SCL-DR-720077

March 1973

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ROLL CONTROL RESONANCE TEST VEHICLE (RCRTV)
SYSTEM POSTFLIGHT EVALUATION REPORT

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ABSTRACT

This report presents the flight test results for a reentry vehicle spin-fin roll control system. The test vehicle, designated the Roll Control Resonance Test Vehicle, was flight tested on May 10, 1972, at the Sandia Laboratories Tonopah Test Range. The purposes of the flight were (1) to demonstrate and evaluate the capability of the control system to overcome a roll resonance flight instability and (2) to confirm the analytic predictions of system performance. The flight test vehicle had built-in asymmetries which, without roll control, would have caused a persistent roll resonance instability to occur. The test was successful in that all systems functioned properly and all program and flight test objectives were accomplished. Flight test data presented herein confirm analytic predictions and flight simulation methods. Results show that the system maintained roll rate control and prevented the roll resonance instability.

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ACKNOWLEDGEMENT

The author wishes to acknowledge the effort and support of a large number of individuals and organizations in connection with the RCRTV program. In particular, the significant contributions made by the following key personnel are gratefully noted: J. E. Suazo and E. W. Hall, Mechanical Design Division 5623; L. R. Rollstin, Rocket Systems Division, 5624; A. E. Hodapp, Aeroballistic Division 5625; D. W. Johnson, Deceleration and Recovery Systems Division 5626; R. G. Newman, Transducer Evaluation and Calibration Division 7511; G. W. Henderson and T. J. Weber, Exploratory Development Division 8176; W. T. Ashurst, H. W. Coleman, and E. C. Lemmon, Aerothermodynamics Division 8351; R. D. Robinett and R. E. Howell, Test Project Management Department 9310A; and R. H. Lanes and K. W. Shrock, Instrumentation Applications Division II 9483.

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ROLL CONTROL RESONANCE TEST VEHICLE (RCRTV) SYSTEM POSTFLIGHT EVALUATION REPORT

Introduction

Flight test experience has indicated that ballistic reentry vehicles (RV) are plagued with problems of anomalous roll rate behavior. These problems, which include spin-up, roll reversal, and persistent roll resonance, can lead to serious flight performance degradation resulting in impact dispersions or catastrophic flight failure. Roll anomalies arise from small aerodynamic and mass asymmetries; and since these asymmetries are often ablation-induced, close control of RV manufacturing tolerances does not always eliminate the resulting flight instabilities. It is evident, therefore, that the flight performance of RVs can be improved by introducing roll rate control.

A study was conducted at Sandia Laboratories, Livermore (SLL), to investigate methods for exercising such control. One phase of the SLL roll control development program involved the design, development, and flight testing of a centrifugally actuated, fin roll control system. A system selected for detailed evaluation was one which employs variable-cant, aerodynamic spin fins. Among systems proposed, this one is unique in that (a) the fin motions are coupled and (b) a four-fin configuration is used, including two fins with variable cant and two with cant angle fixed at zero degrees. Included in the program to study fin roll control, was the task of developing both the analytic methods and the simulation techniques to describe the complex aerodynamics, thermodynamics, and flight dynamics of finned RVs. Results of the analysis indicated that such a system could be made to control the spin rate of a vehicle throughout reentry and that the system operation would remain stable through a wide range of environmental effects.

As a means for flight testing the candidate roll control systems, the Sandia Reentry Vehicle Resonance Test Vehicle (RVRTV) flight system was selected. The RVRTV program consists of a series of flights wherein vehicles with various types of built-in asymmetries are flown to investigate flight dynamic behavior during both persistent and transient resonance.

Preceding the controlled flight reported here, a flight without roll control was made to demonstrate the persistent roll resonance (roll lock-in) behavior of the vehicle and to verify the analytic prediction methods for its uncontrolled flight. Then the present spin-fin roll control system configuration, designated the Roll Control Resonance Test Vehicle (RCRTV)¹ was flight tested to demonstrate the capability of the system to control the spin rate of the asymmetrical vehicle and, again, to verify the analytic method used to predict flight behavior.

The RCRTV test flight on May 10, 1972, at the Sandia Tonopah Test Range (TTR), was made at a low altitude trajectory and a supersonic Mach number. The test was successful. This report describes the RCRTV system and flight test, and it presents the flight test results together with the analysis and evaluation of those data.

Results indicate that (1) the system maintained control through a severe roll resonance instability; (2) the vehicle aerodynamic drag and static stability were sharply increased by the fins; (3) the fin aerodynamic roll moment characteristics were in good agreement with the analytic predictions; and (4) the flight simulations accurately represented the actual flight performance, thus confirming the analytical models.

All program and flight test objectives were accomplished. The test therefore served as an intermediate step to the full-scale flight test of the Roll Control Test Vehicle (RCTV)² flown during October 1972.

RCRTV Program Objectives

The purpose of the RCRTV program is to conduct a free-flight test and evaluate the performance of the centrifugally actuated, hydraulically coupled, spin-fin roll control system. The general program and flight test objectives are as follows:

1. Demonstrate the capability of the fin roll control system to control the spin rate of an asymmetric flight vehicle that would otherwise exhibit a sustained roll resonance instability;
2. Develop the necessary analytical prediction and simulation techniques needed to describe the aerodynamic, thermodynamic, and flight dynamic performance of finned RVs;
3. Develop the necessary analysis and ground testing techniques needed to specify the characteristics and dynamic performance of the roll control system; and

4. Provide an intermediate step to full-scale flight testing.²

In addition, the following detailed flight test objectives are to be accomplished:

1. Obtain aerodynamic data on fins, verify preflight analytic predictions, and demonstrate the credibility of a four-fin configuration;
2. Obtain detailed data on body and control system dynamics (acceleration and angular rate components as well as angular position) and verify flight simulation methods;
3. Obtain fin leading edge and cone surface thermodynamic data (heat flux and temperature) and verify preflight analytic predictions; and
4. Recover payload intact by means of a parachute.

Flight System Description

Booster

The RCRTV boost system is a two-stage NIKE-HERCULES rocket system.* Each stage consists of a Nike M5 motor having a burn time of 3.5 seconds, and each nominally delivers 40,000 pounds of thrust. The boost system is equipped with a telemetry system so that the proper separation and ignition signals may be provided. After first-stage burn-out and before second-stage ignition, the two motors are separated by aerodynamic drag. The vehicle is deployed from the second-stage motor by an explosive separation and booster retardation system. The complete RCRTV flight system before flight is shown in Figure 1.

Flight Vehicle

Shown in Figure 2 is the Roll Control Resonance Test Vehicle before flight. A schematic of the flight vehicle is given in Figure 3, while a description of the physical properties is provided in Table I.**

*Booster interface is provided by L. R. Rollstin, Rocket Systems Division 5624.

**The payload design was provided by J. E. Suazo, Mechanical Design Division 5623.

Figure 1. Roll Control Resonance Test Vehicle (RCRTV) Flight System

Figure 2. Preflight RCRTV

Figure 3. RCRT Vehicle

TABLE I
FLIGHT VEHICLE PHYSICAL PROPERTIES

Cone half angle	10°
Bluntness ratio	0.017
Base diameter	15 in
Vehicle actual length	41.941 in
Weight	106.4 lb
Center-of-mass distance from nose (63% of length)	26.425 in
Pitch moment of inertia I_y	2.945 slug-ft ²
Yaw moment of inertia I_z	2.937 slug-ft ²
Roll moment of inertia I_x	0.317 slug-ft ²
Product of inertia I_{xy}	0.0122 slug-ft ²
Principal axis inclination	0.27°
Center-of-mass offset (along body Z axis)	0.10 in
Static margin (with fins)	6.2% of length

The vehicle is a sharp, 10-degree half-angle cone with a 15-inch base diameter and a total weight of 106.3 pounds. The external geometry of this vehicle, excluding the fins, is the same as that for a standard dynamic stability correlation model used by the Supersonic Tunnel Association and the AGARD.* It has compound mass and inertia asymmetries sufficient to produce, in the absence of roll control, a sustained roll resonance instability. The vehicle principal axis is inclined 0.25 degree relative to the geometric centerline. Normal to the plane containing these axes, the center of mass is displaced 0.1 inch. It is the combined effects of these two asymmetries that is the source of a destabilizing roll torque.

The details of the flight vehicle design are shown in Figure 4. The unit is made of an aluminum-magnesium alloy, Al Mag 35. The forward 10 inches of the nose of the vehicle serves as a telemetry antenna and is therefore insulated from the body by a quartz ring. Four aluminum fins are used to provide spin rate control. Two of these are driven by the centrifugally actuated roll control system; two are fixed at a zero geometric cant angle.

The payload is mounted to the booster adapter and separation system** by three explosive bolts equally spaced around the base of the vehicle. At second-stage burnout, the simultaneous detonation of these bolts and deployment of booster aerodynamic retardation plates (drag brakes) separate the vehicle from the booster.

A parachute recovery system*** is included and packaged in the aft end of the flight vehicle. The system consists of a 3-foot-diameter pilot deployed 23 seconds after launch and a 12-foot-diameter main parachute deployed 33 seconds after launch. The recovery system performed properly and the vehicle was recovered intact (Figure 5).

Roll Control System

The vehicle roll control system is shown conceptually in Figure 6.**** Vehicle roll torque and roll control are provided by means of external aerodynamic control surfaces (spin fins). Fin cant is determined by the radial

*R. Fail and H. C. Garner, Calibration Models for Dynamic Stability Tests, Report 563, Advisory Group for Aerospace Research and Development (AGARD), Nevilly-Sur-Seine, France, 1968.

**Design provided by J. E. Suazo and E. W. Hall, Mechanical Design Division 5622

***Designed by D. W. Johnson, Deceleration and Recovery Systems Division 5626

****The system was designed by T. J. Weber, Exploratory Development Division I 8178 (Reference 3).

1	WASHER, FLAT	1/4-20 UNF x 3/4 LG	1	66
2	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	65
3	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	64
4	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	63
5	WASHER, FLAT	1/4-20 UNF x 3/4 LG	1	62
6	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	61
7	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	60
8	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	59
9	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	58
10	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	57
11	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	56
12	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	55
13	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	54
14	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	53
15	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	52
16	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	51
17	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	50
18	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	49
19	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	48
20	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	47
21	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	46
22	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	45
23	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	44
24	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	43
25	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	42
26	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	41
27	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	40
28	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	39
29	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	38
30	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	37
31	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	36
32	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	35
33	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	34
34	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	33
35	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	32
36	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	31
37	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	30
38	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	29
39	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	28
40	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	27
41	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	26
42	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	25
43	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	24
44	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	23
45	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	22
46	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	21
47	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	20
48	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	19
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53	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	14
54	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	13
55	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	12
56	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	11
57	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	10
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59	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	8
60	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	7
61	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	6
62	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	5
63	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	4
64	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	3
65	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	2
66	SCREW, CAP SOC HD (1/4-20 UNF x 3/4 LG) STEEL		1	1

Figure 4. RCRT Vehicle Design Layout (TO3902)

Figure 5. Recovered RCRTV

Figure 6. Centrifugally Operated Fin Roll Control System

position of the two system sensing/drive masses. Mass position is given by the summation of the system spring forces and centrifugal forces (as provided by the vehicle spin rate). Any movement of a mass causes a displacement of hydraulic fluid in a drive bellows, and through a hydraulic line, an equivalent displacement in a slave bellows. The slave bellows, through a rod and bell crank, actuates a fin. The coupling bellows is hard-mounted to the slave bellows and hydraulically linked to the opposite drive mass. The circuit is completed by the coupling hydraulic line which leads to a second slave unit.

If the hydraulic fluid is considered incompressible, and if there is no compliance in the hydraulic lines and bellows, the masses and fins are coupled. The system is thus centrifugally actuated and hydraulically coupled. Ideally, the system displacement is governed by a second-order differential equation.

The roll control system physical properties are listed in Table II. Each sensing mass weighs 0.75 pound and is constructed of tungsten (18g/cm^2). The total system weight (including instrumentation) is 7.2 pounds.

TABLE II
ROLL CONTROL SYSTEM PHYSICAL PROPERTIES

Weight (including all instrumentation)	7.194 lb		
$\bar{X}$	7.706 in (from base)	$\left. \begin{array}{l} \\ \\ \end{array} \right\} \begin{array}{l} \text{(from geometric} \\ \text{centerline)} \end{array}$	RCS center-of-mass position
$\bar{Y}$	0.0 in		
$\bar{Z}$	0.0 in		
I_x	0.0262 slug-ft ²	$\left. \begin{array}{l} \\ \\ \end{array} \right\}$	moments of inertia about center of mass
I_y	0.0212 slug-ft ²		
I_z	0.0137 slug-ft ²		

Details of the drive and slave unit design are shown in Figures 7 and 8, those of the fin drive mechanism, in Figure 9. Detailed descriptions of the roll control system componentry and its interface with the flight vehicle are given in Figures 10 and 11, respectively. The reader is referred to Reference 3 for further design details.

					17
					16
					15
					14
APC		ANTI-SIEZE LUBRICANT (SUITABLE MOLY TYPE)			13
		HOUSING ASSY CONSISTS OF ITEMS 1,4			12
1		END CAP 300 SERIES SST	1	11	
1	2-119	O-RING	2	10	
1		LOCKNUT AL 6061-T6	1	9	
1		PISTON SLUG MALLORY 2000	3	8	
1		PISTON SKIRT MALLORY 2000	3	7	
1		SCREW, PAN HD #10-32x1 SST	1	6	
1		SPRING TO SUIT NOTE 9 REQUIREMENTS	1	5	
1		NUT AL 6061-T6	2	4	
1	2-128	O-RING	2	3	
1		RETAINER AL 6061-T6	1	2	
1	K76720	TUBE A (TEL)	2	10	
1	123464	BELLONS SUB-ASSEMBLY	1	10	
1		CYLINDER AM 355 SST	2	18	
1		SLEEVE AM 355 SST	2	1A	
		HOUSING WELDMENT	1	1	
REQD	PART OR CONTROL NO.	DESCRIPTION	NOTE	QTY	ITEM
	INSTRUMENTS	APC = AS REQD PER ASSEMBLY	PM = PROCESS MATERIAL		
	INSTRUMENTS	ALT = ALTERNATE	EM = EXPENSE MATERIAL		

Figure 7. Drive Unit Design Detail (K77099)

					14
					13
					12
1	K76723	TUBE D (TEL)			11
1	K76722	TUBE C (TEL)			10
1		TUBE FLANGE, RIGHT	300 SERIES SST		9
2	123668	ROD		5	8
2	123677	TUBE		5	7
1		TUBE FLANGE, LEFT	300 SERIES SST		6
1		BRACKET, RIGHT	300 SERIES SST		5
1	123672	BELLOWS SPACER		5	4
1		BRACKET, LEFT	300 SERIES SST		3
4		NUT, HEX #10-32	SST		2
2	123669	BELLOWS SUBASSEMBLY		5	1
REQD	PART OR CONTROL NO.	DESCRIPTION	NOTE	QUANTITY	ITEM
	123669	AND: AS REQD PER ASSEMBLY	PM - PROCESS MATERIAL		
		ALT - ALTERNATE	EM - EXPENSE MATERIAL		

Figure 8. Slave Unit Design Detail (K77108)

Figure 9. Detail of RCRTV Fin Mechanism

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Figure 10. Roll Control System Detail (K77635)

Flight Vehicle Instrumentation and Telemetry

Instrumentation on board the payload is designed to provide a detailed evaluation of body and control-system dynamics.^{1,4} These data are comprehensive enough to allow an assessment of the analytic prediction techniques used in the development of the system. The payload includes vehicle dynamic, roll control system dynamic, and thermodynamic instrumentation, as well as a p-band telemetry system that transmits through a nose antenna. The function of each sensor is given in Table III, and a detailed listing of each measurement (including range and sampling rate) is given in Table IV.*

The telemetry system is a 2-watt transmitter operating at 248.6 MHz. The transmitter is frequency-modulated with the subcarrier oscillator frequencies and subcarrier assignments shown in Table V. Only the thermodynamic instrumentation (thermocouples and calorimeters) is commutated, and the rate for the commutated information is 30 samples per second.

In addition to the foregoing instrumentation, passive thermodynamic instrumentation in the form of heat-sensitive (temperature-sensitive) paint (Tempilaq)** is used on both the interior and exterior of the flight unit. Tempilaq tints responsive to various temperatures are placed on the top and sides of the fins, on the fin platforms, and on cone surface areas located aft and on either side of each fin. A detailed description of the Tempilaq placement is given in Reference 5.

A detailed description of range instrumentation and support is given in Reference 6.

Flight Trajectory Analysis

General Flight Characteristics

The flight trajectory sequence of events and the altitude history are depicted in Figure 12. Each Nike rocket booster has a nominal burn time of 3.5 seconds. Approximately 1.0 second is expended for staging after first-stage burn. At second-stage burnout, the vehicle is separated from the launch vehicle at the conditions shown in Figure 12.

*Payload instrumentation and telemetry designs were provided by R. H. Lanes and K. W. Shrock, Instrumentation Applications Division II 9483.

**A product of Tempil Corporation, New York, N. Y.

TABLE III
RCRTV INSTRUMENTATION AND FUNCTION

Sensor	Function
Tri-axis rate gyro	Measures pitch, yaw, and roll angular velocities about body fixed axes.
MARS (miniature altitude reference system) stable platform	Measures inertial pitch, yaw, and roll angles.
Accelerometers	Measure accelerations along body fixed axes and provide static and dynamic stability data.
Linear potentiometers	Measure roll control system mass position and fin cant.
Differential pressure transducers	Measure pressure differential across slave bellows assembly (determines applied torque on the fin).
Thermocouples	Measure temperatures on the vehicle shell backface. These sensors are located at two body-axial positions, two each at the fin station and approximately 14 inches forward of the fin station. At each axial location, the thermocouples are diametrically opposed in a plane 45° from the plane containing the moving fins.
Calorimeters	Measure heat flux at the vehicle surface and fin leading edge. A calorimeter is located on the leading edge of each fixed fin. The cone surface calorimeters are located in the same stations as the thermocouples. They are diametrically opposed and in a plane normal to the thermocouple plane.

TABLE IV
RCRTV INSTRUMENTATION

Measurement	Range	Sampling Rate
<u>Vehicle Dynamics</u>		
Roll rate	± 4000 deg/s	Continuous
Pitch rate	± 400 deg/s	
Yaw rate	± 400 deg/s	
Pitch	± 360 deg	
Yaw	± 85 deg	
Roll	± 360 deg	
X acceleration	± 50 g	
X acceleration	± 30 g	
Y acceleration	± 50 g	
Z acceleration	± 50 g	
<u>RCS Dynamics</u>		
*Fin displacements (2)	± 0.5 in	Continuous
*Hydraulic line differential pressures (2)	± 30 psid	
<u>Thermodynamics</u>		
Cone surface temperature (4)	0-600°F	30 samples/s
*Fin leading edge heat flux (2)	0-900 Btu/ft ² -s	
*Cone surface heat flux (4)	0-150 Btu/ft ² -s	

*Instrumentation supplied by Components & Instrumentation Section 8183-1;
all other instrumentation supplied by Instrumentation Applications
Division III 9483

TABLE V
RCRTV TELEMETRY SYSTEM SUBCARRIER ASSIGNMENTS
(MAIN CARRIER FREQUENCY 248.6 MHz, 2 WATTS)

Subcarrier Frequency (MHz)	Assignment
1.3	X acceleration
1.7	X acceleration
2.3	pitch rate
3.0	yaw rate
3.9	roll rate
5.4	yaw
7.35	pitch
10.5	roll
14.5	Z acceleration
22.0	Y acceleration
30.0	0° fin displacement (D2)
40.0	180° fin displacement (D1)
52.5	180° fin differential pressure (P2)
70.0	0° fin differential pressure (P1)
93.0	Commutator
	4 thermocouples
	2 fin calorimeters
	4 cone calorimeters

Figure 12. RCRTV Trajectory Events

The total boost phase duration is therefore approximately 8 seconds. The payload free-flight position of the trajectory simulates the dynamic pressure and vehicle natural pitch frequency histories for a second-resonance encounter. At approximately 15 seconds after vehicle separation, a 3-foot-diameter pilot parachute is deployed. This is followed at $T = 25$ seconds by the 12-foot-diameter main parachute. (Note: Except where otherwise specified, T = time from vehicle separation.)

Flight Trajectory Data

The flight vehicle trajectory is reconstructed from the radar⁷ and payload telemetry data. These data in conjunction with trajectory simulations are used in establishing the payload separation conditions as well as the time history of the trajectory performance variables. The radar measured position and velocity are used primarily to establish the boost phase trajectory and payload separation conditions.

Two accelerometers with different ranges (± 30 g and ± 50 g) were used to monitor payload axial acceleration history. The measured axial acceleration from these instruments is shown as a function of flight time in Figures 13 and 14. These accelerations (excluding any instrument bias) may be integrated to determine the payload axial velocity history. Because the ± 30 g accelerometer becomes saturated from $T = 6.0$ to 7.5 sec from launch and again from $T = 8.0$ to 9.0 seconds from launch, it is not used in determining the velocity history. The axial velocity history is considered equal to the total velocity. This assumption is true since during the boost phase, the angle of attack is essentially zero and during the payload free flight period, the angle of attack is small enough so that the axial velocity is within 1/2 percent of the total velocity.

The results of the trajectory analysis are summarized in Figures 15 and 16, in which the flight vehicle altitude and velocity history are presented. Data from both radar and accelerometers, which yield the same payload separation velocity are used to construct the velocity history. The initial conditions for the payload trajectory simulation included in Figures 15 and 16 are comprised of the separation velocity and separation altitude determined from radar. The simulation is obtained with the SPINFIN^{8, 9} trajectory program. The foregoing methods have provided an excellent characterization of the flight trajectory and payload separation conditions.

Meteorological Data¹⁰

Immediately before the flight, atmospheric data was obtained by means of rawinsonde balloons released in the neighborhood of the launch area. The air temperature and density data obtained (presented as a function of altitude in Figures 17 and 18, respectively) are used with the trajectory computations to determine dynamic pressure and Mach number. The actual atmospheric data used in this case deviate about 5 percent from the standard 1962 model atmosphere in the altitude range of interest. This deviation would produce some small errors in all trajectory performance variables.

Payload Separation Conditions and Vehicle Trajectory

The vehicle separation conditions (Table IV), as determined from the sources previously discussed, were very close to those expected. These conditions are read in as initial conditions to the SPINFIN trajectory simulation program to determine the payload trajectory characteristics. The RCRTV payload velocity, Mach number, and dynamic pressure history thus determined (presented as functions of time from separation, Figures 19, 20, and 21) are used in the aerodynamic and flight dynamic analyses.

Figure 13. Payload Axial Acceleration vs. Time
(± 30 g Accelerometer)

Figure 14. Payload Axial Acceleration vs. Time
(± 50 g Accelerometer)

Figure 15. Flight Altitude History

Figure 16. Flight Velocity History

Figure 17. Air Temperature vs. Altitude

Figure 18. Air Density vs. Altitude

TABLE VI
RCRTV PAYLOAD SEPARATION CONDITIONS

Velocity	4770 ft/s
Mach number	4.3
Flight path angle	14.6°
Altitude	9765 ft
Dynamic pressure	19210 lb/ft ²
Roll rate	5.44 cycles/s

Figure 19. RCRTV Velocity History

Figure 20. RCRTV Mach Number History

Figure 21. RCRTV Dynamic Pressure History

Flight Dynamics Analysis

Flight Motion Data

The flight vehicle carried instrumentation to provide the body fixed angular velocity components, pitch rate, yaw rate, and roll rate p , q , r , and the body fixed components of translational acceleration A_x , A_y , A_z . In addition, the vehicle was equipped with a miniature attitude reference system (MARS) stable platform which provided the vehicle angular attitude, pitch, yaw, and roll θ , ψ , ϕ relative to a nonrolling (aeroballistic) coordinate system. This instrumentation is described completely in a previous section.

Though the motion data was obtained continuously from lift-off to impact, only the data from payload separation to parachute deployment are of interest in this section. Therefore, the motion data are presented here as a function of time from separation. The data reduction,* which included filtering (40 Hz) in most cases, is generally of high quality.

Shown in Figures 22 and 23 are the vehicle pitch and yaw attitude angles (θ and ψ) as obtained from the MARS platform, plotted as functions of time from separation. The data are given up to the time of parachute deployment. The pitch angle shown includes the effect of trajectory curvature. Also presented is the vehicle pitch angle as a function of yaw angle, in Figure 24 at $T = 0$ to 2.0 seconds and in Figure 25 at $T = 2.0$ to 6.0 seconds. The vehicle angular motion, which is essentially circular, diverges initially because of the effects of roll resonance; and it converges again after the vehicle breaks out of resonance.

The vehicle lateral acceleration histories (Figures 26 and 27) are shown as functions of time from separation. The normal (Z axis) acceleration, which reaches a peak of 20 g's at $T = 1.3$ seconds after separation, is about twice the level expected because the flight vehicle experienced resonance earlier in the flight and at a dynamic pressure higher than expected. The resultant lateral acceleration history is given in Figure 28.

The flight vehicle angular velocity components obtained by the tri-axis rate gyro are presented in Figures 29 and 30; and the resultant lateral angular velocity, in Figure 31. The influence of roll resonance is clearly evident in these data.

*The data reduction was provided by Mrs. L. S. Converse, Data Processing Division 8411.

Figure 22. Vehicle Pitch Angle From Stable Platform

Figure 23. Vehicle Yaw Angle From Stable Platform

Figure 24. Pitch Angle vs. Yaw Angle (T = 0. to 4. sec)

Figure 25. Pitch Angle vs. Yaw Angle (T = 4. to 8. sec)

Figure 26. Lateral Acceleration History

Figure 27. Normal Acceleration History

Figure 28. Resultant Lateral Acceleration History

Figure 29. Pitching Velocity History

Figure 30. Yawing Velocity History

Figure 31. Resultant Angular Velocity History

The vehicle roll rates obtained from the rate gyro and from the MARS stable platform, respectively, are shown as a function of time from launch in Figures 32 and 33. The roll rate from the rate gyro is measured directly, while that from the stable platform is obtained from the derivative of the roll angle as measured by the MARS. The roll rate variations obtained from both sources are in good agreement. From the expected zero rate during first-stage burn ($T = 0$ to 3.5 seconds), the booster spins up as the velocity and dynamic pressure increase during second stage burn. This spin-up continues through deployment of the aerodynamic drag brakes and separation of the payload, which at this point has a spin rate of approximately 1960 deg/s (5.44 cps). The vehicle then spins on up into resonance at $T = 8.5$ seconds from launch, attaining a maximum spin rate of approximately 2900 deg/s (8.05 cps). At this point, a roll spin-down begins as the vehicle locks into resonance, and would continue until it reaches a minimum rate of 1750 deg/s (4.86 cps) at $T = 11.0$ seconds. The roll control fins, however, counter this instability and pull the vehicle out of resonance. Though some overshoot of the roll rate is obtained as the terminal rolling velocity of 2150 deg/s (6.0 cps) is reached, the terminal rolling velocity is nevertheless in good agreement with the preflight design value.

Flight Motion Analysis

Since the angular attitude data measured by the MARS are relative to a nonrolling inertial reference system, the vehicle angular motion is governed by the tricyclic theory of angular motion developed by Nicolaidis.¹² For the special case of the inertial angular attitude data, the complex pitch angle is given by

$$\vec{\theta} = K_N e^{(\lambda_N + i\omega_N)t} + K_P e^{(\lambda_P + i\omega_P)t} + K_T e^{iP^t} + K_R \quad (1)$$

where

$$\vec{\theta} = \theta + i\psi \quad (\theta \text{ and } \psi \text{ are the pitch and yaw angles measured by the MARS})$$

The motion therefore consists of three rotating vector modes K_N (nutation), K_P (precession), and K_T (trim). The nutation and precession modes are transient amplitudes determined from initial conditions. These converge or diverge according to the damping factors λ_N and λ_P , and K_T is the magnitude of trim angle resulting from vehicle asymmetries. The K_R term includes the effects of trajectory curvature, instrument bias, and so on; the nutation and precession modes rotate with circular frequencies ω_N and

Figure 32. Vehicle Roll Rate History From Rate Gyro

Figure 33. Vehicle Roll Rate History Obtained From MARS Platform Data

ω_P , respectively; and the frequency of the trim component is equal to the roll rate p . All parameters of the tricyclic equation may vary with time.

The flight dynamic data are analyzed to determine the parameters (characteristic frequencies and damping factors) of the analytic model; then by means of these parameters, to determine the aerodynamic stability coefficients of the flight vehicle. This task is accomplished by means of a differential corrections curve-fitting technique,¹³ used to fit the model to the observed angular attitude data. Since the tricyclic equation is a quasi-steady solution to the linearized equations of motion, if one assumes that the spin rate, dynamic pressure, and so on, are constants, the fitting technique can be applied over small sections of data where these assumptions are satisfied. The methods presented in References 13, 14, and 15 are used in the flight dynamic analysis of the motion data.

Shown in Figures 34 and 35 are the modal amplitudes and frequencies resulting from a fit of Equation (1) to the MARS stable platform data θ, ψ . The complete tricyclic model was not applicable to the data for the entire trajectory. During resonance, the trim and nutational frequencies are the same; therefore, the two modes cannot be distinguished from one another. The maximum angle of attack is nearly 5 degrees at $T = 1.5$ seconds. Only the nutation component of Equation (1) can be fitted from separation to $T = 4.0$ seconds, at which time a slight divergence in the nutation amplitude occurs, and the precession mode also becomes excited. The latter quickly damps out, however; and from $T = 5.0$ seconds on, the trim amplitude can be distinguished. The nutation mode eventually damps out at about $T = 13.0$ seconds, and the steady-state trim angle is determined to be approximately 0.25 degree. The modal frequencies determined from the fitting process are shown in Figure 35.

Figure 34. RCRTV Tricyclic Modal Amplitudes From Stable Platform Data

Figure 35. RCRTV Tricyclic Modal Frequencies
From Stable Platform Data

The precession mode was only determined for a short period of time; however, if the nutation and trim frequencies are known, the precession frequency can be determined from the relation

$$\omega_P = -\omega_N + P \frac{I_x}{I} \quad (2)$$

This calculated precession frequency, shown in Figure 35, agrees well with the fit values at $T = 4$ to 6 seconds. The fitted trim frequency T , presented in Figure 35, is in good agreement with the measured roll rate determined from the rate gyro; and the roll lock-in period from $T = 0.5$ to 3 seconds is clearly evident.

The outputs of the lateral accelerometers may be used to reconstruct the vehicle body fixed angle of attack. The components of the body fixed angle of attack are given as follows:

$$\alpha = \frac{-WA_Z}{C_{N_\alpha} QS} \quad (3)$$

$$\beta = \frac{-WA_Y}{C_{N_\alpha} QS}$$

where

- α - angle of attack (deg)
- β - angle of sideslip (deg)
- W - vehicle weight (lb)
- A_Z - Z-axis (pitch) accelerometer (g)
- A_Y - Y-axis (yaw) accelerometer (g)
- Q - dynamic pressure (lb/ft²)
- S - reference area, vehicle base area (ft²)
- C_{N_α} - normal force coefficient slope (deg⁻¹)

This method assumes, of course, that the vehicle aerodynamic normal force coefficient slope C_{N_α} is known. The theoretical values of C_{N_α} , which are

used in the foregoing calculations, are discussed in the aerodynamic analysis section of this report. The angle-of-attack components determined from

the foregoing relations are given in Figures 36 and 37, respectively. The total angle of attack given by

$$\begin{aligned}\vec{\alpha} &= \sqrt{\alpha^2 + \beta^2} \\ &= \frac{W}{C_{N_\alpha} QS} \sqrt{A_Z^2 + A_Y^2}\end{aligned}\tag{4}$$

is presented in Figure 38. The angular orientation of the vehicle windward meridian is given by the aerodynamic roll angle from the following expression

$$\begin{aligned}\phi_A &= \tan^{-1}\left(\frac{\alpha}{\beta}\right) \\ &= \tan^{-1}\left(\frac{A_Z}{A_Y}\right)\end{aligned}\tag{5}$$

The aerodynamic roll angle is presented in Figure 39. From $T = 0.5$ to 2.5 , the windward meridian is nearly stationary at $\phi \cong \pi$, since the vehicle is locked into resonance. Then, as the vehicle starts to break out of resonance, the windward ray moves around the body and becomes oscillatory just before $T = 4$ seconds. The upper envelope of the angle-of-attack data is in good agreement with the stable platform data (and with the fit of the stable platform data), and it indicates a maximum angle of attack of about 5 degrees.

Aerodynamic Analysis

Aerodynamic Drag

The vehicle axial force coefficient may be obtained from the expression

$$C_A = \frac{-WA_X}{QS}\tag{6}$$

where

C_A - vehicle axial force coefficient

W - vehicle weight (lb)

Q - dynamic pressure (lb/ft²)

S - reference area (base area) (ft²)

A_X - X-axis (longitudinal axis) acceleration (g)

Figure 36. Body Fixed Angle of Attack

Figure 37. Body Fixed Angle of Sideslip

Figure 38. Body Fixed Resultant Angle of Attack

Figure 39. Aerodynamic Roll Angle

The output from the two on-board axial accelerometers (Figures 13 and 14) are used in Equation (6) to calculate the axial force coefficient. The resulting coefficients are shown in Figures 40 and 41 as functions of time from payload separation. In these calculations, C_{X_1} results from the ± 30 g accelerometer A_{X_1} , while C_{X_2} is determined from the ± 50 g accelerometer A_{X_2} . From $T = 0$ to 0.5 seconds, the data do not provide a correct determination of the vehicle axial force coefficient since the base pressure is strongly influenced by the presence of the booster. The coefficients, which peak at a Mach number of approximately 0.9, are determined through supersonic and transonic velocities and are presented as functions of flight Mach number in Figure 42. Data obtained from the two sources are in excellent agreement. Since the angles of attack are small throughout the trajectory, the axial force coefficient is equivalent to the vehicle drag coefficient. In addition, because of the small angles of attack and some minor variations of axial force coefficient with angle of attack for these Mach numbers, the measured value of C_A is considered equivalent to the zero angle-of-attack value C_{A_0} .

The influence of fins on the vehicle drag can be seen in Figure 43. The RVRTV flight data^{16, 17} were obtained on a vehicle configuration the same as that used for the RCRTV but without fins; and a reaction-type hot gas (hydrazine) roll control system was used to keep the angle of attack small and the spin rate under control. The RCRTV coefficients shown in Figure 43 are an average of the data C_{X_1} , C_{X_2} obtained previously from the two accelerometers. Throughout the Mach number range, the 4 fins generate a considerable increase in drag. The relative drag increase is perhaps better illustrated by presenting as a function of Mach number the ratio of C_{A_0} with fins to C_{A_0} without fins (Figure 44). The 4 fins increase the total drag by 35 percent at Mach 4. The drag increment drops to 13 percent near Mach 1 but returns to the 40 percent level at subsonic velocities.

Static Stability

A measurement of the vehicle static stability is given by the pitching moment coefficient derivative C_{M_α} . This is the slope of the pitching moment C_M versus angle-of-attack curve, and for static stability C_{M_α} must be negative. The values of the static stability coefficient are determined from the frequency response of the flight dynamic data by means of the relation

Figure 40. Vehicle Axial Force Coefficient
(From ± 30 g Accelerometer)

Figure 41. Vehicle Axial Force Coefficient
(From ± 50 g Accelerometer)

Figure 42. Axial Force Coefficient vs. Mach Number

Figure 43. Axial Force Coefficient With and Without Fins vs. Mach Number

Figure 44. Ratio of C_{A_0} (With Fins) to C_{A_0} (Without Fins) vs. Mach Number

$$C_{M_\alpha} = \frac{\omega_N \omega_P I}{Q S d} \quad (7)$$

where

C_{M_α} - pitching moment coefficient slope ($\partial C_M / \partial \alpha$) -- (rad^{-1})

ω_N, ω_P - modal frequency histories (rad/s)

I - vehicle lateral moment of inertia (slug-ft^2)

Q - frustum dynamic pressure (lb/ft^2)

S - vehicle reference area (base area) (ft^2)

d - vehicle reference diameter (base diameter) (ft)

The moment reference point is taken as the vehicle center of gravity. The modal frequency histories ω_N and ω_P used in the foregoing relation are those obtained from the fit of the tricyclic equation to the MARS platform data. These frequencies were determined in the previous section and are given as functions of time in Figure 35. When these values are used in Equation (7), C_{M_α} is determined as a function of time from separation, and

the results are presented in Figure 45. The static stability coefficient is represented as a function of flight Mach number in Figure 46 along with the C_{M_α} data on the flight vehicle without fins. These data are a combination

of wind-tunnel and free-flight results on the same configuration without fins. The free-flight data were obtained from the RVRTV flight test program.^{16,17} At Mach number of 3.0 and above, the static stability coefficient is found to be approximately 20 percent higher on the finned configuration than on the finless vehicle; and additional wind tunnel tests at higher Mach numbers verify the 20 percent differential on this configuration. The increase in static stability is obviously attributable to the fins and, as will be seen, is a result of an increase in static margin. As will be shown in a subsequent section, the vehicle asymmetries were still sufficient--even with the increase in stability--to provide sustained roll-resonance instability. The C_{M_α} data increase as Mach number decreases until just before $M = 1$, where

the stability drops off considerably. The vehicle therefore had excellent static stability throughout the flight; and in fact, the fins aided the stability significantly.

Figure 45. Static Stability Coefficient (C_{M_α}) History

Figure 46. Static Stability Coefficient With and Without Fins vs. Mach Number

Aerodynamic Center of Pressure

The static stability coefficient C_{M_α} , determined in the previous section, may be expressed in terms of the vehicle normal force coefficient and center-of-pressure location as follows

$$C_{M_\alpha} = C_{N_\alpha} X_{cp}/d \quad (8)$$

where

C_{M_α} - pitching moment coefficient slope ($\partial C_M / \partial \alpha$) -- (rad⁻¹)

C_{N_α} - normal force coefficient slope ($\partial C_N / \partial \alpha$) -- (rad⁻¹)

X_{cp} - distance from center of gravity to aerodynamic center of pressure (ft)

The normal force coefficient slope is presented as a function of Mach number in Figure 47. As shown, the experimental data¹⁸ are well represented by the aerodynamic characteristics computer program, HANDY,¹⁹ and the theoretical method of Reference 20. Additional wind tunnel data on finned configurations indicate that the normal force coefficients are unchanged by the addition of the fins. The normal force coefficients calculated with HANDY are therefore used directly in Equation (8) along with the static stability coefficients C_{M_α} to determine the aerodynamic center of pressure

X_{cp} . The aerodynamic center of pressure, as so determined, is expressed as a percentage of body length from the nose and is presented as a function of Mach number in Figure 48 along with the center-of-pressure locations for the vehicle without fins. The latter represent a combination of theoretical calculations^{19,20} and experimental data.¹⁸ Since the vehicle center of mass is at 63 percent of its length, the static margin without fins is 5.3 percent. The effect of the fins is therefore to increase the static margin to 6.2 percent at Mach numbers above 3.5. The center of pressure is seen to move aft (increasing static stability) as the Mach number decreases; thus a maximum static margin of 7.4 percent occurs at Mach 1.8, while at Mach 0.9, the minimum static margin of 3.8 percent is obtained.

Figure 47. Normal Force Coefficient (At $\alpha = 0^\circ$) vs. Mach Number

Figure 48. Aerodynamic Center-of-Pressure Location With and Without Fins vs. Mach Number

Dynamic Stability

The dynamic stability coefficients are determined from the aerodynamic damping factors λ_N and λ_P , which appear in Equation (1). Both damping factors are necessary for determining the pitch damping coefficient. Unfortunately, both values were determined only for a brief period at approximately $T = 4.0$ seconds, and even then the damping factors were not well determined. Consequently, the damping coefficients could not be obtained reliably. Nevertheless, the rapid convergence of the angle of attack after the vehicle breaks out of roll resonance (Figure 34) is ample evidence that the vehicle experienced good dynamic stability. A brief period of angular divergence, occurring at approximately $T = 5.0$ seconds after separation, is attributed to the vehicle passing through the sonic condition. At this time, the trim angle of attack, composed of both principal-axis tilt and center-of-gravity offset, increases. The trim due to cg offset increases significantly because of the large increase in drag at Mach 1, and the angle of attack subsequently converges.

The conclusion is that the vehicle exhibited good dynamic stability throughout the flight and that the fins produced no adverse effects on stability.

Rolling Moment Coefficient

The total rolling moment coefficient is given by the following relation:

$$C_{\ell_T} = \frac{\dot{p} I_x}{Q S d} \quad (9)$$

where

$\dot{p}$ - roll rate time derivative (rad/s^2)

I_x - vehicle roll moment of inertia (slug-ft^2)

Q - freestream dynamic pressure (lb/ft^2)

S - vehicle reference area -- base area (ft^2)

d - vehicle reference length -- base diameter (ft)

The roll rate from the rate gyro output was sectionally fitted with a cubic spline, and the fit was then differentiated to determine $\dot{p}$. The total rolling moment coefficients C_{ℓ_T} determined from Equation (9) are shown in

Figure 49. Included in the total rolling moment coefficients are the contributions of the vehicle asymmetries and the roll control system.

Figure 49. Total Rolling Moment Coefficient

The contribution to rolling moment coefficient due to the vehicle asymmetries (trim angle due to principal axis inclination and center-of-mass offset) may be computed from the lateral accelerometer history and the known preflight vehicle mass and inertia asymmetries. The asymmetry roll moment coefficient C_{ℓ_A} is given by the following:

$$C_{\ell_A} = \frac{W A_y z_0}{Q S d} - \frac{W A_z y_0}{Q S d} \quad (10)$$

where

- C_{ℓ_A} - nondimensional rolling moment coefficient due to vehicle asymmetries
- y_0, z_0 - vehicle center-of-mass offsets along body y and z axes, respectively (ft)
- W - vehicle weight (lb)
- A_y, A_z - vehicle lateral accelerations along body y and z axes, respectively (g)
- Q - freestream dynamic pressure (lb/ft²)
- S - vehicle reference area -- base area (ft²)
- d - vehicle reference diameter -- base diameter (ft)

The resulting coefficients are determined as a function of time from vehicle separation and are shown in Figure 50. These data are not reliable after about $T = 4.0$ seconds since the acceleration levels are at small values, thus yielding a small signal-to-noise ratio. The peak value of -0.0009 at $T = 3$ seconds is in good agreement with the preflight expected value. The rolling moments produced by the intentional vehicle asymmetries were therefore sufficient to produce a sustained roll resonance instability in the absence of a roll control system.

Theoretical Fin Aerodynamics

The variables and nomenclature used in describing the fin aerodynamic characteristics are shown in Figure 51. It has been determined both analytically^{21, 22} and experimentally^{23, 24, 25} that the roll coefficient provided by spin fins is a highly nonlinear function of the vehicle angle of attack and the roll orientation of the fins relative to the angle-of-attack plane. The nonlinearities are caused by the relatively large surface cross-flow angularity when the vehicle is at an angle of attack. The cross-flow angle combines with the geometric fin-cant angle to yield an effective fin angle of attack.

Figure 50. Roll Moment Coefficient Due to Vehicle Asymmetries

This effect produces an induced roll moment and an induced side-force and moment, both of which can have a deleterious effect on the vehicle roll rate and flight stability. 26, 27

Figure 51. Spin Fin Aerodynamics

The maximum cross-flow angle is shown in Figure 52 for sharp and blunt cones as a function of vehicle angle of attack. As shown, the maximum cross-flow angle is equivalent to the geometric fin-cant angle (about 10 degrees) for sharp cones and is greater by a factor of 2 or 3 than the geometric cant for blunt cones. The variation in cross-flow angle around the body is a sinusoidal function.

Figure 52. Theoretical Cross Flow Angle for Blunt and Sharp Cones

An analytic technique for determining the fin aerodynamic coefficients as functions of the angle of attack and roll orientation angle has been developed. The details and a description of the computer program, FANDY, which calculates the fin aerodynamics, are presented in Reference 22.

An example demonstrating the accuracy of the analytic technique is given in Figures 53 and 54. In these figures, the experimental roll moment coefficients for a two-finned vehicle from References 23, 24, and 25 are given along with theoretically determined values obtained from the FANDY program. Reference 26 has shown that the roll moment coefficient for a two-finned vehicle can be represented as follows:

$$C_1 = C_{1\delta}(\alpha)\delta + C_{1\phi}(\alpha, \delta)\sin 2\phi \quad (11)$$

Figure 53. Fin Roll Moment Coefficient Correlation (Fin Cant, $\delta = 0^\circ$)

Figure 54. Fin Roll Moment Coefficient Correlation (Fin Cant, $\delta = 10^\circ$)

The first term is due to fin cant (and vanishes for no fin cant), while the second arises from the roll orientation dependence (and vanishes for angle of attack equal to zero). Both terms are nonlinear functions of angle of attack. As Figure 53 shows, considerable roll torque exists (depending on roll orientation), even for the case of no fin cant. The roll coefficients are well represented by the above expression. The fin-cant term which is the control system output is independent of roll orientation, while the induced effect $C_{1\phi}$ is very nearly independent of fin cant. It is possible for the

induced nonlinearity to dominate the fin-cant roll moment and produce the flight instabilities described in References 26 and 27. This is particularly true for the more blunt configurations since the cross-flow angularity increases with increasing bluntness.

To minimize the induced nonlinearity, the two-fin configuration is modified to a four-fin system. The second pair of fins is fixed on the body at a zero cant angle, and the plane of this set is orthogonal to that of the first. This arrangement introduces an additional roll coefficient expansion similar to the previous expression, with the exception that the roll orientation for the second set is advanced by $\pi/2$. By adding the two expressions, the total roll moment coefficient is given by

$$C_{1T} = C_{1\delta}(\alpha)\delta + \overline{C}_{1\phi} \sin 2\phi \quad (12)$$

The amplitude of the term dependent on the roll angle is now reduced considerably because of the addition of the fixed fins, and it vanishes at zero fin cant. This analysis has been verified by extensive wind tunnel tests.

The theoretically determined fin cant and induced roll moment coefficients for the RCRTV (at a Mach number of 5) are presented in Figures 55 and 56, respectively. These values are obtained as functions of vehicle angle of attack from the FANDY computer program. A comparison is made in Figure 56 of the induced roll moments for the two-fin and four-fin configurations, and there is more than a factor-of-three reduction at the worst condition (maximum fin cant of 10 degrees). At zero fin cant, the value of $\overline{C}_{1\phi}$ is zero.

Effects of High Spin Rate

Since the payload flight is at low Mach numbers (low vehicle velocity), it is important to consider the effects of the relatively high spin rate as compared with low vehicle total velocity. The tangential velocity of the fin gives rise to an effective fin cant angle given by the following:

Figure 55. RCRTV Theoretical Roll Moment Coefficient Due to Fin Cant

Figure 56. RCRTV Theoretical Induced Roll Moment Coefficient

$$\delta_{\text{eff}} = \tan^{-1} \left(\frac{P z_f}{V} \right) \quad (13)$$

where

δ_{eff} - fin effective cant angle due to spin

P - vehicle spin rate (rad/s)

z_f - location of fin relative to vehicle longitudinal axis (ft)

V - vehicle total velocity (ft/s)

As the vehicle total velocity decreases, the effective cant angle becomes larger, and this effective fin cant produces a negative roll torque throughout the trajectory. For example, the effective fin cant for RCRTV at Mach 5 and with a spin rate of 6 cps is approximately 0.25 degree for each fin. On the basis of the theoretical fin aerodynamic characteristics, a spin-down roll moment coefficient of -1×10^{-4} is obtained. However at Mach 1, the effective cant is approximately 1.1 degree, thus producing an effective roll moment coefficient of $C_{\ell_\epsilon} = 4.4 \times 10^{-4}$. Although this effect is negligible for hypersonic reentry, it is important for the RCRTV flight.

Fin Aerodynamic Characteristics

The asymmetry effects may be removed from the total roll moment coefficient to obtain the contribution provided by the control system. The roll moment data, however, were not of sufficient quality to provide a well-defined control system coefficient history for the entire trajectory. Where the data were considered unreliable, trajectory simulations were used to match the measured roll history and obtain the roll moment characteristics. This procedure provided a roll control moment due to fin cant of C_{ℓ_δ} . It is

assumed that the roll moment coefficient provided by the control system can be expressed as outlined in the previous section. The fin cant coefficient C_{ℓ_δ} thus determined, is presented in Figure 57 as a function of flight Mach

number. Shown also in Figure 57 is the theoretically determined C_{ℓ_δ} . There is excellent agreement between the theory and the flight-derived C_{ℓ_δ} at the

higher Mach numbers (Mach 2.5 to 5). Although the theory predicts an increase in fin effectiveness with decreasing Mach number, the flight derived C_{ℓ_δ} remains constant down to Mach 2.5 and then decreases.

Figure 57. RCRTV Fin Cant Coefficient vs. Mach Number

The conclusion is that the theory accurately represents fin aerodynamic characteristics at the higher Mach numbers ($\text{Mach} \geq 2.5$) but cannot be extrapolated to the low, transonic Mach number range ($\text{Mach} \leq 2.5$).

Roll Control System Performance

In this section a description of the roll control system, its physical properties, and its flight test performance and evaluation are presented. In addition, trajectory simulations of the flight vehicle with and without the roll control system are discussed. These simulations incorporate the actual flight launch conditions and trajectory parameters, as well as the flight-derived aerodynamic characteristics presented previously.

Roll Control System Dynamics

Although the actual design of the spin-fin system is somewhat complex, it is considered a simple spring-mass damper. The complete differential equations of motion which govern the internal moving masses (and fin deflections) are given in detail in References 8 and 9. A simplified form of this model will perhaps facilitate a discussion of the system. If it is assumed that the two proof masses are equivalent and are properly positioned in the vehicle (no misalignment errors, etc.), the control mass position ρ is given by

$$\bar{m}\ddot{\rho} + C\dot{\rho} + k(\rho - \rho_0) = F_{\text{AERO}} - f + \bar{m}\rho P^2 \quad (14)$$

where

- $\bar{m}$ - effective control mass (slugs)
- C - effective damping coefficient (lb-s/ft)
- k - effective spring constant (lb/ft)
- ρ - control mass position relative to vehicle centerline (ft)
- ρ_0 - relaxed spring position (ft)
- P - spin rate (rad/s)
- F_{AERO} - aerodynamic feedback force on control mass (lb) (Due to fin aerodynamic center of pressure being noncoincident with fin hinge line)
- f - total friction force (lb)

The total friction force includes a term which is proportional to the vehicle axial acceleration, since under such a condition, the control masses are loaded against the housing cylinders. The geometric fin cant δ is determined from the mass position by the linear relation

$$\delta = \delta_0 - \phi_L(\rho - \rho_I) \quad (15)$$

in which δ_0 is the maximum fin deflection, ϕ_L is the fin-mass linkage constant, and ρ_I is the mass position at maximum fin deflection.

The roll control system is therefore considered as a lumped parameter model characterized by Equation (14) and effective values of spring constant, damping, and so on. An extensive ground test program was carried out to determine these parameters and to find out if the control system actually does conform to the second-order model.

Roll Control System Physical Properties

The ground test program and experimental data are described in detail in Reference 3, and the important roll control system performance parameters determined during the test program are summarized in Table VII. The design spin rate was found to be approximately 7.0 cycles per second. Since under axial loads the sensing masses are loaded against the housing cylinders, the total frictional force is dependent on axial acceleration. This induced frictional force was determined to be 0.13 pound per axial g (Table VII).

TABLE VII
ROLL-CONTROL SYSTEM PERFORMANCE PARAMETERS

Sensing mass	0.75 lb
Effective spring constant	98.4 lb/ft
Effective damping coefficient	5 lb-ft/s
Effective spring preload	8.0 lb
Total mass travel	1.0 in
Maximum fin deflection	10°
Minimum fin deflection	-7°
Effective linkage constant	17 deg/in
Static friction (1 g environment)	0.2 lb
Axial loading-dependent friction	0.13 lb/axial g
Design spin rate	7.0 cycles/s

Simulation tests²⁸ were also conducted in the Sandia, Livermore, Flight Dynamics Laboratory (FDL)²⁹ which permits an evaluation of the system response under conditions which closely simulate those encountered in actual flight. These were "system-in-the-loop" tests in which the control system test unit was mounted on a CARCO* angular motion simulator whose flight table was driven by the hybrid-computer solution of the trajectory equation of motion. The facility permits a direct comparison of the dynamic response of the actual control system with that given by the mathematical model of the system, and this series of simulation tests verified that the system is accurately represented by the second-order lumped-parameter model.

Recovery of the flight vehicle permitted postflight roll control system testing, which indicated essentially no change in the properties listed in Table VII.

Uncontrolled Flight Performance

In the absence of roll control, certain types of asymmetric flight vehicles will exhibit a sustained roll resonance instability (roll lock-in), an instability characterized by a roll rate equal to the vehicle natural pitch frequency (P_{crit}) and by a divergence of the total angle of attack. To ensure similar behavior, the RCRTV flight vehicle has built-in mass and inertial asymmetries (nominally a 0.25 degree principal axis inclination and a 0.10 inch center-of-mass offset normal to the inertial asymmetry), as indicated in Table I. It is the roll torque produced by these combined asymmetries that tends to stabilize the roll rate about the natural pitch frequency (sustained roll resonance). The purpose of the RCRTV flight test program, then, is to demonstrate the capability of the spin-fin roll control system to control the roll rate of a vehicle with such asymmetries.

The RVRTV program,^{16, 17} on the other hand, is a continuing study by Sandia Laboratories to investigate roll-pitch coupling under simulated reentry conditions. To date, the following flight tests have been conducted with vehicles having an external configuration the same as that of the RCRTV and dealing with: (1) aerodynamic and mass asymmetries to produce roll lock-in, (2) aerodynamic and mass asymmetries with a hydrazine reaction jet roll control system, and (3) inertial and mass asymmetry to produce roll lock-in. These flights verified the analytic and simulation techniques used to determine the vehicle flight performance.

To establish the vehicle flight performance without roll control, the flight-derived aerodynamic data presented previously, the vehicle physical properties, and the payload separation conditions are used in the SPINFIN

*CARCO Electronics Model S440-R3 three-axis flight table.

trajectory simulation computer program. The flight parameters for the uncontrolled trajectory are illustrated in Figures 58 and 59. In Figure 58, where vehicle spin rate P and natural pitch frequency P_{crit} appear as functions of flight time, the vehicle spins down to near zero roll rate, thus demonstrating a stable roll lock-in. During this time, the total angle of attack, shown in Figure 59, diverges to an amplitude of about 60 degrees. Thus it is quite clear that with no roll control system, the flight vehicle would have experienced sustained roll lock-in.

Figure 58. RCRTV Spin Rate and Natural Pitch Frequency for Uncontrolled Trajectory

Figure 59. RCRTV Total Angle of Attack for Uncontrolled Trajectory

Control System Flight Performance

The important variables which characterize the vehicle and roll control system flight dynamic performance (spin rate p , critical pitch frequency P_{crit} , total angle-of-attack α , and fin cant δ) are presented in Figures 60, 61, and 62. The spin rate (Figure 60) is that obtained from the rate gyro, while the critical pitch frequency is computed from the relation

$$P_{crit} = \sqrt{\frac{-C_{M_\alpha} Q S d}{I - I_x}} \quad (16)$$

by using the previously derived static stability coefficient C_{M_α} and the flight dynamic pressure history. Since the initial spin rate of 34 rad/sec is considerably below the system design value of 44 rad/sec, a rapid increase in spin rate takes place; and this increase causes the vehicle to experience resonance early in the flight ($t \cong 0.8$ second), resulting in an early divergence of the total angle of attack (Figure 61). This occurrence is followed by a prolonged spin-down from $T = 0.8$ to $T = 3.3$ seconds, during which time the spin rate is locked into resonance and decreases to about 32 rad/sec. At $T = 3.3$ seconds, the control system causes the spin rate to recover to the nominal value of 44 rad/sec, and the angle of attack converges. The flight fin cant history is presented in Figure 62.

Figure 60. Flight Vehicle Spin Rate and Natural Pitch Frequency History

Figure 61. Flight Vehicle Total Angle-of-Attack History

Figure 62. Flight Vehicle Fin Cant History

On the RCRTV flight, a difference between the expected and observed performance was obtained. The discrepancy arose from two sources: (1) differences between the expected and actual payload separation conditions, and (2) differences between the predicted and flight-derived fin and body aerodynamic characteristics. In order to construct the postflight simulation, these effects must be taken into account. Thus they are discussed in the following pages.

Flight Simulations

The results of a postflight simulation, initiated at the actual payload separation conditions (Table VI) and using predicted fin and vehicle aerodynamic characteristics, are presented in Figures 63 and 64. A comparison of the vehicle rolling velocity obtained from the flight and from the simulation is shown in Figure 63. The vehicle does not immediately spin up, following payload separation, even though the fins are canted at approximately 7 degrees. The belief is that pressure disturbances due to the proximity of the booster at this time can produce a significant change in the pressure profile over the vehicle surface and in the neighborhood of the fins. Similar effects are apparent in the axial force coefficients shown in Figures 41 and 42. Thus until the payload gets free of the booster and fully establishes its wake, the fin characteristics may be greatly affected. In any case, the delay in initial spin-up is incorporated in the simulation, and the spin rate obtained (Figure 63) is in good agreement with the flight values until about 2.0 seconds after separation. At this time the simulated rolling velocity breaks out of resonance, is stabilized at 44 rad/sec at $T = 3.5$ seconds, and is maintained thereafter.

A comparison of the simulated with the flight test angle-of-attack upper envelope history (Figure 64) shows good agreement in both the initial divergence and the peak amplification. An early convergence of the angle of attack is obtained with the simulation, as expected, due to the early breakout of the simulated spin rate from resonance.

The initial spin-up is attributable primarily to the low separation roll rate. However, the spin-down after $T = 2.0$ seconds, which is not accounted for, is due to a decrease in fin aerodynamic effectiveness, which is in turn attributable to the flight Mach number variation.

A second postflight simulation was performed in which the flight-derived fin aerodynamic characteristics (Figure 57) are employed, and the results are presented in Figures 65, 66, and 67, with the flight test values included for comparison. In each case the agreement is excellent between the simulation and the flight test for spin rate, for angle of attack, and for fin cant history. The conclusion is that the SPINFIN simulation, which uses the actual flight separation conditions and the flight-derived aerodynamic characteristics, accurately represents the flight performance.

Figure 63. Spin Rate and Natural Pitch Frequency (Preflight Simulation with Flight Separation Conditions)

Figure 64. Total Angle of Attack (Preflight Simulation With Flight Separation Conditions)

Figure 65. Spin Rate and Natural Pitch Frequency (Postflight Simulation with Flight Derived Aerodynamics)

Figure 66. Total Angle of Attack (Postflight Simulation with Flight Derived Aerodynamics)

Figure 67. RCRTV Flight Simulation from Steady State Starting Conditions--Spin Rate and Natural Pitch Frequency

Roll Control System Evaluation

Before an actual second-resonance encounter, the vehicle would be at a steady-state rolling velocity with the fins at a near-zero cant angle. Therefore, in order to evaluate properly the control system performance during second resonance, a simulation was performed in which the system was initially at the design condition. In such a manner, the adverse effects of the initial separation conditions were eliminated, thus making possible a more meaningful evaluation of performance. The simulation, which also used the flight-derived fin and vehicle aerodynamic characteristics, was initiated at the separation conditions listed in Table VI, except that the initial spin rate was set at 7.2 cycles/second. The results of the simulation are summarized in Figures 67, 68, and 69.

There was considerably less roll excursion during simulation than there was during the flight. Although the angle-of-attack amplification was the same as the flight value, convergence was obtained much sooner because the roll rate was controlled. Therefore, the conclusion is that most of the spin rate variation obtained on the flight occurred because the roll control system had to recover from a spin rate considerably below the operating value. Further, on a typical full-scale reentry trajectory, there is considerably less roll rate deviation from the design value if the system is at steady-state operating conditions before the resonance encounter occurs.

Figure 68. RCRTV Flight Simulation from Steady State Starting Conditions--Total Angle of Attack

Figure 69. RCRTV Flight Simulation From Steady State Starting Conditions--Fin Cant

Thermodynamic Analysis

Thermodynamic Data

The flight vehicle thermodynamic instrumentation consisted of calorimeters which measure heat flux on the cone surface and fin leading edges. One calorimeter was located on the leading edge of each fixed fin (oriented at 90 degrees and 270 degrees clockwise from the vertical). The cone surface heating calorimeters were located at two stations along the axis of the body. The first two were located at the fin station 4.75 inches forward from the vehicle base at orientations 135 degrees and 315 degrees clockwise from the vertical. The second pair were similarly oriented but were positioned 14.5 inches forward from the vehicle base. In addition to this heating instrumentation, four thermocouples were included to measure shell backface temperature. Two of them were mounted at each of the two axial stations described above, and at each station, they were oriented at 45 degrees and 225 degrees clockwise from the vertical. In addition to the foregoing sensors, there was passive thermodynamic instrumentation in the form of heat-sensitive (temperature-sensitive) Tempilaq paint used on certain portions of both the interior and exterior of the flight vehicle. Tempilaq tints sensitive to various temperatures were placed on the top and sides of each fin, on the fin platform, and also on the cone surfaces aft of the fins and on either side of each fin. A detailed description of the Tempilaq placement is given in Reference 5.

The cone surface heating at the fin station (4.5 inches from the base) is given in Figures 70 and 71, and the heating obtained at the forward station (14.5 inches from the base) is provided in Figures 72 and 73. A peak of 90 BTU/ft²-s was obtained from each calorimeter at the fin station at about $T = 8$ seconds after launch. This is, as expected, slightly less than the 100 BTU/ft²-s maximum obtained at the forward station. The transmission from one of the forward calorimeters (Q5, Figure 73) was not reliable during the peak heating and is therefore not used.

The data obtained from the fin leading-edge calorimeters (Figures 74 and 75) are in excellent agreement and indicate peak heating of about 425 BTU/ft²-s just before $T = 8$ seconds after launch.

The flight vehicle shell backface temperature at the fin station is shown in Figures 76 and 77. The two thermocouple outputs are in excellent agreement and show a maximum temperature of about 300°F at $T = 11$ seconds after launch. The data from the forward station thermocouples (Figures 78 and 79) appear somewhat questionable, however, from $T = 8.0$ to 11.0 seconds. Nevertheless, the maximum temperature at $T = 11.0$ seconds appears to be about 330°F.

Figure 70. Cone Surface Heat Flux at Fin Station
(4.75 in From Base @ 135° Orientation)

Figure 71. Cone Surface Heat Flux at Fin Station
(4.75 in From Base @ 315° Orientation)

Figure 72. Cone Surface Heat Flux at Forward Station
(14.5 in From Base @ 135° Orientation)

Figure 73. Cone Surface Heat Flux at Forward Station
(14.5 in From Base @ 315° Orientation)

Figure 74. Fin Leading Edge Heat Flux (90° Orientation)

Figure 75. Fin Leading Edge Heat Flux (270° Orientation)

Figure 76. Cone Shell Backface Temperature at Fin Station
(4.75 in From Base @ 45° Orientation)

Figure 77. Cone Shell Backface Temperature at Fin Station
(4.75 in From Base @ 225° Orientation)

Figure 78. Cone Shell Backface Temperature at Forward Station
(14.5 in From Base @ 45° Orientation)

Figure 79. Cone Shell Backface Temperature at Forward Station
(14.5 in From Base @ 225° Orientation)

Aerothermodynamic Analysis

Preliminary analysis for the prediction of the aerodynamic heating of fins on reentry vehicles is discussed in detail in References 30 through 33. These methods were used for the preflight analysis and prediction of fin and cone heating for RCRTV. As a result of an extensive aerothermodynamic wind-tunnel testing program on various fin geometries, a more comprehensive analytic model of fin heating has been developed. This analysis³¹ was used for postflight predictions of fin leading-edge heat transfer coefficients which were then used as boundary conditions in a two-dimensional thermal model of the fin leading-edge calorimeters in order to obtain the fin heat flux. Results of the computational procedure are compared with the heat flux data obtained from the two fin leading-edge calorimeters (Q3 and Q4) in Figure 80. Good agreement was obtained, thus verifying the analytical technique, the details of which are given in Reference 32.

Figure 80. Comparison of Actual and Theoretically Determined Fin Leading Edge Heat Flux

A similar approach was used for calculating the cone heating. Heat transfer coefficients were predicted by means of the SLAP aeroheating computer program.³³ Results of the calculations for the two body stations where calorimeter data were obtained are compared with the data in Figures 81 and 82. Again the data are well represented by the theory.

Figure 81. Comparison of Actual and Theoretically Determined Cone Surface Heat Flux at Forward Station (14.5 in From Base)

Figure 82. Comparison of Actual and Theoretically Determined Cone Surface Heat Flux at Fin Station (4.75 in From Base)

As outlined in the first section of this report, the aerothermodynamic objectives were to obtain fin leading-edge and cone surface thermodynamic data (heat flux and temperature) and to verify analysis methods developed in support of the flight program. As indicated by the data obtained, these objectives were successfully accomplished, and the theoretical methods were fully verified.

Tempilaq Experiment

In order to obtain qualitative thermodynamic information on fins, passive thermodynamic instrumentation, Tempilaq,³⁵ was used on both the interior and exterior of the flight unit. Tempilaq is an inexpensive heat-sensitive (temperature-sensitive) paint, various formulations of which change phase at predetermined critical temperatures. Tempilaq of various formulations was applied on the top and sides of the fins, on the fin platforms, on the cone surface aft of the fins, and on either side of each fin. This experiment yielded important and inexpensive qualitative information concerning the nature of the flow field and heat transfer in the vicinity of the fins. Examples of the Tempilaq patterns before and after the flight are shown in Figures 83 through 86.

Figure 83. Preflight Fixed-Fin Tempilaq Pattern

Figure 84. Preflight Moving-Fin Tempilaq Pattern

Figure 85. Postflight Fixed-Fin Tempilaq Pattern

Figure 86. Postflight Moving-Fin Tempilaq Pattern

Summary and Conclusions

The RCRTV flight test was conducted in order (1) to demonstrate the capability of the spin-fin roll control system to control the roll rate of an asymmetric flight vehicle through a roll resonance encounter, and (2) to obtain detailed flight performance data to verify, and to compare with that obtained from the analysis procedures used. The flight test data and post-flight analyses have led to the following conclusions:

1. All flight subsystems performed properly. The payload was boosted to the desired separation conditions. The booster drag brake system and payload separation system functioned nominally with no detectable separation disturbances imparted to the vehicle. The on-board instrumentation and payload

telemetry system performed properly and yielded high-quality data from each sensor. The payload parachute recovery system functioned as planned, thus allowing the vehicle to be recovered intact and permitting postflight investigation and analysis to be made. Also, because the flight system employed in the RCRTV program represents a unique and relatively inexpensive means for low-altitude flight testing, it constitutes an attractive test bed for flight verification of proposed systems.

2. The flight vehicle trajectory was reconstructed from good-quality radar, optical tracking, meteorological, and telemetry data. The payload flew the desired trajectory, which simulated a severe second-resonance encounter and yielded a good test of the roll control system performance.
3. Good-quality data was obtained from the payload MARS stable platform, tri-axis rate gyro, and linear accelerometers. These data made possible a detailed investigation and analysis of the reentry vehicle flight dynamics. The flight motion was analyzed by means of the linear tricyclic theory. Well-determined histories for the characteristic model amplitudes and frequencies of the angular motion were obtained from the MARS data. Reconstruction of the angle-of-attack history from the lateral accelerometers confirmed the MARS data analysis.
4. The flight dynamics analysis led to a determination of the various aerodynamic coefficients. These included the aerodynamic drag, pitching moment, center of pressure, and fin roll moment characteristics. The effects of fins on the body aerodynamics were determined. Of particular interest are the following:
 - a. The four fins increase the total drag coefficient by 35 percent at Mach 4. The drag increment drops to +13 percent near Mach 1 and returns to +40 percent at subsonic velocities.
 - b. An increase in static stability (C_{M_α}) of 20 percent was observed at Mach numbers of 3.0 and greater. The conclusion was that this increase is due solely to an increase in static margin. At Mach numbers greater than 3.0, the static margin is 6.2 percent for the finned vehicle, as opposed to 5.3 percent for the configuration without fins.
 - c. The vehicle exhibited good dynamic stability throughout the flight. No adverse effects on dynamic stability were attributed to the fins.

- d. The rolling moment coefficient due to the vehicle asymmetries was concluded to be more than sufficient to cause roll lock-in. The roll control moment coefficient due to fin cant ($C_{l\delta}$) was found to be in excellent agreement with the theoretical values for Mach numbers of about 2.5 and greater. The theory cannot be extrapolated to Mach numbers below about 2.5.
- 5. The roll control system performed properly, and it completely satisfied all mission requirements. The system exhibited good stability, and it controlled the spin rate as desired, thus demonstrating fully its capability to control a severe roll resonance instability.
- 6. Postflight simulations revealed:
 - a. In the absence of roll control, the vehicle would have exhibited a sustained roll resonance instability, with the total angle of attack amplifying to 60 degrees or greater.
 - b. The postflight simulation, which incorporated the actual payload separation conditions and the flight-derived aerodynamic characteristics, closely matched the flight performance parameters. Thus the analytic models of the control system, aerodynamics, and so on, as well as the simulation program, accurately represented the physical system.
- 7. Good quality aerothermodynamic data were obtained, including cone and fin leading-edge heat flux, as well as cone temperature. These data verified the fin heat transfer theory which was developed in support of the program. The Tempilaq experiment demonstrated the feasibility of using temperature-sensitive paint and yielded inexpensive qualitative information on fin aerothermodynamics.
- 8. All program and flight test objectives were successfully accomplished. In addition, the aerodynamic, aerothermodynamic, and flight simulation analysis methods for finned reentry vehicles were fully verified.

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