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Performance Evaluation Team

JAN 3 1972

MISSION 1201

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

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PERFORMANCE EVALUATION TEAM

MISSION 1201

JAN 3 1978

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27 SEPTEMBER 1971

This report consists of 85 pages.

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PUBLICATION REVIEW

This report has been reviewed and is approved.

[redacted]

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

Lt Col, USA
Chairman, Performance Evaluation Team

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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 1201. Volume II is entitled Sensor Subsystem Post Flight Analysis Report, TCS 354016-71.

The report was prepared by the 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).

Editorial assistance and publication services were provided by the Air Force Special Projects Production Facility (AFSPPF). The PET wishes to commend the AFSPPF for the support provided by [redacted] Commander, and his most able staff. (b)(1)
(b)(3)

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SECTION I

SUMMARY

1.1 INTRODUCTION

The first HEXAGON Program Satellite Vehicle was placed into a planned 100 x 165 NM sun-synchronous orbit by the Titan III D Booster Vehicle at 1841Z, 15 June 1971. Photographic operations were performed on days two through 31. The Reentry Vehicles were deorbited on Revs 82, 179, 405 and 502. RV-1 was recovered from the water and RVs 2 and 4 were aerially recovered. RV-3 was not recovered. The best ground resolution (average in-track and cross-track) subjectively obtained from the CORN tribar targets was 2.3 feet.

1.2 CONSTRAINTS

The constraints outlined below are those applicable to HEXAGON Mission 1201.

1.2.1 Preflight

The following constraints were imposed before launch:

A. Solar heating (Beta) angle from -20 to +10 degrees.

B. Altitude range of 93 to 135 NM over illuminated area of interest.

C. No orbit adjusts prior to first recovery.

D. First recovery on day five.

E. Fixed minimal rewind.

F. No negative scan centers during mono operations.

G. No operational photography prior to sensor subsystem validation through engineering photography within COOK tracking station cone.

H. RV mass imbalance not to exceed 60% for first unbalanced RV.

1.2.2 On-Orbit

Constraints imposed on-orbit were:

A. After Emergency Shut Down (ESD) on Rev 314, the 30° scan width was prohibited from Revs 326 to 440.

B. After ESD on Rev 445 all negative scan centers and 30° scan widths were prohibited in the mono mode from Revs 471 to 476.

C. Recovery for RVs 3 and 4 was restricted to areas within range of land based helicopter support.

D. Max allowable load for RV-4 was 50%.

E. Sensor subsystem operating time was restricted to 17 minutes per fixed block of four revs from Revs 20-35 and 30 minutes from Revs 36-100. From Rev 101 to end of mission, operating time was restricted to 30 minutes per sliding four-rev span.

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1.3 SENSOR SUBSYSTEM PERFORMANCE

During Mission 1201 the sensor subsystem had four ESDs. Three of these shut downs were caused by problems in the film path. The fourth was caused by incompatible Vx/h and film velocity (V_s) inputs from the Extended Command System (ECS). On-orbit efforts to clear these ESDs were successful in every case. Nominal performance was experienced in the remainder of the sensor subsystem.

1.4 SATELLITE VEHICLE SYSTEM PERFORMANCE

The Titan III D Booster Vehicle (BV) performed satisfactorily, injecting the Satellite Vehicle (SV) into orbit with the following differences from the planned parameters.

Perigee Altitude	-0.216 NM
Perigee Argument	+6.168 Deg
Period	+0.004 Min
Inclination	-0.005 Deg

The BV Command Guidance System performed satisfactorily. The overall performance of the SV was excellent and all HEXAGON mission objectives except the recovery of RV-3 were met. Following a 21 day SOLO (operations beyond the primary mission) operation (which is not described in this report), the SV was successfully deboosted. The overall SV performance for the ascent phase and each of the four mission segments is summarized as follows:

A. Ascent. Ascent events were nominal and stabilization of the SV allowed deployment of the Solar Arrays on Rev 1. Apparent contamination of Aft Section thermal control surfaces during ascent caused an over-temperature condition in the battery module which remained constant throughout the mission.

B. Segment One. By Rev 16, all subsystem health checks had been completed and operational photography began on Rev 24. On Rev 82, RV-1 was successfully separated with a total film load of 40,000 feet. Damage to the aerial retrieval target cone was observed which led to the decision to allow the RV to water impact. The RV was recovered from the water with no damage to the payload.

C. Segment Two. Operational photography continued on Rev 88 using RV-2. On Rev 179, RV-2 was successfully separated with a total film load of 52,000 feet. Main parachute damage occurred but aerial recovery of the RV was successful.

D. Segment Three. Operational photography continued on Rev 185 using RV-3. On Rev 405, RV-3 was successfully separated but was not sighted nor recovered. Major damage to the main parachute apparently occurred during deployment. As a result of this malfunction, the film load of 54,000 feet was lost.

E. Segment Four. Following some difficulties with ESDs, operational photography was resumed on Rev 470. The premature degradation of the pyro batteries led to the decision to separate RV-4 on

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Rev 502. Separation and aerial retrieval of RV-4 was entirely normal and 26,000 feet of film was recovered.

1.5 SATELLITE BASIC ASSEMBLY PERFORMANCE

The performance of the Satellite Basic Assembly (SBA) subsystems was acceptable with few anomalies, with the exception of the electrical distribution and power subsystem. All primary equipment functioned throughout the four segments and no backup equipment was required. Subsystem performance is summarized below with detailed discussions provided in later sections of this report.

A. Attitude Control System (ACS). The ACS met performance requirements in all operating modes.

B. Reaction Control System (RCS). Apparent early degradation of thruster pulse shape was observed but with no impact on control of the SV. The cause has not yet been determined, but no changes are indicated at this time. Propellant capacity is adequate for the design mission.

C. Electrical Distribution and Power (EDAP). The Main Power System meets requirements. The Reserve Power System (RPS), carried on this vehicle only, was not required. No adjustments are necessary for the design mission. Relocation of Bay 12 batteries to Bay 3 is necessary until the ascent contamination problem is eliminated. The pyro power system is marginal.

D. Orbit Adjust System (OAS). The system meets performance requirements. No adjustments are necessary for the design mission. Propellant capacity is adequate for the design mission.

E. Tracking, Telemetry, and Command (TT&C). The system meets performance requirements and no adjustments are necessary for the design mission. Antenna performance lead to the recommendation to restrict secure word block commanding to favorable station elevation angles. Extended Command System (ECS) logic errors prohibit use of certain 11- and 12-bit commands. Impact of these constraints is minor. Instrumentation was adequate for SV control and diagnosis.

F. Lifeboat II. Health checks showed performance met the requirements for this mission. However, neither a demonstration of operation at the extreme of the orbit regime nor a simulated Lifeboat capture was attempted during primary mission.

G. Structures and Mechanisms. Performance requirements were met and no adjustments are necessary for the design mission. Solution of the contamination problem may require new ejectable shields over the Aft Section thermal control surfaces.

H. Thermal Control System (TCS). Active and passive thermal control designs met operational requirements except for the Bay 12 over-temperature condition associated with ascent contamination. A solar heating (beta) angle constraint is necessary until the contamination problem is eliminated but no changes to the basic thermal design are indicated.

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1.6 ANOMALY SUMMARY

Significant anomalies and malfunctions are listed chronologically in Table 1-1. The list includes a description of the anomaly, the mission consequences and in some cases, the changes indicated for subsequent vehicles. Detailed discussions of these anomalies can be found in appropriate sections of this report.

1.7 CONCLUSIONS

A. There was thermal surface contamination over the entire SBA on Mission 1201.

B. The catastrophic parachute failure experienced on RV-3 requires a fix prior to Mission 1202.

C. The validity of the current center of gravity (CG) offset criteria is questionable in view of the RV performance on Mission 1201.

D. The demonstrated RV performance was less than predicted.

E. The overheating of the Type 29 and Lifeboat batteries requires a fix prior to Mission 1202.

F. The available pyro battery energy for flight must be increased.

G. The power system demonstrated adequate capacity for unrestricted vehicle operation of block one vehicles. However, management of the system while anomalies are present, such as the overheated batteries, was inadequate.

H. There is a logic error in the A decoder of the Extended Command System (ECS).

1.8 RECOMMENDATIONS

A. The parachute design and fabrication should be improved to assure aerial retrieval as soon as possible.

B. Constrain the orbit for Mission 1202 to be compatible with the RV and SBA capability limits demonstrated on Mission 1201, i.e., nominal design orbit, and solar heating (beta) angle similar to that of Mission 1201.

C. Institute a program to adequately define the requirements and capabilities of the Reentry Vehicles.

D. Eliminate the decoder A logic error. Until such time as a permanent fix is effected, procedures should be developed to work around the decoder logic problem. These procedures should have minimum impact on mission operations. In this light, a thorough review of the feasibility of using decoder B as primary should be made.

E. Develop a power management system which will provide full knowledge of the energy available considering the most probable power system anomalies.

F. Provide more pyro battery capacity.

G. The antenna performance should be reviewed to determine possible improvement.

H. Move the batteries from bay 12 to a colder location, i.e., bay 3.

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TABLE 1-1

ANOMALIES

Rev	Description	Impact
Ascent	HSA output transient	Primary roll and pitch output transients observed during SV/BV separation. Apparently caused by microphonics resulting from separation shock. Operation normal on orbit so no mission impact. Design changes on SV-2 are not indicated although SV-7 and up design will not be as susceptible to microphonics.
Ascent	Acoustic microphone	Microphone sensitivity to static pressure pulses at lift-off caused data degradation. Corrective high pass filters to be installed on SV-2 microphone amplifiers.
2	Hot Aft Section	Over-temperature condition on Bay 12 battery module resulting in degraded main power system capability and early pyro battery depletion. SV-2 to fly with battery module moved to Bay 3, beta angle restricted to one near the one flown on Mission 1201, and contamination experiments will be installed in Bays 11 and 12 to identify problem solution. Appropriate solution planned to be implemented on SV-3.
Throughout	Data drop-outs	Space Ground Links System (SGLS) antenna pattern holes excluded secure work loading at high station elevations. Operational restrictions are not severe and will be continued on future vehicles. No design changes indicated.
82	RV-1 chute damage	Damaged chute sighted by recovery forces. Water impact allowed with successful recovery.
179	RV-2 chute damage	Damaged chute sighted but successful aerial recovery implemented.
314	Emergency shut down	Jam in fine film drive system. Normal ops resumed following constant velocity and engineering tests. No design changes indicated.
402	Emergency shut down	Apparent temporary obstruction in coarse film path. Normal ops resumed following creep and constant velocity tests. No design changes indicated.
405	RV-3 chute failure	RV not sighted, nor recovered. Apparent premature disreef of main chute caused structural failure.
435, 459, 460	RCS Thruster pulse shape distortion	Distortion similar to but not as severe as ground test experience occurred prematurely. Vehicle control entirely adequate and extra propellant usage not indicated. Cause and corrective action under investigation. No design changes indicated.

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TABLE 1-1 (CONT'D)

ANOMALIES

Rev	Description	Impact
474, 476	RCS Thruster thrust level shift	Thrust level shift similar to ground test experience. Thrust remained within limits and showed a tendency to return toward normal conditions in a subsequent firing. In mono ops, roll rate exceeded fine mode rate limits. To be studied further. No design changes indicated.
445	Emergency shut down	Imput drive capstan in fine film drive system stopped rotating. Cause is unknown. Normal ops resumed following mono ops during 5 revs and a recycle operation. No design changes indicated.
484	Pyro battery depletion	Pyro battery number 1 began rapid voltage decay earlier than anticipated. Review of battery duty cycle and temperature environment revealed degradation to be predictable. Reduced loading and cooler environment for SV-2 are planned.
492	Emergency shut down	ECS logic problem caused erroneous commands which caused ESD. SV-2 through SV-4 will restrict use of certain Variable Stored Program Commands (VSPC). SV-5 will incorporate modified ECS.

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Satellite Basic Assembly Subsystem

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SECTION II

SATELLITE BASIC ASSEMBLY SUBSYSTEMS

2.1 ATTITUDE CONTROL SYSTEM

The SV-1 Attitude Control System (ACS) performed as expected and met all specifications that could be measured. The summaries presented in the ACS section detail those requirements that could be verified from flight data.

2.1.1 Booster Vehicle/Satellite Vehicle (BVSV) Separation

BV/SV separation occurred at 545.2 sec vehicle time. Master clear OFF (which enables the pitch, roll and yaw integrators to accumulate angle), was at 513.4 sec and Stage II Engine Cut Off (SECO), which terminates BV attitude control, occurred at 533.3 sec vehicle time. The maximum rate and attitude excursions attained by the SV following master clear OFF and the times in which the SV attitudes and rates came back within the specified limits following BV/SV separation are shown in Table 2-1.

TABLE 2-1

BV/SV SEPARATION

	From Master Clear Off to Separation				Capture			
	Rate		Attitude Excursion		Attitude Angle Spec/ HS Meas (deg)	Attitude Set Time Spec/ Meas (sec)	Rate Spec/ Meas (deg/sec)	Settling Time Rate Spec/ Meas (sec)
	Max Spec (deg/sec)	Actual (deg/sec)	Max Spec (deg)	Actual (deg)				
Pitch	±.752	-.22	-22.85 to +9.63	+.94	±.70/ <±.30	1500/ 667	±.014/ +.010	1500/ 83
Roll	±.786	+.34	-7.50 to +10.94	+.75	±.70/ <±.30	1500/ 667	±.021/ +.020	1500/ 31
Yaw	±.752	+.19	-7.66 to +11.50	+2.1	±.64/ N/A	1500/ 667	±.014/ -.010	1500/ 630

2.1.2 Payload Operations

To evaluate the SV rate performance for the stereo payload (P/L) operations, one typical operation from each RV load was closely examined and the results are shown in Table 2-2. The rate data from several other P/L operations during each segment of the mission were scanned and the values shown are representative of those observed. In all cases the SV rates and integrator attitudes were held within their respective switching lines during the P/L operations following the startup transient. P/L operations

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TABLE 2-2

STEREO PAYLOAD OPERATIONS

Segment/ Rev	Peak Rates During Start/Stop (deg/sec)			Peak Rates During Steady State Ops (deg/sec)			Settling Time (sec)
	Pitch*	Roll	Yaw	Pitch	Roll	Yaw	
	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.
1/74.3	None/-.001	None/ .009	None/-.007	±.014/-.001	±.021/-.005	±.014/-.001	0.2/0
2/155.4	None/-.003	None/-.006	None/ .009	±.014/-.001	±.021/-.006	±.014/ .009	0.2/0
3/381.3	None/-.001	None/ .004	None/-.010	±.014/-.001	±.021/-.006	±.014/-.001	0.2/0
4/477.4	None/-.006	None/ .008	None/ .011	±.014/-.001	±.021/-.002	±.014/-.001	0.2/0

MONO PAYLOAD OPERATIONS

Segment/ Rev	Peak Rates During Start/Stop (deg/sec)			Peak Rates During Steady State Ops (deg/sec)			Settling Time (sec)	Roll Control Torque P/R Avail	P/L Roll Torque Required
	Pitch	Roll	Yaw	Pitch	Roll	Yaw			
	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.	Spec/Act.
4/476.3	None/-.012	None/ .047	None/ .007	±.014/ .002	±.021/ .008	±.014/-.007	6.6/0.0	>5.0/>7.0	<5.0/<6.0
4/474.4	None/-.012	None/ .026	None/ .008	±.014/-.012	±.021/ .010	±.014/ .007	6.6/0.0	>5.0/>7.0	<5.0/<6.0

*Geocentric Rate removed from pitch rate values.

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were examined that used several combinations of scan angle and scan center positions with no apparent differences between them.

In mono operations, the roll rate stayed within the required limits for P/L operations following the film transport on command; however, the roll rate during the start and stop of the Optical Bars (OBs) exceeded the fine mode rate switching line of 0.0153 deg/sec. This was not expected since the level of control torque was thought to be more than the level of OB roll torque. This discrepancy is under study and SOLO experiments (operations beyond the primary mission) were carried out on Revs 629 and 645 to help resolve this question. See also discussion on Reaction Control System (RCS) thrusters in paragraph 2.2.2.

A time history of the vehicle attitude and rates for a typical stereo operation are shown in Figure 2-1 (2 pages). Figure 2-2 (1 page) contains vehicle attitude control data for a payload Emergency Shut Down (ESD), while Figure 2-3 (6 pages) illustrates the system performance observed for RV-1 pitch down (PDWN) and RV separation. A compilation of vehicle attitude and rates for all typical maneuvers has been compiled and will be published as an Aerospace report (Attitude Control System Maneuver Histories) in the near future.

2.1.3 Recovery

The pitch down maneuvers preceding the RV separations are summarized in Table 2-3. The pitch down angle is read from the Satellite Test Center (STC) PDWN real-time data which is a software computed number - not a directly measured value. The maneuvering time is the time interval from the pitch down rate command to the time the rate returns to a value of less than 0.1 deg/sec after removal of the command. Prior to the first RV separation, a pitch down and pitch up test was run on Rev 66 which is included at the bottom of Table 2-3. No other data on a pitch up maneuver without the RV separation impulse was obtained during the basic mission.

The attitude channel maneuver rate command was -.705 deg/sec; however, the nominal expected coast rate is the summation of the rate channel maneuver rate command of 0.5 deg/sec plus the effective attitude channel saturation level of 0.5 deg divided by the rate to attitude gain ratio of 2.0 sec for an expected rate of $0.5 + 1/2 \times 0.5 = 0.75$ deg/sec. The RV/SV separation performance is summarized in Table 2-4.

2.1.4 Orbit Adjust

The disturbances resulting from the Orbit Adjust (OA) firings were within the predicted magnitudes and well within the dead band limits of the attitude control system and satisfied the settling time requirements of 20 sec. Typical attitude and rate histories for an orbit adjust are contained in the Attitude Control System Maneuver Summary Report.

The OA burn influence on the control system followed the predicted trends. The predicted and measured propellant expenditures are shown in paragraph 2.1.8.1.

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ACS PERFORMANCE FOR STEREO OPERATION EVENT TIMES:

Camera A on	396448.0
Camera B on	396448.4
Optical bars on	396450.6
Film transports on	396460.6
Optical bars off	396562.8
Film transports off	396564.0
Camera power off	396607.4

Data is provided in Figures 2-1a through 2-1b

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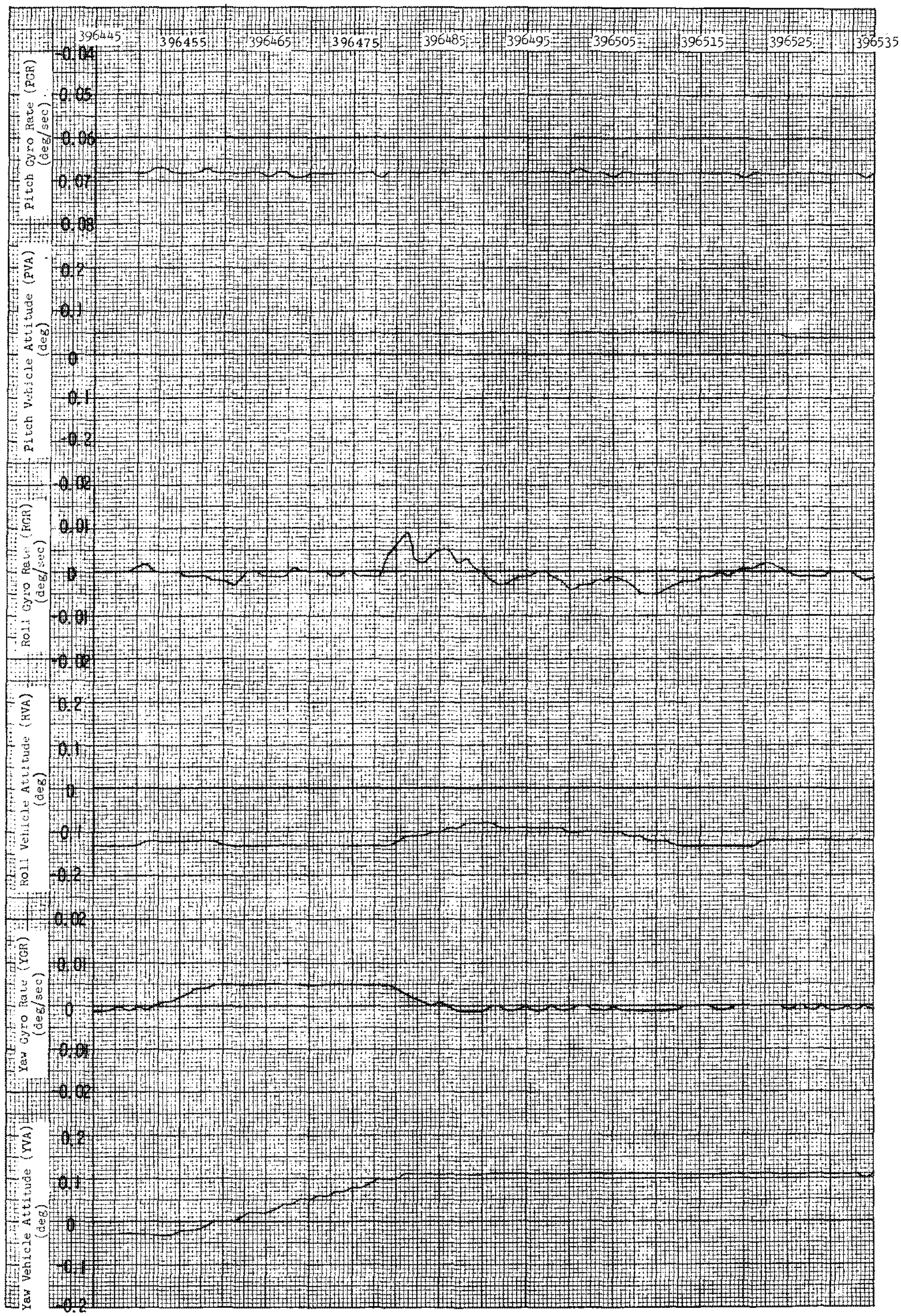

FIGURE 2-1a

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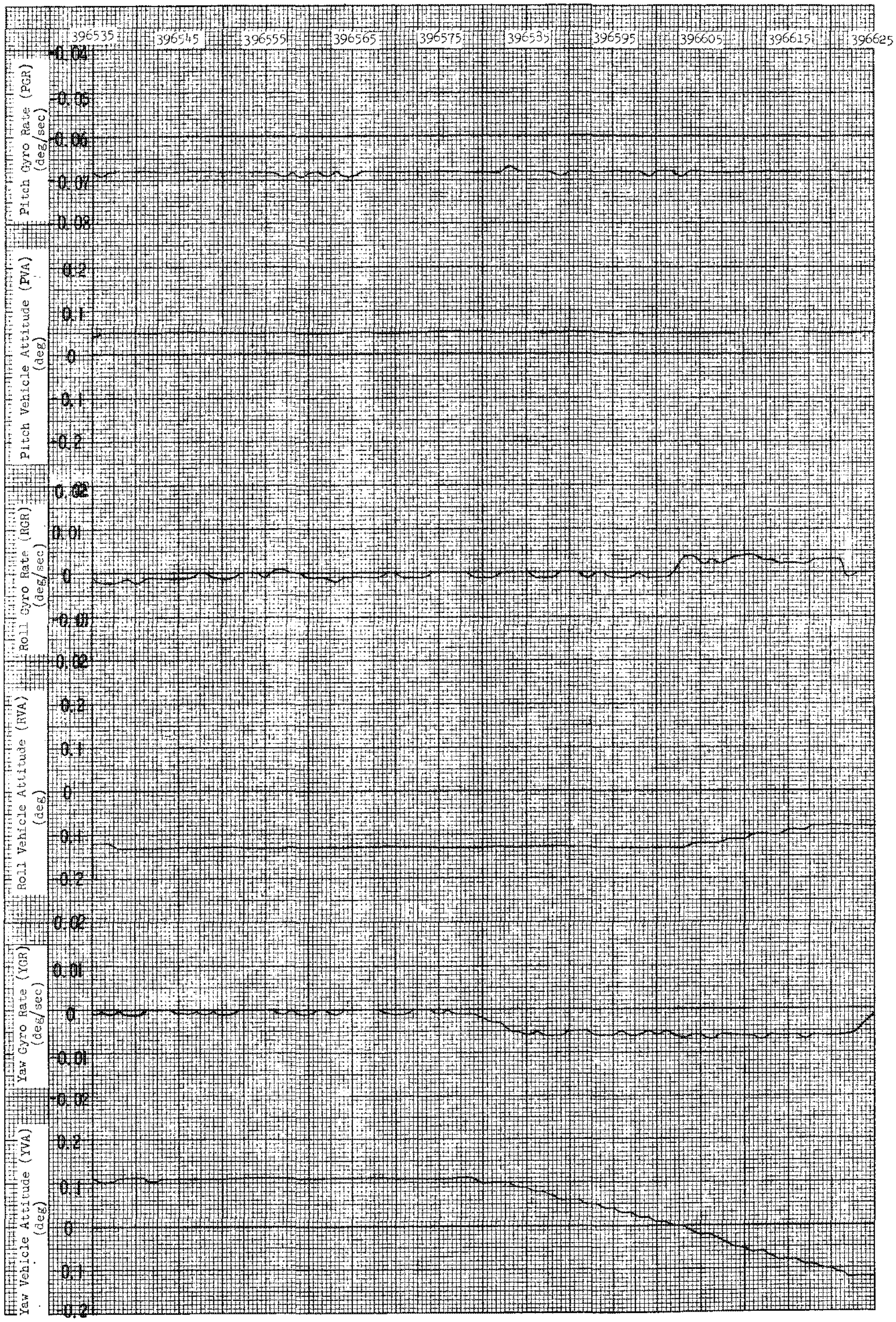

FIGURE 2-1b

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ACS PERFORMANCE FOR PAYLOAD ESD

EVENT TIME 4620.6

Data is provided in Figure 2-2

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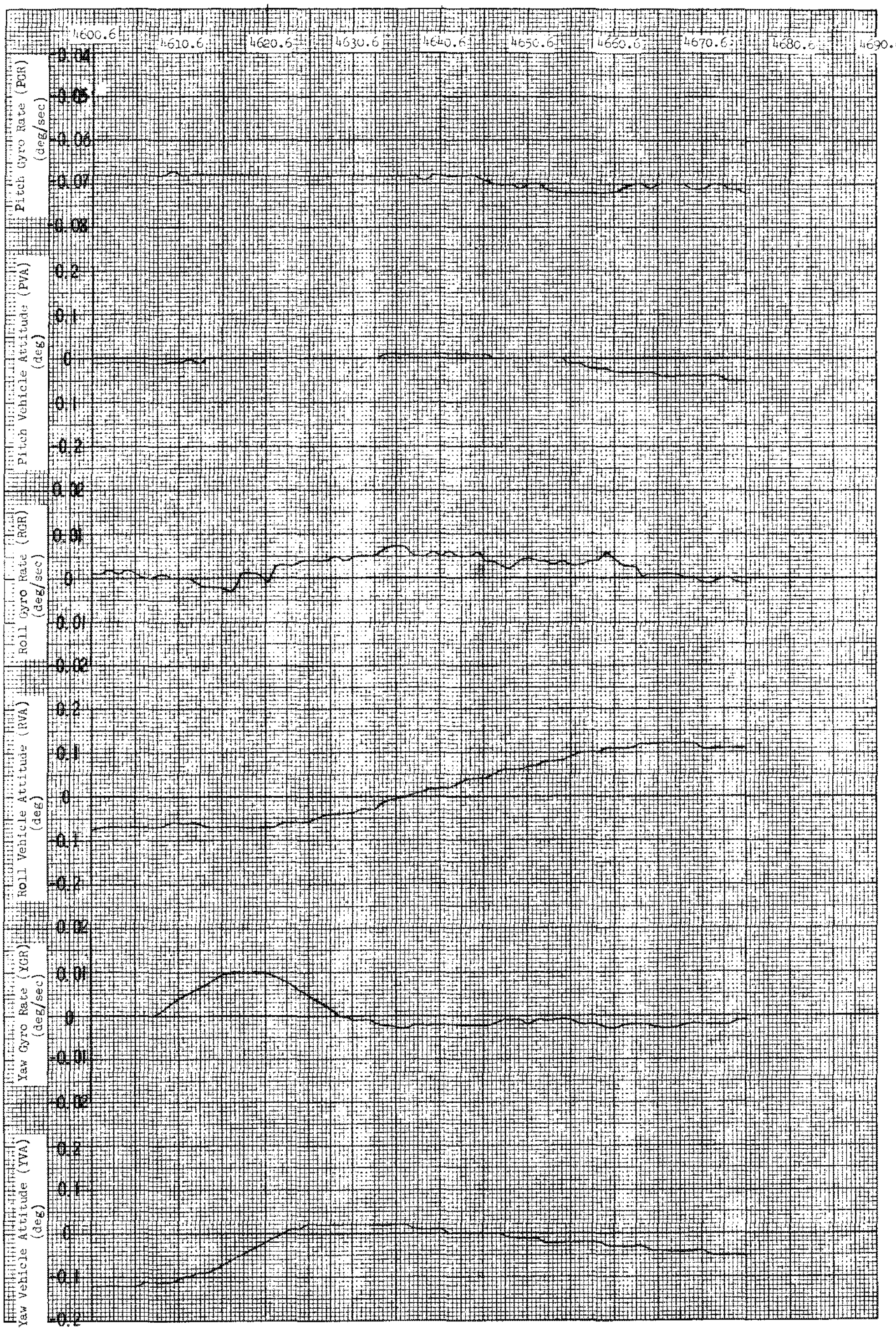

FIGURE 2-2

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ACS PERFORMANCE FOR RV-1 PITCH DOWN AND SEPARATION

EVENT TIMES:

Start Pitch Down	740280.5
Stop Pitch Down	740830.0
RV Separation	746603.5
Start Pitch Up	746604.7
Start Pitch Down	748804.5

Data is provided in Figures 2-3a through 2-3f

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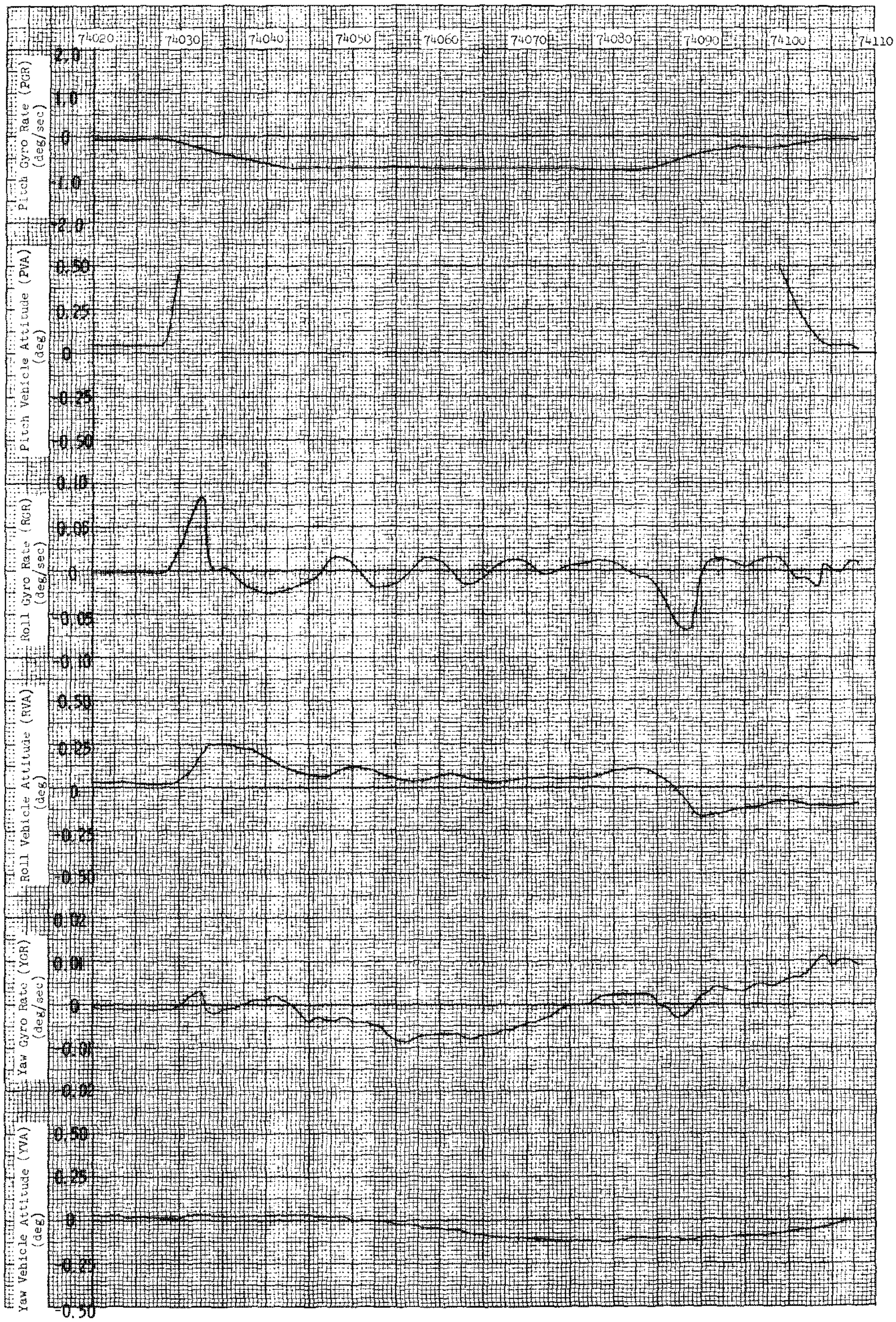

FIGURE 2-3a

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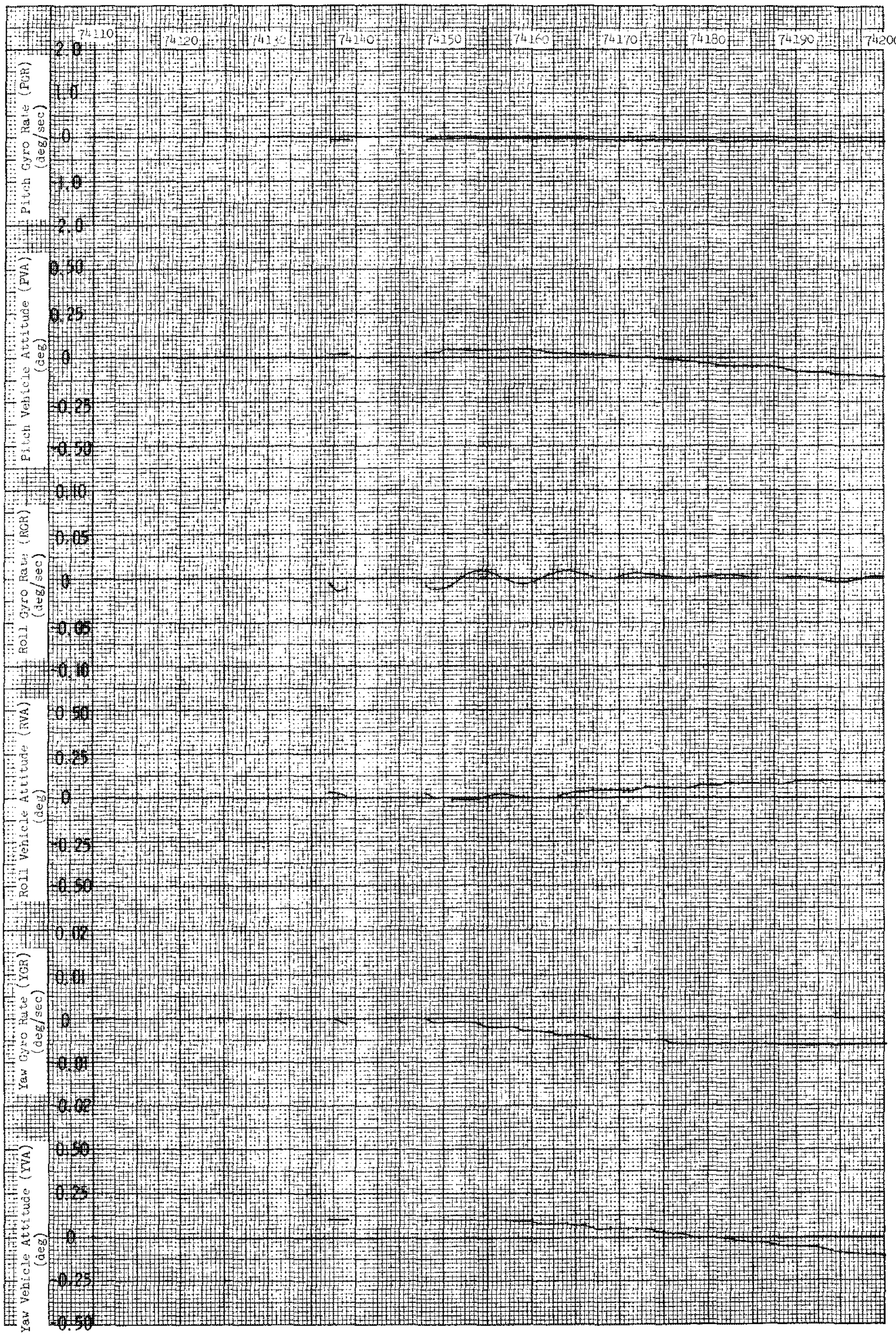

FIGURE 2-3b

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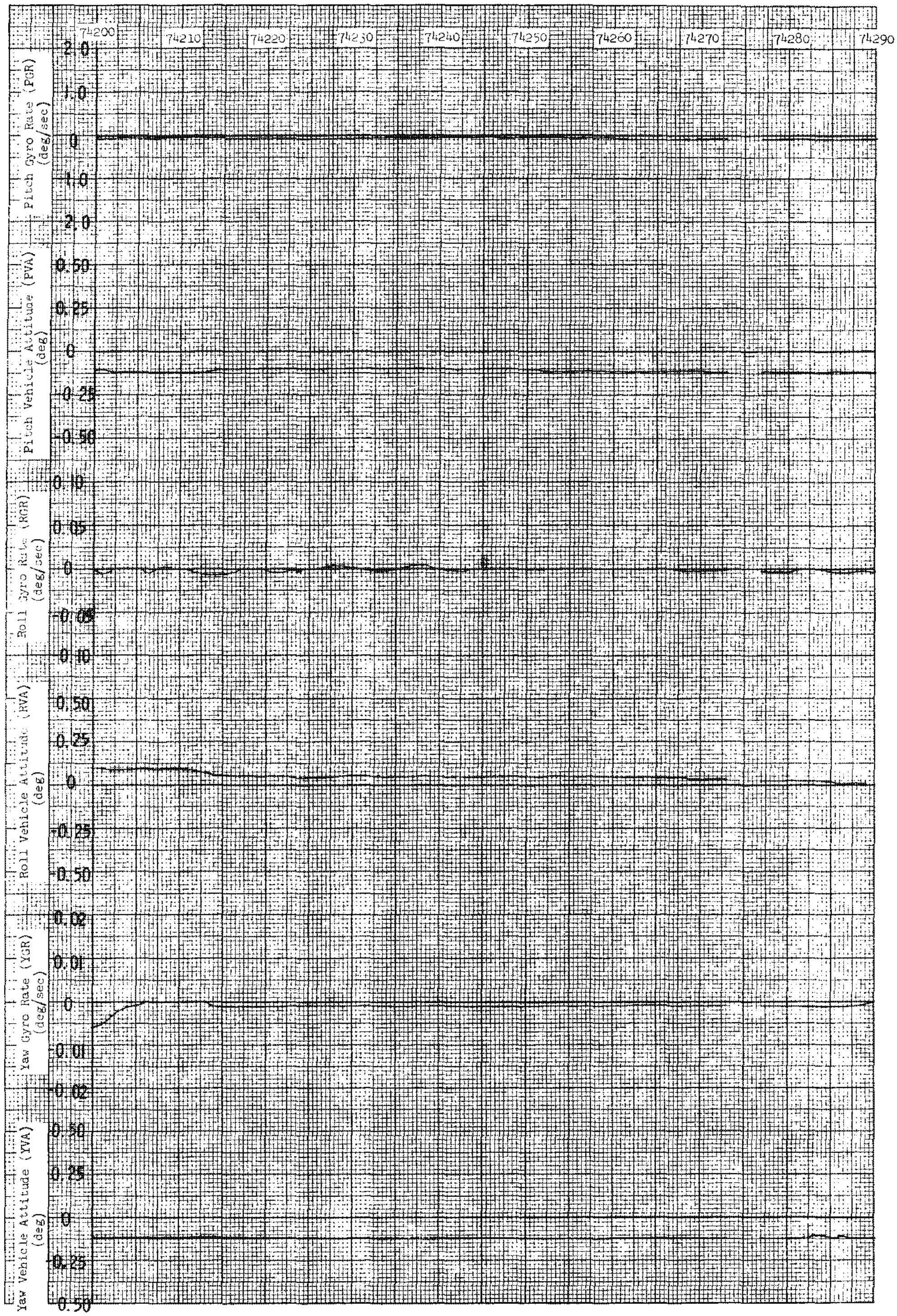

FIGURE 2-3c

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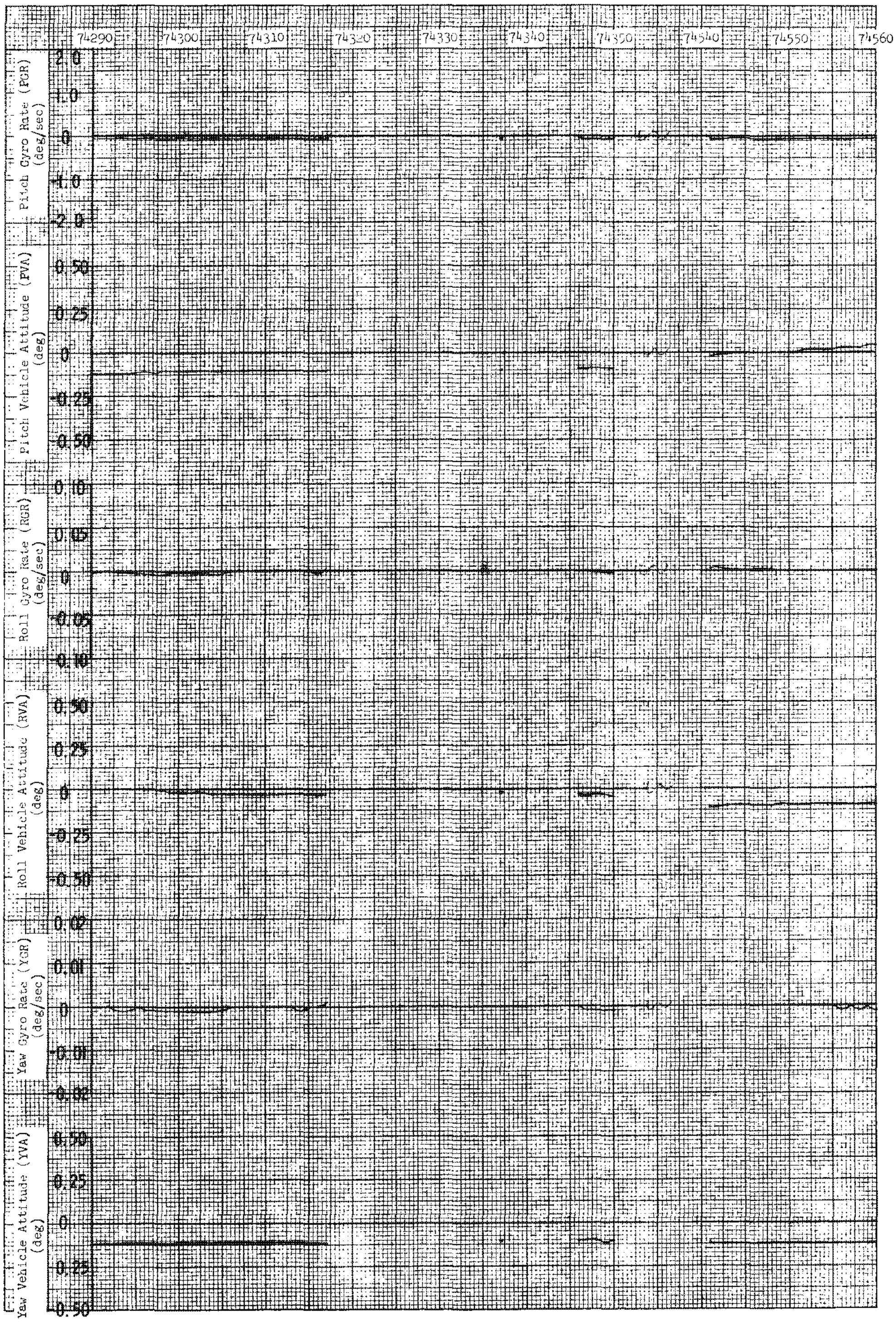

FIGURE 2-3d

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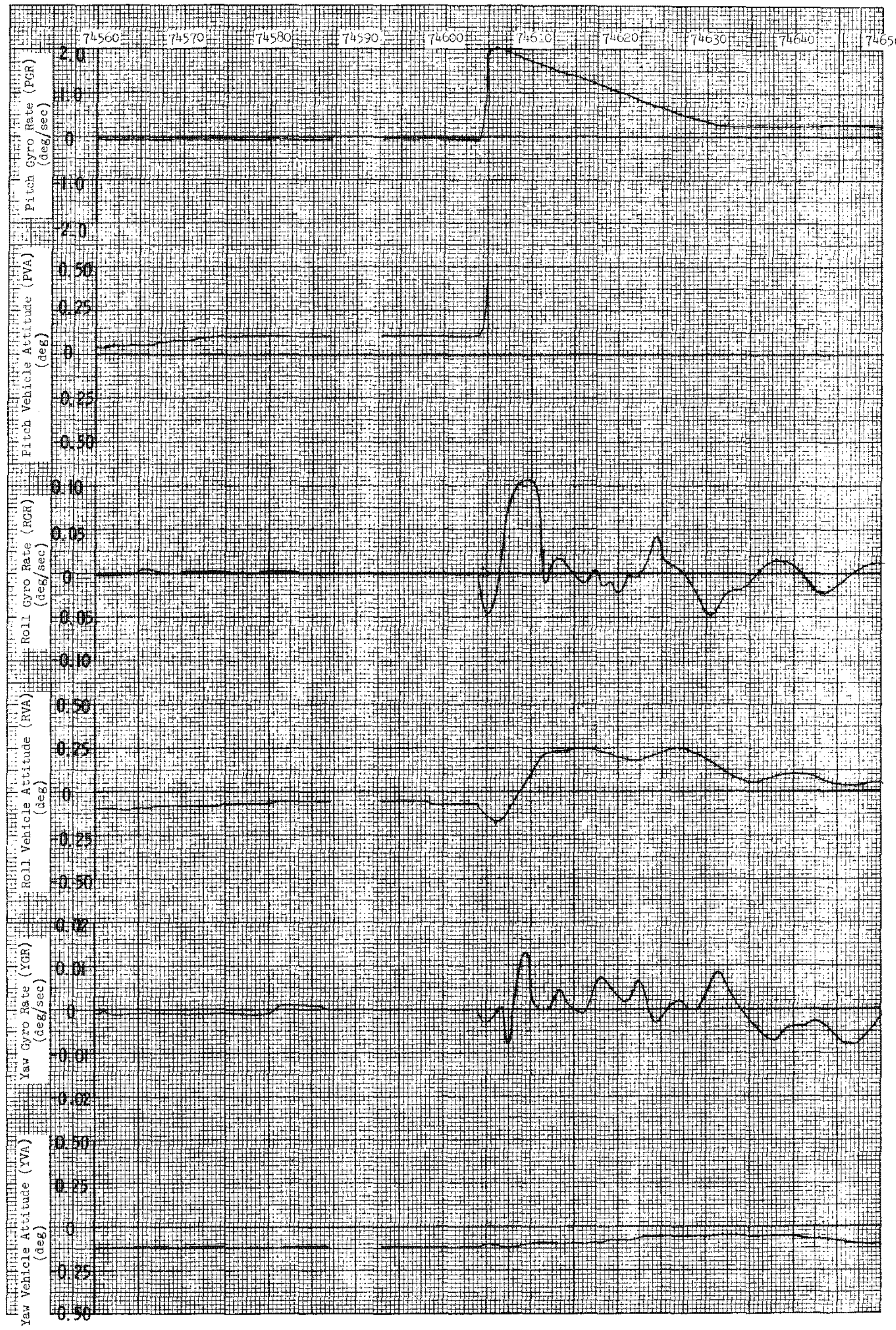

FIGURE 2-3e

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FIGURE 2-3f

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TABLE 2-3

PITCH MANEUVER PERFORMANCE PRECEDING RV SEPARATIONS

	Pitch Down Angle		Maneuvering Time		Pitch Down Coast Rate		
RV/Rev	Desired (deg)	Actual (deg)	Spec (sec)	Actual (sec)	Command (deg/sec)	Expected (deg/sec)	Actual (deg/sec)
1/82.2	-38.8	-38.3	150	76	-.705	-.75 ±.05	-.74
2/179.1	-44.1	-43.6	150	82	-.705	-.75 ±.05	-.74
3/405.2	-45.0	-44.8	150	83	-.705	-.75 ±.05	-.74
4/502.3	-42.0	-44.6	150	78	-.705	-.75 ±.05	-.74
Pitch down test Rev 66.3	-30.0	-29.7	150	60	-.705	-.75 ±.05	-.74
Pitch up test Rev 66.3	+30.0	+30.5	150	54	+.705	+.75 ±.05	+.73

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Table 2-4

SUMMARY OF RV/SV SEPARATION PERFORMANCE

RV/Rev	Peak Pitch Rate (deg/sec)	Maximum Pitch Integrator Angle (deg)	Impulse Induced by RV (lb-sec)	Pitch Valve(2, 3) Thrust Level (lbf)	Pitch Down Prior to Sep (deg)	Pitch up Following RV Sep to Removal of Maneuver Command (deg)	Pitch Inertia (slug-ft^2)	Pitch Moment Arm (ft) (Thruster)	Roll Angle Spec (deg)	Roll Angle Meas H/S (deg)
1/82.3	2.01	13.4	126.5	4.82	-38.3	84.6(1)	105121	15.3	±1.0	-.008
2/179.3	2.0(2)	15.7(2)	112(2)	–	-43.6	90.2	82661	13.9	±1.0	+.04(3)
3/405.3	2.35	21	128	3.67	-44.8	90.5	65990	12.8	±1.0	+.048
4/502.3	2.8	31.3	160	3.2	-44.6	93.4	55975	11.7	±1.0	+.08

(1) Complete data not available

(2) Estimates. Data not available for 17 sec after separation

(3) At POGO fade before separation

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2.1.5 Deboost

The deboost sequence utilized a planned ΔV of 400 ft/sec with no pitch down. Since an excess of orbit adjust propellant was available, no attempt was made to minimize propellant consumption. A minimum risk maximum propellant consumption vehicle reentry mode was selected. In addition, due to the less than maximum utilization of RCS propellant, the pressure in the attitude control system was significantly above the minimum design value. While vehicle stability was maintained down to 56NM during reentry operations, this does not mean that this capability can be attributed to all vehicles.

Preliminary evaluation of the vehicle reentry trajectory indicates that no specific impulse degradation was noted during the long duration deboost operation.

The SV deboost sequence on Rev 839 was as shown in Table 2-5:

TABLE 2-5

DEBOOST SEQUENCE ON REV 839

	System Time (sec)	Veh. Time (sec)	ΔT (sec)
Yaw around	60824.7	292128.4	
OA On	66542.9	297846.6	1003.2
OA Off	67546.2	298849.8	
Unable to maintain positive geocentric pitch rate of 0.0687 deg/sec	66740	299043.7	194
Last data point at COOK RTS	67768	299071.7	28

SV attitude and rate control was lost at an altitude of about 56NM. The last rates and attitude observed were:

Pitch Horizon/Sensor	-0.32 deg	Yaw Gyro Rate	+0.026 deg/sec
Roll Horizon/Sensor	+0.44 deg	Pitch Vehicle Attitude	-0.44 deg
Pitch Gyro Rate	-0.66 deg/sec	Roll Vehicle Attitude	>+0.5 deg
Roll Gyro Rate	+0.030 deg/sec	Yaw Vehicle Attitude	+0.1 deg

With the Aft end forward, the final motion observed was the Aft end pitching up out of control.

2.1.6 Experimental Maneuvers

Yaw maneuver performance was evaluated prior to orbit adjust (OA) burns by performing a -180° yaw to a nose aft attitude and then a -180° yaw return to the nose forward attitude on Rev 240 and 241 with the results shown in Table 2-6. The maneuvering time is the time interval from the yaw rate command to the time the rate returns to a value of less than the specified rate after removal of the

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command. Subsequent yaw maneuvers preceding OA burns were performed satisfactorily.

TABLE 2-6

YAW MANEUVER PERFORMANCE

	Yaw Angle (deg)	Valve Thrust (Lb_f)	Spec Rate/Time (deg/sec) (sec)	Actual Time (sec)	Yaw Coast Rate Cmd (deg/sec)	Yaw Coast Rate Expected (deg/sec)	Yaw Coast Rate Actual (deg/sec)
Yaw Reverse Rev 240	-180	Valve 4&5 3.85	0.15/600	271	-0.705	-0.705 ±0.002	-0.705
Yaw Forward Rev 241	-180	——	0.014/1100	287	-0.705	-0.705 ±0.002	-0.705

Pitch maneuver performance was evaluated prior to RV separation on Rev 66 and this performance is included in Table 2-3.

2.1.7 Attitude Reference Module Alignment Verification

Correlation between predicted target coordinates (derived from flight data) and measured photo coordinates was very good. Photographic coordinates for 63 targets were measured on both the Forward and Aft-looking photography. A standard grid and a 7X loop with a 0.001 foot reticle were used for measuring the target photo coordinates.

The results of these measurements show a minor systematic pitch down error and a positive roll bias which are well within the alignment error budget.

2.1.8 Reaction Control System

2.1.8.1 Control Gas Usage

The RCS propellant consumed over the 31 days (502 Revs) was 134 ±10 lbs compared to an available 390 ±10 lbs with OA transfer as shown in Fig. 2-4. The flight measured consumption was computed from the RCS tank temperature and pressure telemetry data and has an uncertainty of ±10 lbs at beginning and end of mission. Also shown in the figure is the pre-flight worse case prediction which was based on P/L ops per Interface Control Document (ICD) and includes contingencies such as magnetic torques and leakage.

The control gas usage during the six OAS burns and deboost was as indicated in Table 2-7.

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FIGURE 2-4

RCS PROPELLANT CONSUMPTION

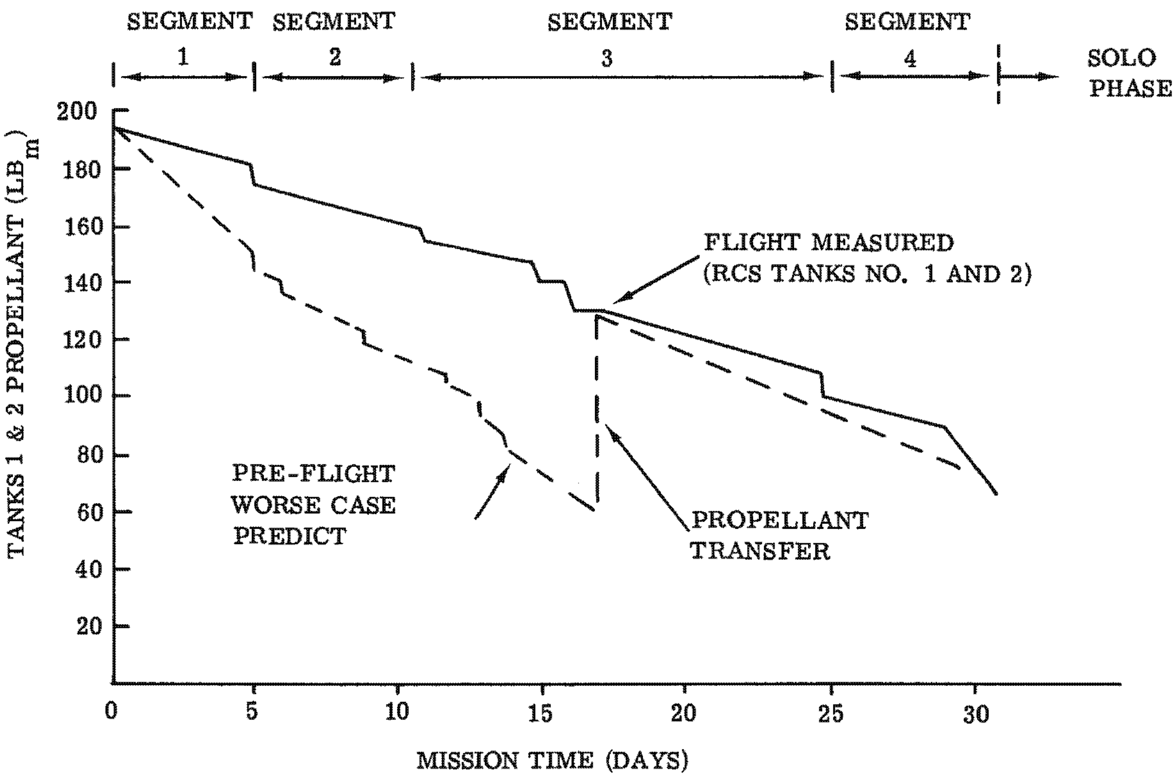

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TABLE 2-7

CONTROL GAS USAGE

	Flight Measured (lb m)	Nominal Predicted (lb m)
6 OAS Burns	6.2	8.7
Deboost	10.7	9.8
Total	16.9	18.5

The results indicate that the OAS induced misalignment was nominal and the impulse prediction and selection of the thrust vector aim point were entirely adequate.

The nominal gas consumption predictions (preflight) for orbit adjusts were over estimated by 29% while deboost estimates (preflight) are 10% less than measured. The 17 pounds of propellant used for propulsion attitude control are small compared to the total used (134±/12.6%). Thus the imbalance payload torque effects, vehicle aerodynamics, and other effects are significantly over estimated in the worse case predictions.

2.1.8.2 RCS Thruster Performance

Three RCS Thrusters showed some sluggishness in pulse shape tailoff at the end of the mission as shown in Fig. 2-5. The degree of tailoff change was minor and much less than observed during ground testing which is shown for comparison. The distortion of the pulse shape suggests some catalyst bed wear, particularly at low duty cycles. Vehicle control was not compromised and no change in propellant consumption could be detected. Pulses with tailoff sluggishness have the same total impulse as pulses with crisp shapes and thus no change in specific impulse is encountered. The small change in response and centroid is minor, not affecting vehicle control.

RCS performance appears to have been within specification limits during the mono ops in Rev 474 when the roll rate exceeded the expected value at the start and stop of the OBs as discussed in Paragraph 2.1.2. RCS firings accurately reflect Attitude Control System (ACS) driver commands and delay times between driver commands and thrust response appear to be normal in all cases.

OB START resulted in normal thrust levels (4.0 lbs) for both Thrusters (No. 3 and 7). On OB STOP, where roll rates were again exceeded, the thrust level for Thruster No. 2 was 4.0 lbs but Thruster No. 6 was 2.8 lbs (minimum allowable: 2.5 lbs). Although No. 6 Thruster was within minimum allowable limits, it appears to have shifted to a lower thrust level than the other three thrusters. Subsequent long firing operation of thruster No. 6, during RV-4 separation, showed an increase in its thrust to a level more closely approaching that of the other thrusters. Further evaluation of thrust level shift and OB-induced roll rates will be performed using the results from the SOLO

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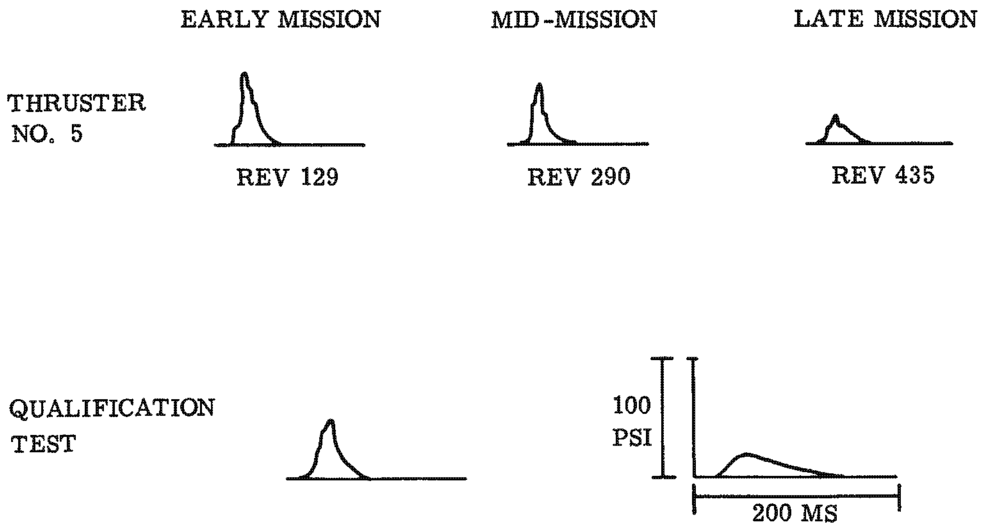
FIGURE 2-5

RCS THRUSTER PERFORMANCE



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experiments.

A series of payload mono operations were run on side A and side B. While the vehicle attitude and rate data for mono operation are acceptable, no analysis has been made of the vehicle attitude control gas consumption. However, with the large propellant margins available for the early series vehicles, no system problem is anticipated.

2.1.9 Problems

2.1.9.1. Horizon Sensor Assembly (HSA) Transient at Separation

A transient occurred on the primary horizon sensor outputs at separation which was not present on the redundant horizon sensor outputs. The roll output exhibited a +4.0 deg transient and the pitch output a +2.5 deg transient, both lasting approximately 5 sec. The probable causes investigated were (1) sun interference, (2) reflection off particles resulting from separation, (3) microphonics resulting from separation shock and (4) an intermittent circuit in the HSA. Sun interference was eliminated because the sun was overhead at the time of separation. Particle reflection was considered unlikely because the redundant horizon sensor did not exhibit transients. The horizon sensor is known to be susceptible to microphonics; however, an intermittent circuit can not be ruled out. This is not considered to be a flight problem since the horizon sensor is not connected until sometime after separation. Block II HSAs will not be as susceptible to microphonics due to redesign of the sensor head.

2.2 ORBIT ADJUST SYSTEM

2.2.1 Orbit Control

The Orbit Adjust System (OAS) was utilized six times during the primary mission for drag makeup and perigee location control. The OA firings were all normal and the engine performance was well within specifications. Pertinent performance factors are summarized in Table 2-8.

TABLE 2-8

OAS PERFORMANCE

OA Firing Number	Rev Number	Impulse Delivered (lb-sec)	Planned ΔV (ft/sec)	Achieved ΔV (ft/sec)	Percent Error in ΔV	Percent Allowable Impulse Error
1	127	9433	15.74	16.27	2.71	54.0
2	190	5457	9.98	10.24	2.58	25.8
3	254	33171	60.8	62.2	2.30	18.2
4	255	22669	-41.9	-42.7	1.91	45.9
5	336	2713	5.05	5.38	6.58	30.0
6	385	11877	22.11	22.29	0.81	20.5

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2.2.2 Deboost

The deboost burn on Rev 839 was engine firing number 37. The performance of the OAS during the 1003 sec burn was nominal and indicated a healthy catalyst bed resistance. Delta velocity was 405 ft/sec and the SV impacted at 11 deg North and 127 deg West.

Approximately 370 pounds of propellant were used for orbit control (85,320 lbs/sec, ΔV = 155.6 ft/sec) and 690 pounds were used for the 400 ft/sec deboost. Thus the total propellant consumption was 1,060 pounds of the available 2,900 pounds.

2.3 TRACKING, TELEMETRY AND COMMAND (TTC)

2.3.1 Tracking

Tracking accuracy was within mission requirements.

2.3.2 Telemetry

2.3.2.1 General Performance

The general performance of the telemetry was nominal. No anomalies occurred with the system during the flight. The following list summarizes the approximate usage through Revolution 839:

	Side 1	Side 2
A. Space Ground Link System (SGLS)		
Number of station contacts	896	116
Operational time (min)	5,600	725
B. Pulse Code Modulation (PCM)		
Operational Time (min)	17,957	25
Mode, Operational Time (min)		
Ascent	10	
Orbit-Engineering	640	
Orbit-Record	10,617	
Orbit-Operational	6,722	
C. Tape Recorder		
Number of Record Operations	3,380	14
Number of Playback Operations	717	14
Operational Time (min)	12,987	300

D. Instrumentation

Three temperature transducers were defective at launch. These were two shroud skin temperatures and one mid section temperature. The sound pressure transducer assemblies employed in the Ascent Telemetry System

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exhibited a slow recovery when the static pressure pulses at ignition and liftoff placed the transducer amplifier in or near saturating range. Changes will be implemented on SV-2 to eliminate this problem.

E. Ascent Telemetry System

With the exception of the sound pressure transducers noted above, all Ascent TM System data was acceptable. The processing of the mid section Frequency Modulated (FM) data through the SGLS 1.7 MHz FM Voltage Control Oscillator (VCO) also performed satisfactorily during the ascent phase of flight.

2.3.2.2 SGLS Performance

All Remote Tracking Stations experienced fluctuations during the mission. These fluctuations ranged from minor changes (5 db) to complete drop-outs. The majority of fluctuations were identified and plotted. The observed SGLS 1 fluctuations were symmetrical (right/left side of SV). Solar array positioning (during SOLO) to other locations resulted in no significant pattern changes. As a result of the SGLS 1 data obtained, a secure block loading constraint was recommended. Also, predictions were made and provided for both block loading and tape recorder playback. These methods of avoiding the drop-outs were followed with no data loss. SGLS 2 was also used during the mission to obtain signal strength data. The data obtained indicated that the fluctuations were not symmetrical (i.e., left side different than right side). Review of the SV hardware and the data obtained in flight indicate that the fluctuations are due to the SV antenna ground plane for both SGLS 1 and 2.

While some drop-outs were expected, the extent of the drop-outs experienced during Mission 1201 were more severe than predicted. Even though mission operations were not impacted due to the work arounds mentioned above, antenna design studies should be continued.

2.3.3 Command

2.3.3.1 Extended Command System (ECS)

The health of the ECS was good throughout the entire operation. There were no equipment failures. The ECS was not subject to out-of-specification temperatures or voltages. There were no power drop-outs, relay driver overloads, or clock status errors experienced.

The ECS responded properly in all modes into which it was commanded. There were a total of 223 messages loaded in the ECS. This resulted in 135,226 Stored Program Commands (SPC) being stored for readout from the Programmable Memory Units (PMU). The PMUs output 77,429 commands for processing by the decoders. The remainder were erased prior to time label matches. There were 16 ground station software anomalies on the initialize command following the selected read of the memory search upper bound at the end of an SPC load. These anomalies were caused by a timing problem in the Remote Tracking Station (RTS) software. Correctors were incorporated into the RTS software during the

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SOLO phase.

The accuracy of the ECS Clock was 2.32 parts in 10^7 (about 1 sec every 50 days). This corresponds to an average frequency offset of 0.237 Hz above the nominal frequency of 1.024×10^6 Hz. The frequency of the clock oscillators changed 0.066 Hz in 50 days. This results in a stability of 0.64 parts in 10^7 over a 50 day period. All of these values are well within system specification.

An anomalous 11EV20640 Variable Stored Program Command (VSPC) scheduled for execution on Rev 492 did not perform its required function. Analysis revealed that the VSPC executed Normal Stored Program Command (NSPC) EN02640 which is the Internal Decoder ON command. The decoder remained on for 214.6 sec until a TT&C OFF was commanded which is wired into the external decoder OFF.

A study of the decoder logic associated with these two unrelated commands revealed a logic design error existed. The logic equation for the EN02640 command is satisfied by the 11EV20640 bit pattern. The EN02640 is an internal command and the decoder inhibits the generation of a relay driver pulse. Since the 11EV20640 satisfied the EN02640 equation, relay driver pulses are inhibited. This produces a loss of 11EV20640 relay driver outputs. The design error also results in the loss of relay driver pulses for 15 other combinations of this 11 bit variable in Decoder A. In addition, 16 combinations of each of the 12 bit variables in Decoder A will not generate relay driver pulses. This brings the total number of commands in Decoder A that will not perform the required function to 48. A complete listing of these commands is provided in Table 2-9. This problem does not exist in Decoder B due to the difference in the Decoder Address Plug wiring.

2.3.3.2 Minimal Command Subsystems (MCS)

The MCS responded correctly to all commanding. The MCS was in the operate mode on Rev 20. The Rev 20 operation was for a health check that performed according to expectations. The short duration of MCS operations did not permit clock frequency or stability calculations to be made. The Remote Decoder was used for each of the four recoveries. The performance of both sides was verified from telemetry to be proper in each case. No commands were issued from the Backup Decoder in this operation.

Decoder in this operation.

2.3.3.3 Uplink Operation

The SGLS command link was used to transmit a total of 223 command messages. No anomalies were experienced with this link. The 375 MHz Receiver was powered during the entire mission. Approximately 20 commands were processed by the Receiver with no anomalies.

2.3.3.4 Data Interface Unit (DIU)

The DIU performed 406 operational cycles without malfunction. No spurious request pulses

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were indicated by the operation counters and the predicted operation count equaled the DIU counter reading.

TABLE 2-9

ILLEGAL VARIABLE STORED PROGRAM COMMANDS

11EV20640	12EV40640	12EV60640
11EV20641	12EV40641	12EV60641
11EV20642	12EV40642	12EV60642
11EV20643	12EV40643	12EV60643
11EV21640	12EV41640	12EV61640
11EV21641	12EV41641	12EV61641
11EV21642	12EV41642	12EV61642
11EV21643	12EV41643	12EV61643
11EV22640	12EV42640	12EV62640
11EV22641	12EV42641	12EV62641
11EV22642	12EV42642	12EV62642
11EV22643	12EV42643	12EV62643
11EV23640	12EV43640	12EV63640
11EV23641	12EV43641	12EV63641
11EV23642	12EV43642	12EV63642
11EV23643	12EV43643	12EV63643

Preliminary information indicates a design change will be incorporated effective with SV-5 and later models. For SV-2 through SV-4 the following operational constraints are scheduled to be followed:

A. 12EV4WXYZ (MCM Assigned): Spare until SV-7

B. 12 EV6WXYZ (TT&T Control): Bit 33 assigned "0"

C. 11EV2WXYZ (V_s): 16 film velocity speeds not available

2.4 ELECTRICAL DISTRIBUTION AND POWER

2.4.1 Solar Arrays

Solar Arrays were extended on Rev 1. Power output from each leg equaled or exceeded the specification value. Degradation for 31 days was approximately 2 percent of initial output. This is within the 5 percent allocated for degradation over 30 days and 1 percent less than engineering predictions.

2.4.2 Main Bus Voltage

Main Bus voltage varied from a low of 26.8V to a high of 31.6V. The allowable range was 25.5V to 33V. The low voltage data was obtained during dark COOK engineering passes with bus loads of 45 to 50 amps. High voltage data was obtained during the charge cycles. Daily voltages are

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summarized in Table 2-10.

2.4.3 Power Capability and Usage

Power usage ranged from 201 to 256 amp-hour/day which was below the available hour/day capability. The quantity of excess amp-hours/day available are tabulated in Table 2-11.

2.4.4 Type 29 Battery Performance

Batteries 3 and 4 operated at an undesirably high temperature (88-100° F) during this flight. The battery thermal switches opened the K2 Solar Array circuits within 12 Revs from launch and K1 circuits in 13 Revs. With the K1s and K2s open the batteries were not being charged and therefore cooled, causing the K1 circuits to close by Rev 17.

TABLE 2-10

VOLTAGE CHART

Day	Maximum Voltage	Minimum Voltage*
1	30.9	28.0
2	30.0	27.6
3	30.0	27.5
4	30.0	27.4
5	30.2	27.8
6	30.2	27.4
7	30.0	27.1
8	30.4	27.4
9	30.3	27.3
10	30.1	27.3
11	30.2	27.2
12	30.0	27.1
13	30.2	26.8
14	30.4	27.5
15	30.5	27.7
16	30.6	27.3
17	30.0	27.8
18	30.8	27.4
19	30.4	27.2
20	29.9	27.5
21	30.3	27.7
22	30.1	27.6

Day	Maximum Voltage	Minimum Voltage
23	30.3	27.3
24	30.3	27.6
25	30.3	27.7
26	31.4	28.0
27	31.2	27.8
28	31.1	27.7
29	31.6	28.6
30	31.2	27.6
31	31.5	27.4

* Minimum voltage obtained during dark engineering pass.

TABLE 2-11

POWER CHART

Day	Amp-Hour Used/Day	Amp-Hour Available/Day*	Amp-Hour Excess/Day
1	226	342	116
2	241	264	25
3	256	264	8
4	245	258	13
5	241	255	14
6	246	251	5
7	247	256	9
8	242	255	13
9	255	253	-2
10	235	250	15
11	239	246	7
12	237	246	9
13	219	242	23
14	235	240	5
15	233	240	7
16	223	245	22
17	228	244	16
18	230	244	14
19	224	244	20

Day	Amp-Hour Used/Day	Amp-Hour Available/Day*	Amp-Hour Excess/Day
20	217	235	18
21	229	235	6
22	217	233	16
23	212	233	21
24	215	231	16
25	216	236	20
26	207	240	35
27	207	236	19
28	208	230	22
29	201	226	25
30	215		
31	232		
Total	**7078 ***7075	7174	537

*6/8 Array **Calculated ***Amp-hour Meter

This heating and cooling cycle continued throughout the remainder of the mission and resulted in limiting the recharge of these batteries to about 35 amp-hr instead of the normal 40.5 amp-hr. The average Solar Array charge rate to these two batteries was therefore reduced by approximately 50 percent, thereby reducing the daily capability. On the other hand, higher temperatures prevailed for other equipment reducing or eliminating the power required for heaters. This offset the power generating loss and provided a positive margin. A detailed discussion of the Aft Section thermal anomaly that produced the over-temperature condition for these batteries is presented in Section V.

Batteries 1 and 2 operated at 44 to 47° F throughout the mission and performed normally. Note that normal K2 opening for these batteries occurred only rarely since the fractional Solar Array output caused by the hot Batteries 3 and 4 was used to supply main bus requirements. This left insufficient charging current to drive Batteries 1 and 2 to K2 cutoff. However, as indicated in paragraph 3 above, the power capability of the system was adequate to support the mission demands.

2.4.4.1 Power Management

The power management technique Lockheed Missile & Space Center (LMSC) attempted to implement on SV-1 was the "bookkeeping" method. With this technique an analyst adds up commendable power usage in ampere-hours and adds it to a base power "estimate." With an initial state-of-charge based on a "K_2 open" event, and an estimated re-charging capability, he predicts the depth of discharge that will result for a given command load or message.

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During the SV-1 operation, the bay 12 batteries overheated and "tripped off" thus changing the system capacity over a range of 4/8 to 7/8 of the total capacity. The Electrical Distribution and Power (EDAP) analyst did not know the energy balance for each battery, i.e., the charge/discharge amp-hour summary. In addition the "K_2" events did not occur in the expected manner, i.e., voltage cutout, therefore the LMSC state-of-charge plots were erroneous and were discontinued early in the mission. To replace the bookkeeping technique a "minimum voltage" technique was adopted. The minimum bus voltage was plotted during payload operations with the expectation of establishing trends toward low battery state-of-charge. The basic weakness of this technique is that it is an inexact "indicator," it only gives one a feel for state-of-charge, but there is no correlation factor between a "low" voltage and the allowable payload operation on the next 6 - 8 revs. The Cook nighttime pass was the best indicator of system state-of-charge, but it has the same problem of low voltage correlation and in addition occurs only once per day.

The bookkeeping method will only work if the starting point can be established, i.e., "K_2" relays controlled by voltage cut off. Any power source anomalies prevent the establishment of this start point, rendering this system useless.

The low voltage technique is considered an emergency expedient due to the limitations cited above. Therefore a better power management system must be developed for future vehicles.

2.4.5 Pyro Battery Performance

The planned duration for RV-4 was 20 days (i.e., day 25 to day 45) however on Rev 440 (day 27) the pyro battery voltage monitor began dropping indicating that the battery was approaching capacity depletion. Since it was expected that similar effects would occur on pyro battery No. 2, it was decided to terminate the active portion of the mission. The predicted life of both pyro batteries was 45 days minimum and was based on wet stand separator life being the limiting factor. The premature depletion of capacity was due to the self discharge rate being higher than allowed for at the 90-95° F temperatures experienced by the batteries (see paragraph 12.3). Under these conditions, self discharge would account for approximately 30 percent of the 8 amp-hr capacity at launch. Normally, self discharge would account for no greater than 10 percent of capacity if batteries had been in the temperature range of 45° F to 50° F. This was verified after the fact by a review of previous laboratory testing data and was further reinforced by Pyro Battery No. 2 following a similar degradation with a lighter load.

From the above data it is evident that the premature battery depletion was due not only to the greater than expected internal dissipation but also to a lack of adequate power management for both preflight operations and flight instrumentation loads. Corrective action, follows two paths:

A. Activate pyro batteries as close to launch as possible. Late activation will depend on the battery's ability to provide surge loads when in the peroxide regions. (This would increase capacity

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available for orbit use by 5 amp-hours if predischarge were to be deleted.) The battery is noted nominally at 8 amp-hours at lift-off.

B. Command significant steady state loads to OFF when not required for data gathering during health checks and pyro operations. (This would increase capacity available for orbit use by 3.5 amp-hr.)

2.5 LIFEBOAT II

2.5.1 Health Checks

2.5.1.1 Rev 13

The Lifeboat II System was enabled for approximately 95 sec and executed for the last 35 sec of the 95 sec. From real-time data, proper phasing of the magnetometer outputs was verified. From post test data, the R and Q magnetometer outputs indicated agreement with expected outputs within 0.5 deg. The P sensor output was at TM saturation. Rate gyro evaluation showed the pitch gyro responding to orbital geocentric rate within 0.01 deg/sec of the primary ACS rate indication. Since vehicle rates were very low in yaw and roll, no quantitative comparison was possible. Proper thrust valve phasing in response to magnetometer outputs was verified during the execute period. Since the regulator valve was closed, the vehicle remained under the control of the primary ACS.

2.5.1.2 Rev 82

The System was enabled for 31 sec. There was no execute mode and reset occurred approximately 3 sec after RV separation. Approximate comparison of the P and Q magnetometer outputs from vehicle tape recorder playback data showed agreement within 2.5 deg with the expected outputs. The R sensor was TM saturated during the time the system was in the SV deboost mode. Rate gyros indicated approximately correct rates although the yaw and roll rates were again very low.

2.5.1.3 Revs 179, 405, and 502

The System was enabled for 31 sec in each case and reset occurred about 2 sec after each RV release. As of this writing, post test data from the tape recorder playback during these tests have not been processed for review so no results can be reported.

Despite the lack of data from the last three health checks, data from the first two verified that the Lifeboat II system performed satisfactorily in all modes (RV recovery, SV deboost, South-to-North and North-to-South).

2.5.2 Usage

The Lifeboat II system was not used for active attitude control during the primary mission. Since the regulator valve was closed during the execute portion of the Rev 13 health check, no cold gas was expended.

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Payloads

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SECTION III

PAYLOADS

3.1 SENSOR SUBSYSTEM

3.1.1 Film Path

The Forward Camera course and fine film paths operated properly throughout the mission. The Aft Camera had three ESD's. One occurred in the course film path on Rev 402. The remaining two occurred in the fine film path on Revs 314 and 445.

On Rev 314 there was an ESD at brake release of the fifth operation. Telemetry data indicated a jam in the fine film drive system. A constant velocity run was made on Rev 322 in real-time during the COOK pass. The constant velocity run was nominal so a six frame engineering sequence was run in real-time on the Rev 323 COOK pass. This sequence was performed satisfactorily and normal payload operations were resumed on Rev 326. Inspection of the film was not possible since this ESD occurred while operating in Takeup 3.

Subsequent to the RV-3 to RV-4 cut and wrap sequence on Rev 402, a constant velocity (CV) test was performed during which the course film path system caused an ESD. At Rev 411 COOK a mini-creep was performed with the tensions returning to their normal level. At Rev 417 BOSS another mini-creep was performed indicating the system was responding normally. At Rev 419 COOK a CV Test was performed without a malfunction. It appears there was an obstruction in the film path between the looper output and takeup. An SV-1 test anomaly exhibited signatures similar to those of the flight anomaly. The cause of the test anomaly was a physical drag on the material caused by epoxy chips wedged between the material and idler roller in the takeup. Examination of the processed film indicated no physical damage that would provide a clue to the exact location and cause of the ESD.

On Rev 445 there was an ESD at brake release of the fourth operation. Telemetry data indicated film not being driven properly through the fine film drive system causing the looper to be driven to the stop on the takeup side which generated the ESD. An abbreviated mini-creep was run in real-time on the Rev 452 COOK pass. The data from this run indicated that, although a small amount of film moved through the fine film drive, the Input Drive Capstan was not rotating. In an attempt to free the suspected jam in the input drive, several sequences were programmed. On the Rev 459 COOK pass, the Optical Bars were run to cycle the platen but this had no effect and the looper remained at the takeup side limit switch. Four more abbreviated mini-creeps were run on Revs 465 and 466 on the POGO and BOSS passes and these also did not alleviate the ESD condition. On the Rev 468 COOK pass a 20 inch/sec constant velocity was programmed for a 5.4 sec duration which transported film through the fine drive system but did not free the drive capstan. On Rev 472 a monoscopic B side run was undertaken and this data indicated that the looper moved from the ESD position and tensions had normalized at the beginning of this

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run. On the Rev 476 COOK pass a stereo engineering sequence was run which showed the B Side Input Drive Capstan driving properly. Normal stereo payload operations were resumed on Rev 477. Detailed analysis of the processed film indicated no signs of physical damage on the material. There were no tears, foldovers, dimples, scuffs, embossing or scratches to indicate a mistracking or hangup that would cause an ESD due to the film web itself.

3.1.2 Command and Control

Sensor system performance with respect to the command and control subsystem was nominal with one exception that caused the fourth emergency shutdown on both cameras during Rev 492. The system was commanded to a 120 deg scan angle configuration from a 60 degree scan angle. Due to a logic design error in the Extended Command System the film velocity (Vs) command necessary for the 120 deg scan angle was decoded incorrectly as a Decoder A ON command, and the coarse film speed was not changed from the previous operation. The resultant incompatibility between the coarse and fine film speeds was too large for looper compensation, and the shutdown occurred when the looper was driven against the stops. In its present configuration, Decoder A will not allow the commands 11EV20640, 11EV20641, 11EV20642, and 11EV20643, which correspond to command coarse film speeds of 53.5190 ips, 53.6475 ips, 53.7760 ips, and 53.9045 ips respectively, to be executed. The illegal Vs commands noted will be sent only during an operation which requires a scan angle of 120 degrees and a Vx/h of 0.0417 rad/sec (orbit altitude of approximately 98 miles). Until this problem can be eliminated from the command system, several work-around procedures could be utilized. These are:

A. Software can be made to set a malfunction flag in the command message to identify the problem. Upon recognition of the flag, the message could be manually altered in either of two ways.

1. Switch to Subsystem Command and Control (SCC)2 which would not exhibit the Vs problem because Decoder B would be used to execute the commands.

2. Change Vs command to a legal value as close as possible to the correct value. The new value would, at most, be two steps away from the correct value. This variation would not significantly degrade the photography and would not cause a system shutdown.

B. Since this problem is peculiar to Decoder A, consideration is also being given to using Decoder B as a primary system.

Additional discussion of the ECS logic anomaly is presented in Section II.

3.1.3 Optical Bar Assembly

Both Optical Bar assemblies performed properly throughout the mission. The only point of significance noted during the operation of the OBs was the slightly higher torque required from the OB servo motors at higher film chute pressures. This did not affect performance in any way and also was observed in prolonged runs during ground test prior to flight.

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3.1.4 Instrumentation

All instrumentation points operated correctly throughout the flight except P552 (Metering Rod 3B Drive End Temperature) which provided erratic readings. Other temperature monitors in the vicinity supplied adequate data for thermodynamic analysis of the system. The instrumentation system provided ample data for evaluation of normal sensor system operation. Additional data was needed to adequately analyze the ESD which occurred at Rev 445. For this situation, addition of input and output drive capstan tachometers and summed errors to Telemetry Format C would have presented a more complete record of events leading to the shutdown. However, the available data was adequate to isolate and diagnose the problem, consequently no major changes to the instrumentation are anticipated at this time.

[redacted]

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

3.3 SUBSATELLITE

There were no Subsatellites flown on SV-1.

3.4 STELLAR TERRAIN SUBSYSTEM

There was no Stellar Terrain Subsystem flown on SV-1.

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Reentry Vehicles

PERFORMANCE EVALUATION TEAM
REPORT NO. 1201/71

SECTION IV

REENTRY VEHICLES

4.1 SUMMARY

The four Reentry Vehicle (RV) recoveries were attempted after on-orbit times of 5, 11, 25, and 31 days. RVs 1, 2, and 4 were successful and the payloads transported undamaged. Aerial recovery was made of RVs 2 and 4. RV-1 was recovered by the Surface Recovery Unit because the recovery parachute did not present an acceptable target for aerial retrieval. RV-3 was not sighted nor recovered, and it is concluded that the main parachute was damaged extensively. It is also concluded that the capsule impacted the water at a velocity in excess of 400 fps at a location very near the predicted impact point (PIP).

Data indicate that all on-orbit, separation, reentry, and recovery events occurred as planned, and the RVs followed predicted trajectories. All subsystems performed satisfactorily and met the entry requirements from the 100 x 165 NM orbit, with the exception of the main parachute deployment of RV-3.

4.2 REENTRY VEHICLE PERFORMANCE

The following performance statements apply to all four RVs. The Satellite Vehicle (SV) provided a satisfactory pitch angle for each RV separation. All other SV/RV interface functions were satisfactory. All RV on-orbit functions were normal and occurred on time. A summary of the performance of the four reentry vehicles is given in Table 4-1. The payload weights in Table 4-1 are measured weights for the recovered capsules. The payload was essentially balanced for each flight. The impact locations reported by the recovery message, and not corrected for wind drift, were very close to the predicted impact point determined by computer program at the Satellite Test Center (STC).

Stability margins during the retrograde and exoatmospheric coast phase were high for each flight. Data obtained from onboard instrumentation show body transverse rates were less than 5 deg/sec, which is within the 1σ calculated values for balanced payload conditions. However, the data also show that the spin rate and residual spin rate were lower than predicted.

The entry (400,000 ft altitude) velocity and flight path angle for each recovery are shown in Figure 4-1 compared to envelopes of possible entry conditions for which the reentry vehicle was designed to be compatible. During the early period of the entry phase, the residual roll rate, which had been nearly constant during the exoatmospheric coast phase, began to decrease. After approximately 110 sec, the roll direction reversed. This reversal was typical for all four recoveries and had no measurable affect on the capability of the RVs to enter successfully. Angle of attack was less than the predicted values throughout the significant heating portion of the entry due to roll reversal and associated phenomena. Velocity and altitude time histories apparently correlated well with predicted normal values as evidenced by the accuracy of the impact.

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TABLE 4-1

SV-1 RV RECOVERY SUMMARY

	Recovery 1	Recovery 2	Recovery 3	Recovery 4
RV Serial No.	8	7	5	6
Recovery Rev No.	82	179	405	502
Recovery Date (1971)	20 June	26 June	10 July	16 July
Payload Weight (lb)	319.0	418.0	419.5*	204.1**
Unbalance (percent)	1.2	1.5	0.22	2.3
SV Pitch Angle (deg)	-39	-44	-45	-42
Nominal PIP Latitude	17.5 N	25.0 N	25.0 N	27.0 N
Impact Location Error (BFE vs Test Report TWX)				
Overshoot (nm)	8.4	10.8	6.6	2.4
Cross-track (nm)	3.6 W	7.2 W	13.8 E	9.0 W
Recovery				
Altitude (ft)	Water 0	Aerial 9400	None -	Aerial 12000
Parachute Condition	Severe Damage No Target Cone	Severe Damage	Failure	No Damage**
RV/Payload Condition	Good	Good	Not Recovered	Good

*Value taken from on-orbit telemetry
**RV-4 returned with approximately 45 percent film capacity

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FIGURE 4-1

REENTRY PARAMETER COMPARISON

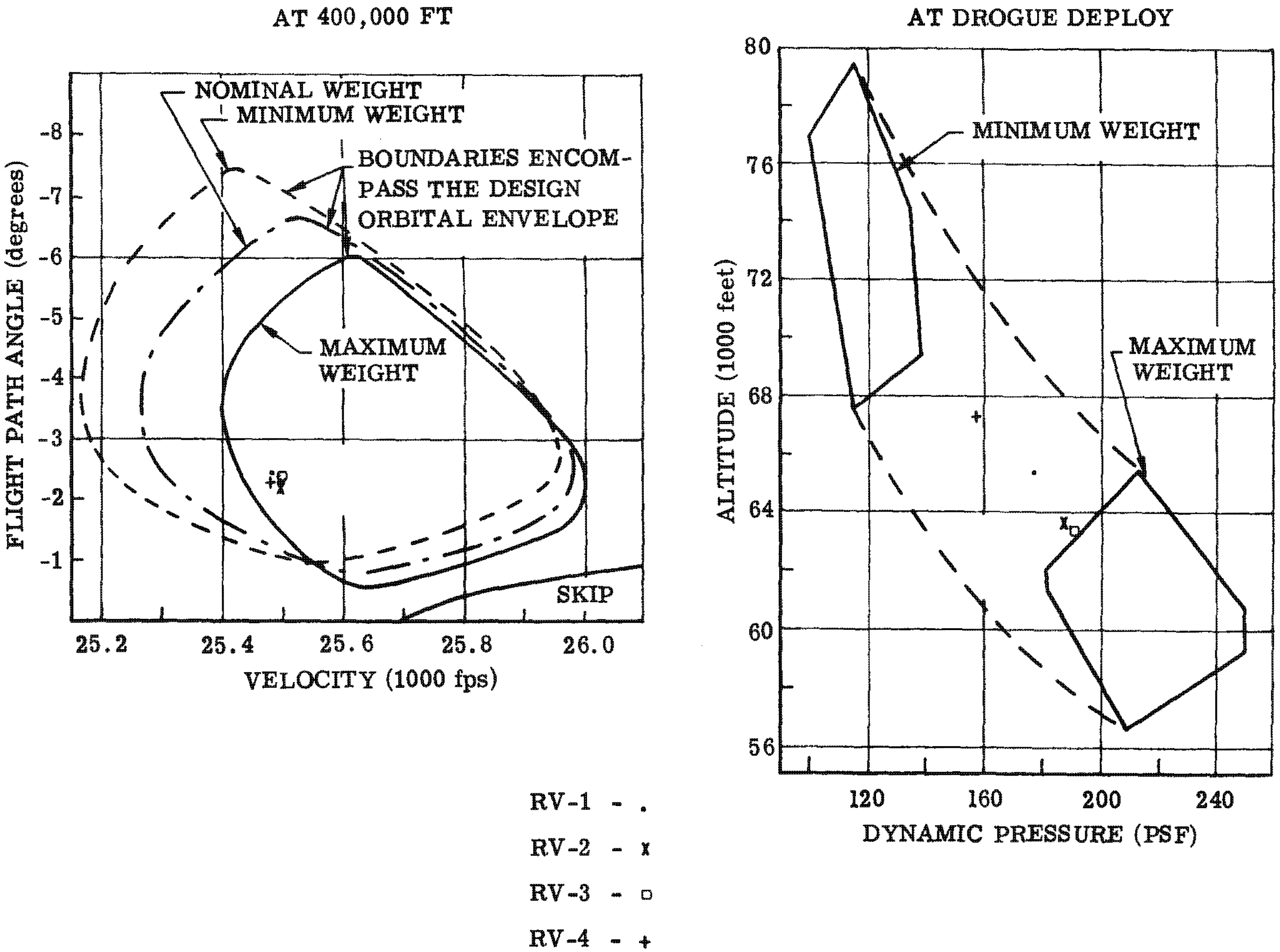

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Figure 4-1 also shows the altitude and dynamic pressure for each recovery at the time of drogue parachute deployment with respect to the design envelope. Flight instrumentation data show that drogue performance as a decelerator was essentially as predicted. For example, the maximum dynamic pressure at drogue release and main parachute deployment was 58.7 psf observed on Recovery 2, and the predicted value was 60.2 psf.

RV-4 performed nominally throughout entry and retrieval. Recovery Vehicles 1, 2, and 4 were sighted while descending on the main parachute. Each was reported to have a descent rate satisfactory for aerial retrieval in spite of the damage to the main parachute. RVs 2 and 4 were retrieved in the air. RV-1 survived water impact with no damage or leakage. RV-3 entry was normal until main parachute deployment. Telemetry data shows that abnormally high axial 'g' levels were induced indicating a premature reefed-open stage of the main parachute. It is concluded that the parachute was extensively damaged and that RV-3 impacted the water at a velocity in excess of 400 fps.

4.3 REENTRY VEHICLE SUBSYSTEM PERFORMANCE

The performance of the Reentry Vehicle subsystems is summarized in Table 4-2. Review of the flight data and post flight inspection and test of the recovered vehicles shows six areas of anomalous conditions as follows:

- A. Major damage in main parachute.
- B. Heat shield bond line temperatures higher than predicted.
- C. Spin and residual spin rate lower than predicted.
- D. Spin reversal during the early period of atmospheric entry.
- E. Base heat protection door not closed to latched position.
- F. Significant heat shield material loss in high density area.

For the four recoveries, only the main parachute demonstrated performance which compromised mission success and resulted in the loss of RV-3. The other five anomalous conditions did not compromise the performance of these four segments. However, they are exceptions to the design and performance criteria established to insure compatibility with entries from the most critical true anomalies for orbits up to 70 x 404 NM and for unbalanced payload conditions. The RV contractor has initiated analysis and test effort to resolve causes for these anomalous conditions and to determine needed changes.

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TABLE 4-2

RV SUBSYSTEM PERFORMANCE SUMMARY

RV Subsystem/Function	Performance Assessment
On-Orbit Thermal Protection	Normal $T_{P/L}$ Container = T_{REF} $^{+0°F}_{-5°F}$ Power Usage (Watts/RV) Max = 12.6 (first day in orbit) Stabilized = 3.5 Allowable = 20.0
Trim and Seal	Normal
Electrical Power and Distribution	Normal All Batteries Activated. All Voltages 25.5 volts.
Sequential Subsystem	Normal Both redundant systems of recovered RVs 1, 2, and 4 were verified to have functioned properly by telemetered data and factory test. RV-3 event sequencing verified to have functioned properly by telemetered data.
Pyro Subsystems	Normal All primary and redundant pyrotechnics in each recovered vehicle (RVs 1, 2, and 4) were verified by factory inspection to have functioned properly.
Spin Stabilization	Spin motor function - normal. Spin rate during retro 0.5 rad/sec below nominal. Spin residual rate 0.6 rad/sec below nominal.
Retro Motor	Normal
Tracking, Telemetry, Instrumentation	Normal
Heat Shield	Bondline temperatures higher than predicted. Adequate protection RVs 1 through 4. Recovered HS from RV 2 shows more recession than predicted.

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TABLE 4-2 (Continued)

RV Subsystem/Function	Performance Assessment
Base Thermal Protection	Adequate thermal protection -- No evidence of backside temperature in excess of 100° F. One door did not latch on Flights 1 and 4. Hinge cover door missing on Flight 2.
Structure	Normal
Recovery System	Drogue performance normal. Main parachute anomalies. Recovery 1 -- major canopy and target cone damage. Recovery 2 -- major canopy damage with moderate target cone damage. Recovery 3 -- major damage -- capsule not recovered. Recovery 4 -- no damage in canopy or target cone.

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Thermal Control

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PERFORMANCE EVALUATION TEAM
REPORT NO. 1201/71

SECTION V

THERMAL CONTROL

5.1 FORWARD AND MID-SECTION

The Forward and Mid-Section structural temperature control is summarized in Table 5-1. The data indicates that the Forward and Mid-Section thermal designs are adequate and no changes are forthcoming as a result of SV-1 experience.

TABLE 5-1

FORWARD AND MID-SECTION THERMAL TEMPERATURES

Parameter	Design Limits (°F)	SV-1 Actuals (°F)
T_{FWD}	47/93	72/77
$T_{FWD} - T_3$	$\leqq 6$	4/6
$\lvert T_{FWD} - T_{TCA} \rvert$	$\leqq 20$	5/7
T_{TCA}	49/91	67/70
T_{SU}	47/93	71/74
$T_{SU} - T_{TCA}$	-4/5	4

where

T_{FWD} = Orbit average temperature of the active RV bays derived from an average of the bulkhead temperatures for each bay.

T_3 = Orbit average temperature of the upper pylon structure in the active bays.

T_{TCA} = Orbit average temperature of the forward compartment of the mid section derived from an average of several internal structural temperatures.

T_{SU} = Orbit average temperature of the aft supply compartment of the mid section derived from an average of several internal structural temperatures.

5.2 ACTIVE THERMAL CONTROL

The Active Thermal Control System performed normally throughout the primary mission. The redundant system was not used. T_{REF}, which represents an average Mid-Section film path temperature, varied between 67°and 71°F during the mission. The RV control zone temperatures tracked T_{REF} within 1°F throughout the mission. The actual flight heater power requirement listed in Figure 5-1 shows an unexplained downward trend with time. There is some concern that the low power consumption may indicate insufficient ability to reject heat should T_{REF} be at the lower end of its range.

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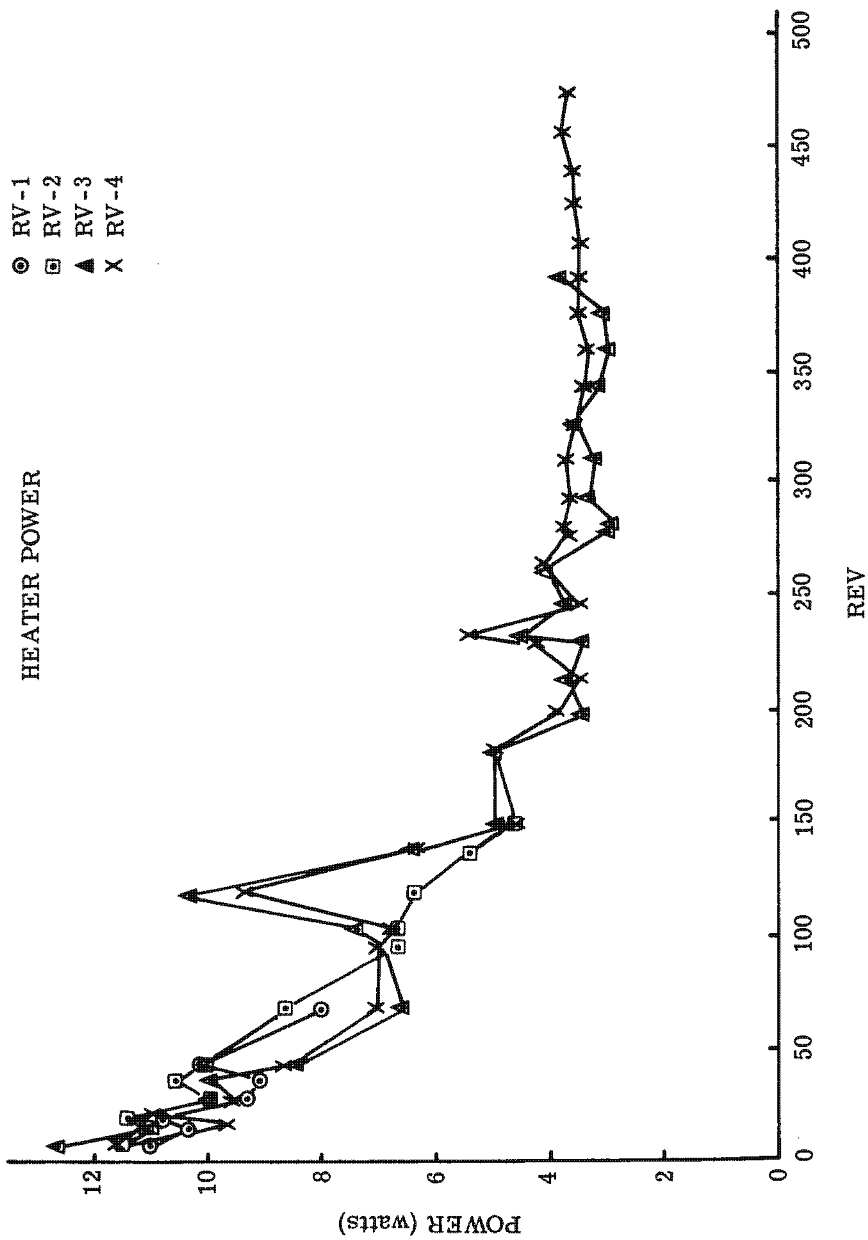

FIGURE 5-1

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5.3 AFT SECTION

Acceptable Aft Section temperature control was achieved with all equipment within design temperature limits except for an over-temperature condition for the Bay 12 Battery Module. The orbital beta angle for this vehicle was +20 deg and nearly constant for the entire 31 day mission. A summary of critical component temperatures is shown in Table 5-2. The Battery Charge Control System protected the Bay 12 Batteries from extreme over-temperatures and allowed for acceptable electrical system performance throughout the mission.

The temperature level of the Aft Section equipment versus predictions is shown in Table 5-2. Temperatures of areas of the skin which received direct solar energy impingement had orbit average temperatures as much as 55° F above predictions. The cause for this anomaly is believed to be contamination of the external thermal control surfaces of the Aft Section occurring during the ascent events (see paragraph 5. 4). The net result of this contamination was an increase in solar absorptivity of the white paint and bare aluminum external thermal control surfaces. The thermal design of the Aft Section is such that most equipment near these hot skins is also thermally coupled to the colder areas of the Aft Section, and therefore was not significantly hotter than predicted. However, the active temperature control scheme of the Bay 12 Battery Module is such that the batteries are directly coupled to the skin. The Bay 12 skin temperatures running 55° F hot tended to make the batteries run nearly 55° F hot. The problem was further aggravated by reduced battery electrical efficiencies at the higher temperatures which also increase battery temperatures.

Built into the electrical charge system design are relays which remove electrical charge from the batteries if certain temperatures are exceeded. This design worked and resulted in removing all charge from the batteries at approximately 98° F. Shortly after this event occurred, the batteries would begin to cool and would continue to cool until the charge was again applied at a lower temperature level. This scheme resulted in Bay 12 Battery temperatures which cycled between 88° F and 100° F, and allowed for acceptable electrical performance. (See additional discussion in Section 2. 4).

5.4 AFT SECTION TEMPERATURE CONTROL ANOMALY

5.4.1 Causes

The possible causes for the anomalous Aft Section temperature levels are:

- Item 1 Unexpected behavior of corrugated Aft Section surfaces in direct sunlight (effective solar absorptivity higher than predicted).
- Item 2 Improper application or preflight damage to Aft Section thermal control surfaces.
- Item 3 Basic thermal design error that was not detected during thermal-vacuum acceptance testing.
- Item 4 Contamination of external Aft Section surfaces by ascent events.

TABLE 5-2

AFT SECTION CRITICAL COMPONENT TEMPERATURES

Critical Component	Design Limits (°F)	SV-1 Actuals** (°F)	Nominal Predictions
	EDAP		
PDJB	-30/170	67/70	68/68
CCCs	-30/170	85/97	65/78
RPS Bay 3	30/110	58/68	59/59
RPS Bay 5	30/110	75/82	70/70
Batteries Bay 12	35/70	88/100	46/50
Batteries Bay 1	35/70	44/47	46/50
PDAs	-30/160	70/95	43/99
Solar Arrays	-125/225	-74/150	-75/130
	ACS		
IRA	60/130	109/111	102/102
HSA Heads	0/130	66/80	74/75
FCEA	-30/160	103/106	88/88
	OAS		
Tank	70/100	92/96	84/84
Quad Valve	35/200	114/118*	118/118
Catalyst Bed	-	129/134*	94/100
	T&T		
Tape Recorders	20/130	88/102	80/84
Transmitters	-30/170	88/115	89/101
PCM Master	-30/170	96/122	77/81
PCM Remote Bay 2	-30/170	61/69	58/62
PCM Remote Bay 10	-30/170	104/110	85/89
	COMMAND		
PMU A	-40/145	101/103	91/93
PMU B	-40/145	109/111	104/106
Clock	-40/153	111/113	111/113
MCS	-40/149	95/99	78/78
	RCS		
Tanks	40/140	77/98	78/78
REM Valves	$\geqq$45	100/158	95/105
Plumbing Bay 12	35/140	91/98	75/82

*Data with orbit adjust engine not firing.
**Stabilized orbital operation (most equipment 70 to 90°F at lift-off).

Item 1 was investigated by conducting a special test of actual corrugated Aft Section panels in a simulated solar flux (parallel light) environment. The results of this test confirmed the validity of the basic design values used for the effective solar absorptivity of the corrugated panels.

Investigation of Item 2 included a review of (1) the actual solar absorptivity and emisivity measurements made before SV-1 was shipped; (2) all quality assurance checks, (3) cleaning processes, and (4) white paint batch data. These reviews failed to reveal any evidence of improper application or preflight damage of Aft Section thermal control surfaces.

A complete review of the Aft Section thermal math model and the results of the Satellite Development Vehicle (SDV-3) and SV-1 thermal vacuum tests was made to investigate Item 3. The result of this review was that the thermal math model and confirmation thereof by the thermal-vacuum tests is valid.

Contamination of Aft Section surfaces during ascent (Item 4) is the apparent cause of the anomalous Aft Section temperature levels. Ascent events that could contribute to contamination include the liftoff ground cloud, Solid Rocket Motor (SRM) staging, Stage I/II separation, RV separation, Stage II retro, and Shroud separation. Although it is impossible to distinguish between these possible sources, review of available ascent temperature data, review of movies of SV-1 and other vehicles during lift-off and ascent, discussions with other program offices, and review of prior analyses of these contamination sources has resulted in the conclusion that most of the contamination occurred during one or more of the following: liftoff, SRM staging, or shroud separation.

Analyses of the magnitude of contamination have been made for Bays 1, 6, 8, and 12. In Figure 5-2, a comparison of analytical model predictions and flight data indicate a solar energy absorption (α_S/ϵ) of 0.43/0.90 for Bay 6. In addition, a comparison of analytical versus flight data for Bay 8 (Figure 5-3) shows good correlation to Bay 6 results with a α_S/ϵ of 0.60/0.90. From the fly-reverse experiment, the data correlated with a value of solar absorptivity of 0.75 for the white paint. A comparison of flight data and analytical results with respect to the absorptivity is shown in Figure 5-4 and Figure 5-5 for Bays 1 and 2 respectively. The expected characteristic of the white paint is $\alpha_S = .18$, thus significant increases were noted over the entire Aft section.

In addition, analysis of the horizon sensor fairing indicated it was contaminated. This fairing is protected by an ejectable cover. Therefore, a large fraction of the previously estimated surface contamination for the uncovered equipment doors must have occurred after SRM separation.

Significant on-orbit temperature changes due to contamination from reentry vehicle separations, attitude control engine operation, orbit adjust engine use, or ultra-violet degradation of the thermal control surfaces were not noted.

5.4.2 Action for SV-2

Since all Aft Section components other than the Bay 12 Batteries remained within their respective

HORIZON SENSOR FAIRING
(Temperature History)

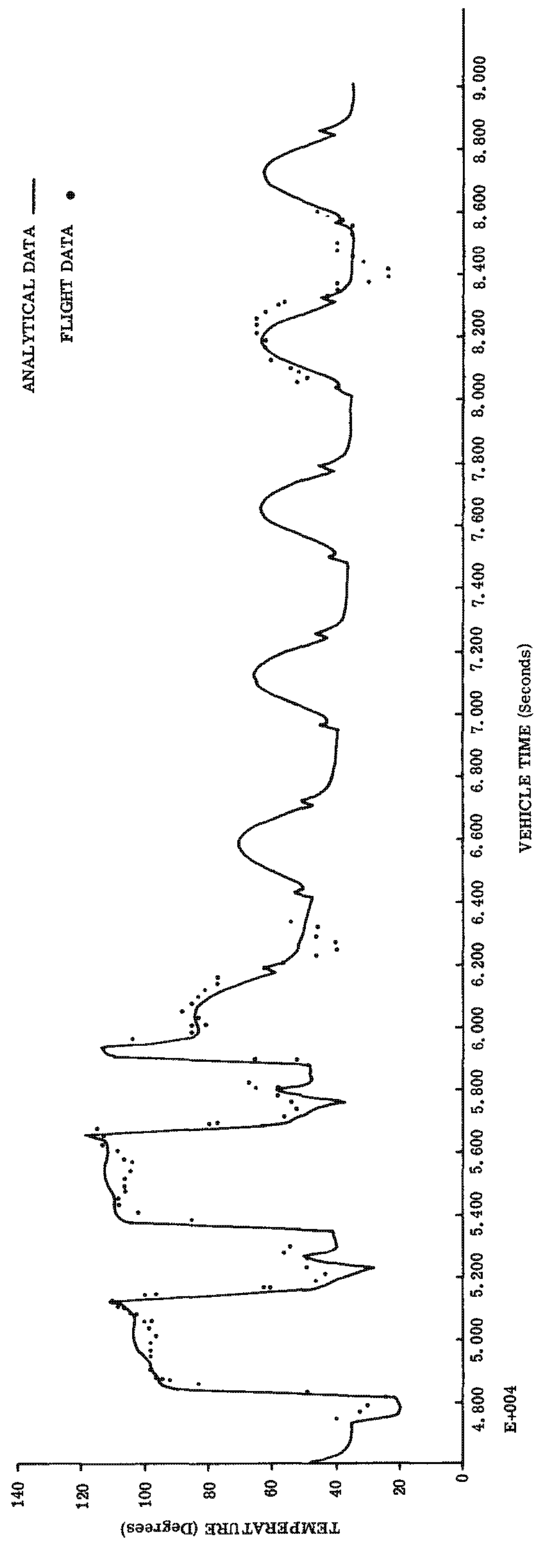

FIGURE 5-2

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FIGURE 5-3

BAY 8 DOOR TEMPERATURE HISTORY

(Fly Reverse Experiment)

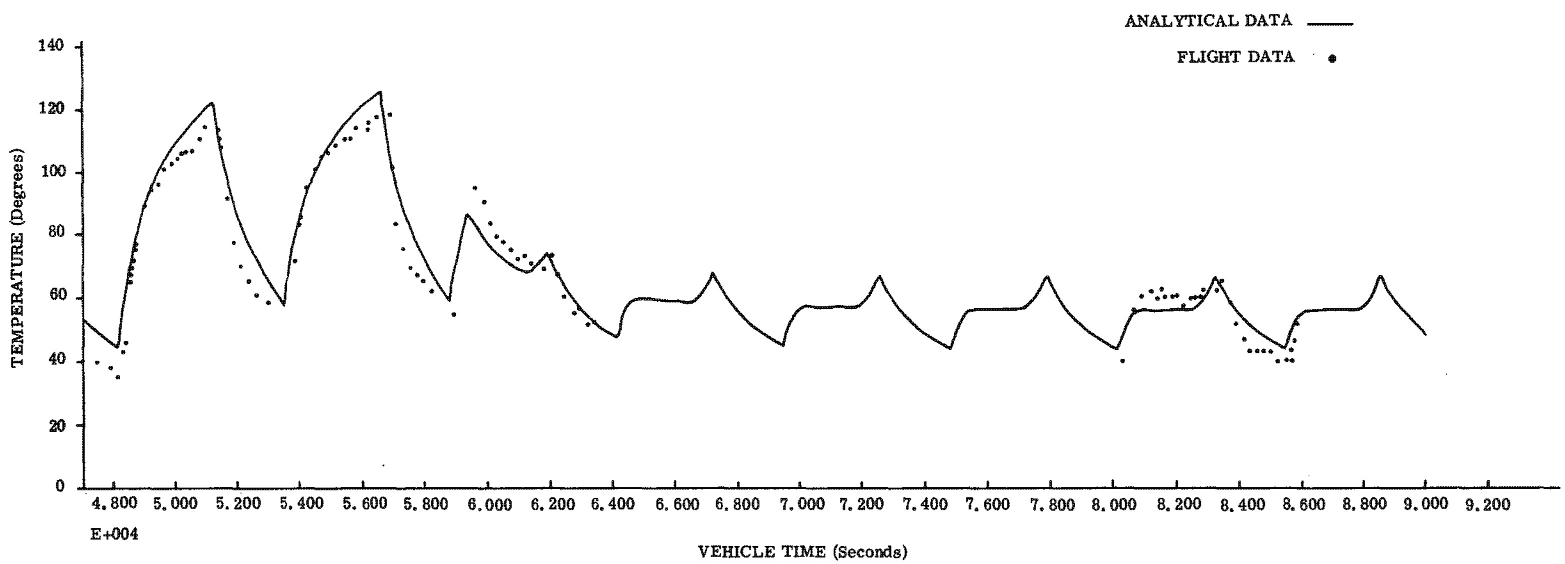

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FIGURE 5-4

BAY 1 DOOR TEMPERATURE HISTORY

(Fly Reverse Experiment)

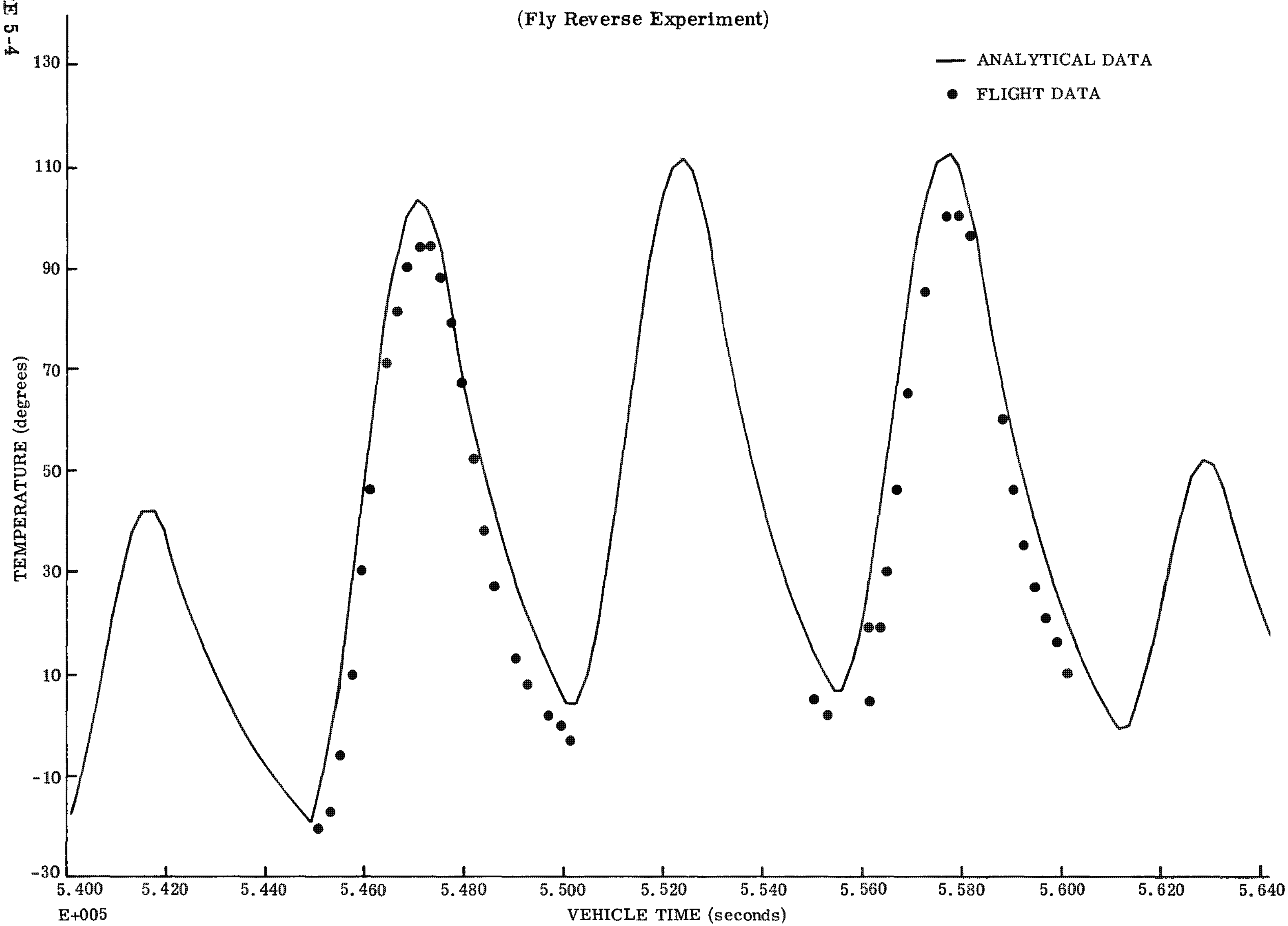

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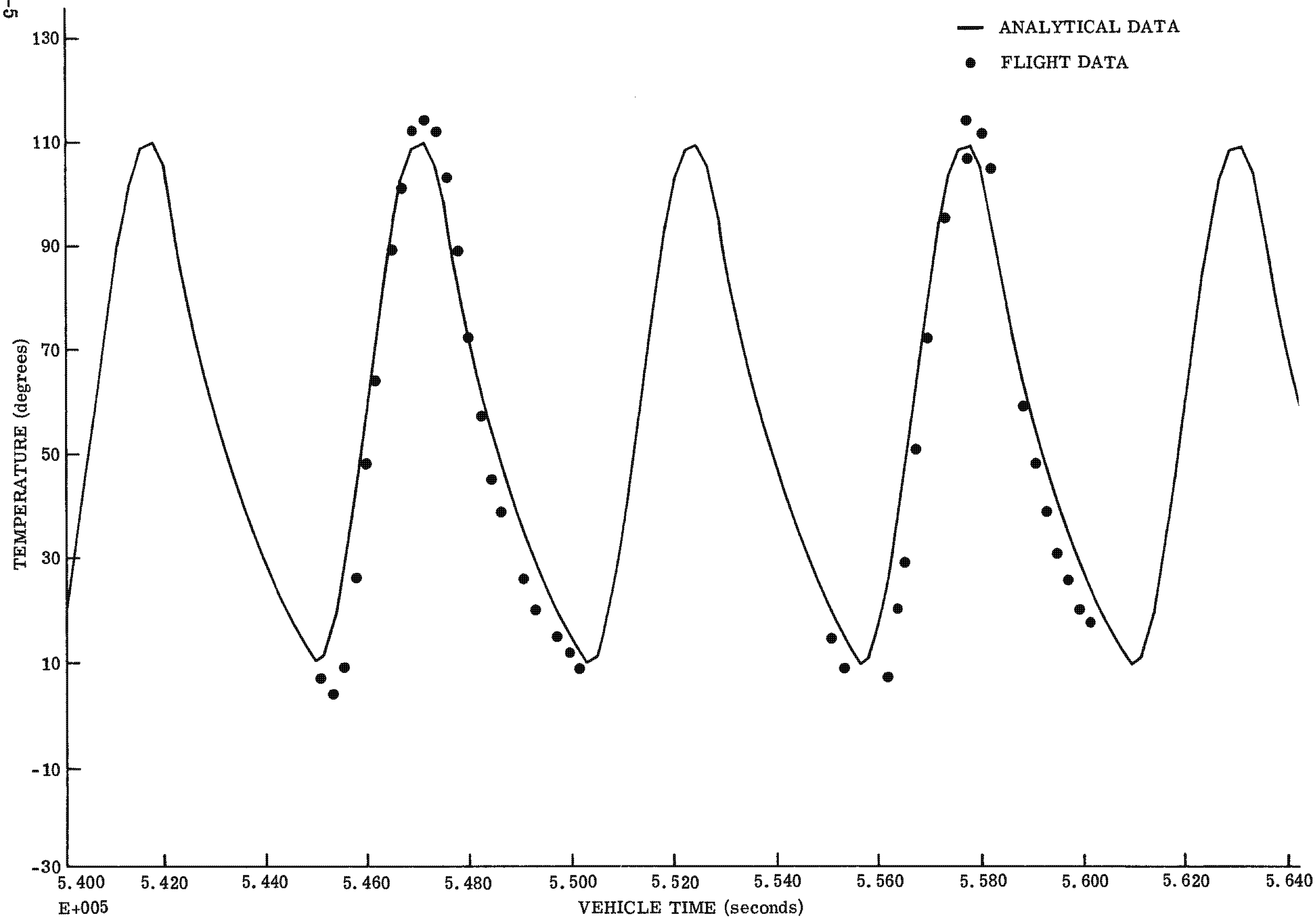
FIGURE 5-5

BAY 2 DOOR TEMPERATURE HISTORY

(Fly Reverse Experiment)



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temperature limits, SV-2 will be flown at or near the same beta angle as SV-1 (+20 deg). To protect the Bay 12 Batteries from overheating again, the entire Bay 12 Battery Module will be relocated to Bay 3 which will be vacant on SV-2 due to deletion of the Reserve Power System. Minor changes to the Bay 3 thermal design will be made to provide the proper environment for the Battery Module, thus insuring that all four Type 29's will run within limits on SV-2.

Having insured proper temperature control of all Aft Section equipment by flying the same beta angle as SV-1 and relocating the Bay 12 Batteries to Bay 3, the following objectives pertaining to the source of the contamination and the evaluation of possible fixes will be addressed on SV-2:

A. Distinguish source - liftoff cloud

B. Distinguish source - SRM staging

C. Evaluate degradation of present thermal control surfaces (white silicone, bare aluminum, black Kemacryl).

D. Assess smooth skin over corrugations as a possible fix.

E. Assess Flexible Optical Solar Reflector (FOSR) - aluminum foil with a thin layer of teflon as a substitute for white paint.

F. Assess Z-93 - an inorganic, ceramic-based white paint - as a substitute for white paint.

G. Evaluate nature of liftoff cloud contaminants.

In order to accomplish these objectives on SV-2 the following experiments will be performed:

A. Bay 11

The existing Bay 11 Door will be replaced with a modified Bay 6 door. The Bay 6 door will not have a Horizon Sensor Fairing and the four Horizon Sensor Head holes will be used for calorimeters having white silicone, black Kemacryl, bare aluminum and Z-93 surfaces. One of the four calorimeter panels will be protected through the lift-off cloud event, a second panel will be protected through the SRM staging event, and the remaining two panels will be exposed throughout ascent. The layout of this experimental Bay 11 door is shown in Figure 5-6. Orbital temperature data from these calorimeter panels are expected to satisfy objectives A, B, C and F as stated above.

B. Bay 12

The existing Bay 12 door will be covered with a dummy corrugated door mounted on thermal stand-offs. This dummy door will be segmented into three different configurations: (1) dummy corrugations painted white silicone (like original Bay 12 door); (2) dummy corrugations covered with a smooth skin which is painted white silicone; and (3) dummy corrugations covered with a smooth skin which is finished with FOSR. This experimental Bay 12 door is shown in Figure 5-6. Thermocouple data from each of these three sections on orbit is expected to satisfy objectives C, D, and E.

C. Umbilical Contamination Samples

Three boxes containing eight thermal control surface samples will be mounted on the

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SV-2 CONTAMINATION EXPERIMENTS

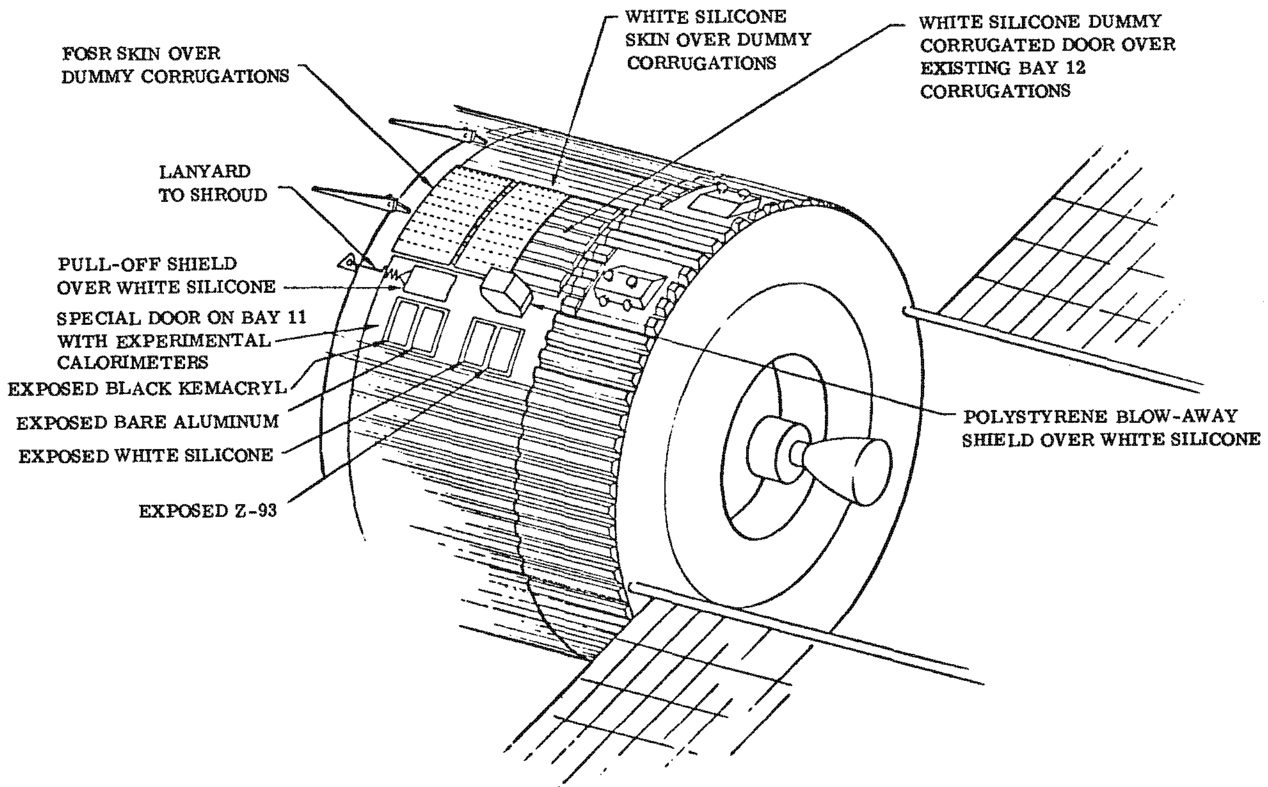

FIGURE 5-6

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umbilical arms near the SV aft and Mid-Sections. The boxes will be closed shortly after SRM ignition so that the samples are exposed to the lift-off cloud but are subsequently protected from direct SRM exhaust impingement. The layout of these boxes is shown in Figure 5-7. Surface property data pre- and post-launch should satisfy objective G.

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UMBILICAL ARM CONTAMINATION SAMPLE BOXES

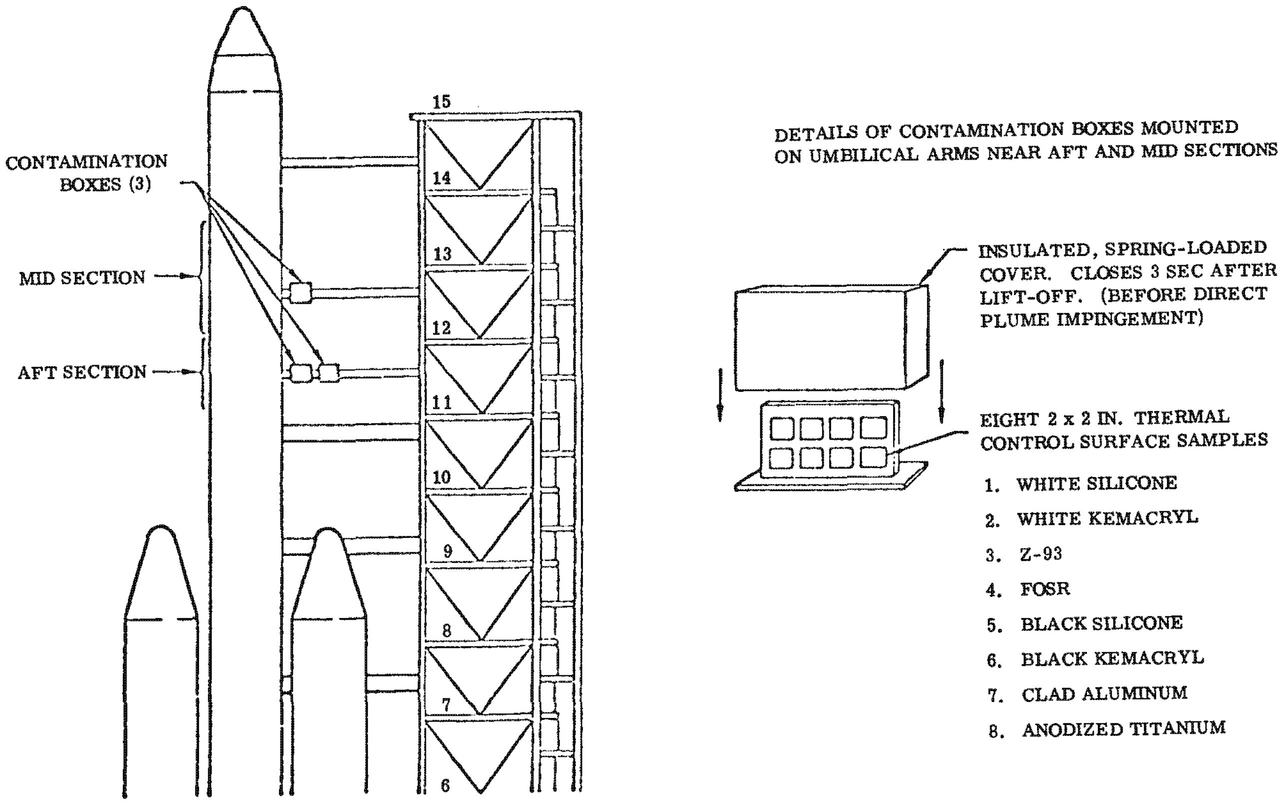

FIGURE 5-7

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Mass Properties, Structures & Dynamics

VI

SECTION VI

MASS PROPERTIES, STRUCTURES AND DYNAMICS

6.1 MASS PROPERTIES

The history of the Satellite Vehicle (SV) mass properties throughout the flight are given in Table 6-1. Table 6-2 indicates the expendable usage for a 30-day mission plus the vehicle deboost operation.

6.2 STRUCTURES AND DYNAMICS

6.2.1 Ascent Acceleration

The axial accelerations were measured at Station 1642 and 2180 and are shown in Fig. 6-1 along with the design and static test levels for the complete ascent. The significant dynamic acceleration levels (measured on the SV forward bulkhead at Station 1642) are presented in Table 6-3.

TABLE 6-1

MASS PROPERTIES

Event	Weight (lb)	SV Sta (in.)	$\bar{Y}$ (in.)	$\bar{Z}$ (in.)	I_x (slug-ft^2)	I_y (slug-ft^2)	I_z (slug-ft^2)	I_{xy} (slug-ft^2)	I_{xz} (slug-ft^2)	I_{yz} (slug-ft^2)
Separation from Stage II	20627	2016.9	1.59	4.21	5004.4	134820.1	134156.1	-966.9	2026.3	68.7
Solar Array Extended	20627	2017.4	1.59	4.21	6160.0	135773.7	136210.0	-963.6	2035.2	-153.2
Before RV 1	20608	2011.4	1.61	4.60	6146.7	142505.0	142942.2	-1022.3	2424.8	-160.0
After RV 1	19177	2036.4	1.74	3.68	5950.9	105121.6	105687.3	-838.5	1067.4	-153.2
Before RV 2	19112	2028.5	1.77	4.26	5943.9	112016.3	112572.1	-911.6	1550.7	-161.9
After RV 2	17601	2053.1	1.92	3.15	5734.1	82661.2	83359.0	-729.9	229.3	-153.7
Before RV 3	17210	2043.6	2.04	3.94	5714.5	86564.1	87250.4	-841.8	581.1	-172.4
After RV 3	15682	2066.7	2.24	2.63	5497.4	65990.6	66824.8	-664.6	-574.5	-162.4
Before RV 4	15649	2063.5	2.30	3.04	5492.2	67541.0	68365.7	-708.6	-422.8	-172.6
After RV 4	14346	2081.0	2.50	1.86	5299.3	56047.6	56999.5	-572.0	-1192.1	-163.3
Begin Deboost	12844	2069.5	2.56	1.68	5202.4	52478.6	53472.8	-584.0	-1124.7	-198.8
End of Deboost	12111	2062.3	2.70	1.75	5190.3	50119.3	51115.3	-628.1	-1146.1	-201.4

TABLE 6-2

EXPENDABLES

	AMOUNT (lbs)
ALTITUDE CONTROL PROPELLANT	
ORBIT CONTROL	6
DEBOOST	11
FLIGHT CONTROL	118
TOTAL USED	135
LOAD	390
ORBIT ADJUST PROPELLANT	
ORBIT MAINTENANCE	370
DEBOOST	690
TOTAL USED	1060
LOAD	2900
LIFEBOAT PROPELLANT	
OPERATING SEQUENCE	0
LOAD	240

TABLE 6-3

PYLON RESPONSES
(Station 1642)

Event	Axis	Level g's Zero to Peak	Frequency (Hz)
Lift Off	Z	0.75	4.0
SRM Burn	Y, Z	0.5	4.0, 4.0
SRM Burn out	Z	0.5	16.0
SRM Separation	X, Z	0.5	20.0, 11.0
Stage I (POGO)	X	0.4	19.0
Stage I Shutdown (STG II IC)	X	1.0	19.6
	Y	0.4	20.0
	Z	0.8	20.0

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AXIAL ACCELERATION HISTORY

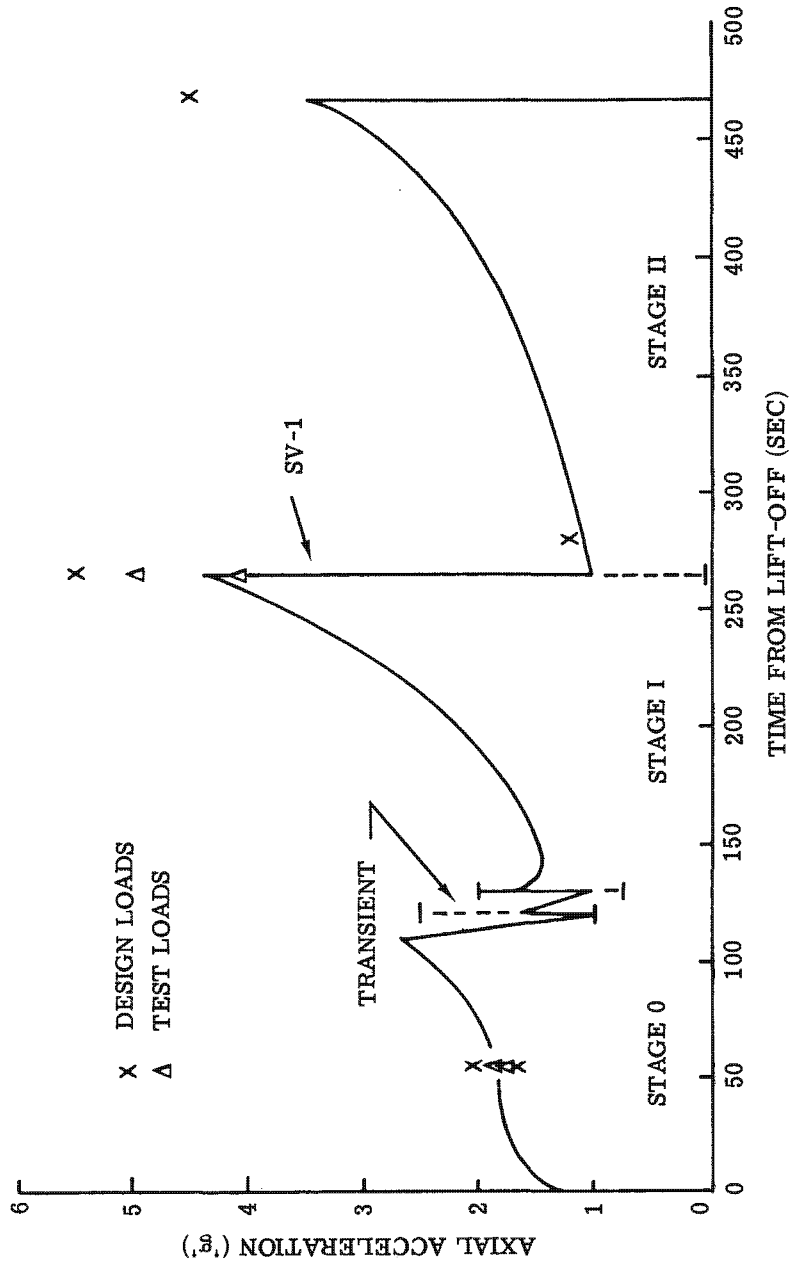

FIGURE 6-1

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The accelerations measured during Stage 0 flight indicate no excessive dynamic loadings on the SV. Stage II shutdown resulted in very low dynamic accelerations such that no particular fundamental mode excitation could be identified from the accelerometer traces.

Of particular interest on any Titan flight is the Stage I shutdown event and the fundamental longitudinal mode excitation phenomena. Stage I shutdown is the most severe structural dynamic loading event for loads on local structure and payload masses.

The maximum dynamic acceleration levels measured on the SV structure (at Station 1642) are presented in Table 6-4 along with the predicted levels for that location. The predictions are based on the set of 27 depletion shutdown transients used as design criteria for this event.

TABLE 6-4

STAGE I SHUTDOWN ACCELERATIONS

Location	Axis	Flight	Model Results Actual Engine	Mean + 3σ 27 Engines
Fwd Pylon	X	+4.3/-0.2	+4.6/-0.5	+5.38/-3.31
Fwd Pylon	Y	±0.2	±0.9	±2.75
Fwd Pylon	Z	±0.65	±1.45	+1.9/-1.5
Aft Section	X	+4.2/+0.2	+4.5/0	
Aft Section	Z	±0.2	±0.4	
Shroud	Z	±0.6	±1.1	

NOTES: 1. The flight data is filtered to include data for 40 cps or less.

2. The model results are for the actual engine chamber pressure history for the flight booster.

As can be seen from Table 6-4, the measured response levels are significantly lower than the predictions. In order to compare the principal response frequencies, the accelerations measured due to Stage I shutdown were processed to obtain the response in various frequency ranges. The response levels and frequencies in three frequency ranges are shown in Table 6-5. The first three longitudinal modes of the vehicle are identified along with the predicted frequencies.

The fundamental longitudinal mode response, which can occur anytime during Stage I flight, reached its highest level shortly before Stage I burnout. The maximum level measured was 0.4 g's (zero to peak) at station 1642 which was well below the design level of 1.0 g at the same location.

In conclusion, the low frequency accelerometer measurements indicated no load problems during ascent flight. The severe loading event, Stage I shutdown, resulted in response levels lower than expected.

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TABLE 6-5

BAND PASS ANALYSIS RESULTS
(Stage I Shutdown Transients)

Frequency Range	Axis	Frequency (Hz)	Acceleration (g's)	Mode	Predicted Frequency (Hz)
16-18	X	17.0	±0.31	1st Longitudinal	16.9
	Y	17.0	±0.1		
	Z	17.0	±0.2		
18-23	X	20.4	±0.6	2nd Longitudinal	20.6
	Y	20.4	±0.1		
	Z	20.4	±0.35		
23-30	Z	29.0	±0.1	3rd (Booster Tank Mode)	28.5

6.2.2 Ascent Acoustic and Vibration Environment

The acoustic and vibration measurements are shown in Fig. 6-2. Acceptable data quality from all nine channels were transmitted except for a data drift problem on the microphones at lift off. Most severe drifting occurred on measurements 960 and 967; these channels drifted beyond their band edges and the data was lost for a short time. The drift was due to microphone sensitivity to static pressure pulses at ignition and lift off. Corrective high pass filters are being installed on the SV-2 microphone amplifiers.

A tabulation of maximum overall sound pressure levels and G_{rms} values are also presented in Fig. 6-2. The flight reading of 120 db for sensor 961 is lower than the anticipated 75% worse case environment. This is attributed to the very local and transient nature of the external shock and the benign flight trajectory. In general, the flight data show that test levels are not exceeded except for certain frequency bands. More comprehensive data reduction is under way and the test procedures are being reviewed with respect to the flight data; however, no changes in test specifications or procedures are indicated at this time.

6.2.3 Ascent Vehicle Loads

Launch vehicle loads were predicted for the actual wind aloft profiles obtained from Rawinsonde data at T-3 hours and at launch. The ground winds at launch were 8 knots from 330° and the maximum wind aloft was 42 knots from 238° at 43,000 feet. The predicted loads for these data were: shroud bending 45% of allowable, shroud side force 34% of allowable, and booster thrust vector control required 37% of allowable capacity.

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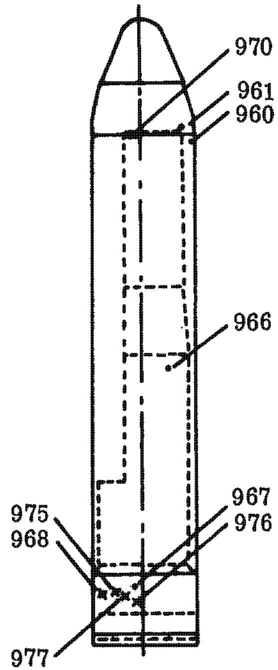
FIGURE 6-2

SUMMARY OF MAXIMUM OVERALL SOUND PRESSURE LEVEL AND G_{RMS} VALUES

MEASUREMENT LOCATIONS



• MICROPHONES
× VIBRATION PICK-UPS

	LIFT OFF	MAX FLIGHT	ACCEPTANCE TEST LEVELS
MICROPHONES			
960 FWD EXT	154 dB	151 dB	159 (2)
961 FWD INT	144	120	139.5 (3)
966 MID INT	134.5	119	136 (3)
967 AFT INT	134.5	125	139.5 (3)
VIBRATION PICK-UPS			
970 FWD BULKHEAD	$6.0G_{RMS}$	$0.6G_{RMS}$	$12.2G_{RMS}$
968 PCM MODULE	0.75	0.5	(1)
975 T&T MODULE	1.0	0.8	1.3
976 ARM MODULE	1.6	0.4	2.0
977 LB MODULE	1.4	1.0	2.7

(1) NO TEST DATA

(2) EXTERNALLY IMPOSED ENVIRONMENT

(3) MEASURED DATA FOR SV-1 ACCEPTANCE

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6.2.4 Shroud Temperatures and Thermal Deformations

The lofted-dogleg trajectory chosen for the first launch was benign with regard to both maximum temperature and the circumferential gradient. The measured temperatures went from a high of 245°F at the 10°/25° bi-conic intersection to a low of 140°F at the aft end of the shroud. The corresponding design temperatures were 570°F and 405°F. As a result of flying such a cool trajectory, the ability of the Invor Rings, which were added to prevent interference with payload, was not demonstrated.

6.2.5 Solar Array

The history of the solar array erection and deployment are shown in Fig. 6-3. Since the arrays were deployed and erected in the proper position for this flight's beta angle, no positioning was necessary and none was performed.

Data exists to describe completely the erection of the left solar array and it was completed 230 sec after deployment was commanded. Only partial data is available for the other motions; however, data for the final portion of the deployment of the right array permits probable histories for all the motions to be sketched in on Fig. 6-3. It is estimated that the right solar array erected in 320 sec. The time for the right deployment was 650 sec and the time for the left deployment is estimated at 510 sec. Temperatures of the left and right erection dampers were 63°and 65°F respectively. The right array also took longer to erect and deploy during ground tests.

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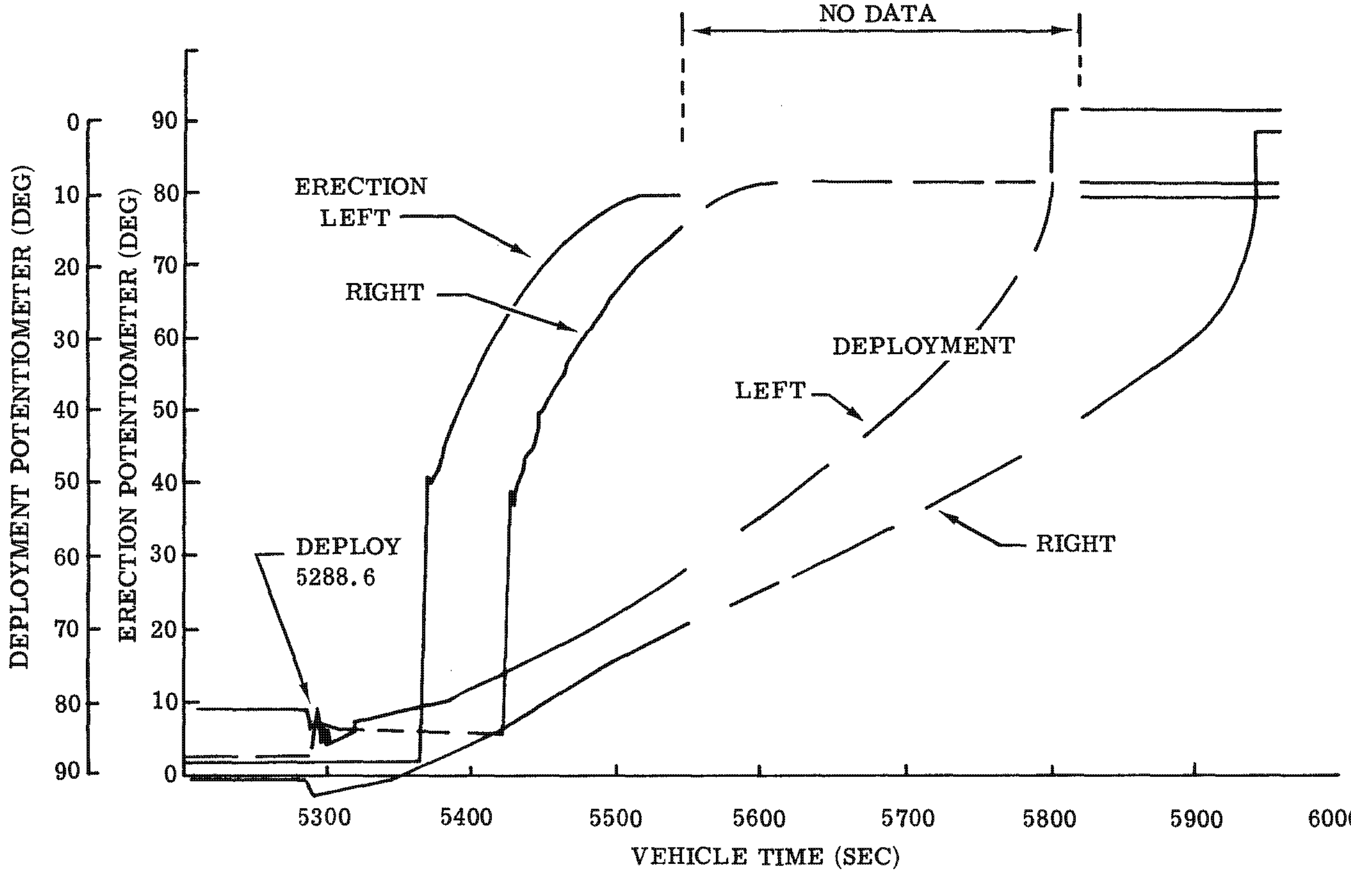
FIGURE 6-3

SOLAR ARRAY ERECTION AND DEPLOYMENT



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Operational Support

VII

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SECTION VII

OPERATIONAL SUPPORT

7.1 SOFTWARE

The software configuration used to support this mission was TUNITY (IOC) and AOES/Tracking Station Model 13G. Using a nominal three rev load cycle, 223 command messages were loaded. There were no software problems identified during the mission which impacted flight objectives. A problem known prior to 1201 launch within the selection algorithm slightly favored the first operation on a rev over the others. This resulted in some operations being chosen at less than the desired efficiency though they were still valid intelligence requirements. Based upon experience from this mission, immediate corrective action within the software for Mission 1202 has been initiated.

7.2 SATELLITE TEST CENTER (STC)

There were a total of 310 messages generated and checked. Of these messages, 256 were approved for loading, of which 33 were contingency messages generated but not loaded. Of the 54 rejected messages, 32 were a result of communication problems or misunderstandings regarding original generation requirements. The remaining 22 were caused by problems within the software, for which corrective action is under consideration or being implemented. There were no STC problems which impacted the mission objectives.

7.3 REMOTE TRACKING STATION (RTS)

With one minor exception, the Remote Tracking Stations furnished all support required to meet mission objectives. The exception was a COOK power failure which forced deletion of a planned engineering test on Rev 55.

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Appendix A – References

Appendix B – Glossary

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APPENDIX A

REFERENCES

1. HEXAGON Program Preliminary Post Flight Report For Flight No. 1. Technical advisory report, BIF-107W-71054-71, 11 August 1971. (S/H)

2. Flight Test Engineering Analysis Report For The HEXAGON Program Satellite Vehicle No. 1. LMSC Integrating Contractor, BIF-003-W/2-065766-71, 25 August 1971. (TS/H)

3. Flight Test Objectives/Performance Analysis (T III D-1 Post Flight Report). Martin-Marietta Report, MCR-7-251, September 1971.

4. Titan III D-1 Flight Test Report. Western Electric Company Report, 82,493 28 July 1971.

5. Flight Analysis Report -- Reentry Vehicle 1-1 thru 1-4. McDonald-Douglas Reports

   1-1 BIF-077/001W-1167-71

   1-2 BIF-077/001W-1168-71

   1-3 BIF-077/001W-1169-71

   1-4 BIF-077/001W-1170-71

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APPENDIX B
GLOSSARY

ACS	Attitude Control System
BFE	Best Fit Ephemeris
BV	Booster Vehicle
CCC	Charge Current Controller
CORN	Controlled Range Network
CV	Constant Velocity
DIU	Data Interface Unit
ECS	Extended Command System
EDAP	Electrical Distribution and Power
ESD	Emergency Shut Down
ESO	Emergency Shut Down Override
FCEA	Flight Control Electronics Assembly
FOSR	Flexible Optical Solar Reflector
Fps	Feet per Second
FST	Flight Support Team
g	Gravity
HS	Horizon Sensor
HSA	Horizon Sensor Assembly
Hz	Cycles per Second (HERTZ)
ICD	Interface Control Document
Ips	Inches per Second
IRA	Inertial Reference Assembly
MCM	Mapping Camera Module
MCS	Minimal Command System
MONO	Monoscopic Operation
NSPC	Normal Stored Program Command
OA	Orbit Adjust
OAS	Orbit Adjust System
OB	Optical Bar
OP	Camera System Operation
P	X-axis Magnetometer Output
PCM	Pulse Code Modulation

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APPENDIX B
GLOSSARY (CONT'D)

PDA	Positional Drive Assembly (Solar Array)
PDJB	Power Distribution J-Box
PDWN	Pitch Down
PFA	Post Flight Analysis
PGR	Pitch Gyro Rate
PIP	Predicted Impact Point
P/L	Payload
PMU	Programmable Memory Unit
PVA	Pitch Vehicle Attitude
Q	Y-axis Magnetometer Output
R	Z-axis Magnetometer Output
RCS	Reaction Control System
REM	Reaction Engine Module
REV	Orbital Revolution
RGR	Roll Gyro Rate
RPS	Reserve Power System
RTS	Remote Tracking Station
RV	Reentry Vehicle
RVA	Roll Vehicle Attitude
SBA	Satellite Basic Assembly (Aft Section)
SBAC	Satellite Basic Assembly Contractor
SCC	Subsystem Command and Control
SCF	Satellite Control Facility
SDV-3	Satellite Development Vehicle
SECO	Stage II Engine Shut Off
SGLS	Space Ground Link System
SOLO	Operations Beyond the Primary Mission
SPC	Stored Program Command
SPEC	Specification
SPL	Sound Pressure Level
SRM	Solid Rocket Motor
SSC	Sensor Subsystem Contractor
STC	Satellite Test Center

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APPENDIX B
GLOSSARY (CONT'D)

SV	Satellite Vehicle
TM	Telemetry
TT&C	Telemetry, Tracking and Control
VCO	Voltage Control Oscillator
Vs	Coarse Film Path Velocity
VSPC	Variable Stored Program Command
Vx/h	Orbital Angular Rate, In-Track
Vy/h	Orbital Angular Rate, Cross-Track
YGR	Yaw Gyro Rate
YVA	Yaw Vehicle Attitude

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