillator.
VDP	Vehicle Disturbance Program.
VEM	Visual Edge Match.
VI	Verification Interlock.
VIS	Vertical Integration Stand.
Vs	Coarse Film Path Velocity.
VSPC	Variable Stored Program Command.
Vx/h	Orbital Angular Rate, In-track.
Vy/h	Orbital Angular Rate, Cross-track.
WCFO	West Coast Field Office (Contractor).
WCPO	West Coast Project Office (Government).
YGR	Yaw Gyro Rate.
YVA	Yaw Vehicle Attitude.