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TECHNICAL NOTE

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VANGUARD SATELLITE SPIN-REDUCTION MECHANISM

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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VANGUARD SATELLITE
SPIN-REDUCTION MECHANISM

by
Robert C. Baumann
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SUMMARY

The Cloud Cover Satellite flown in Vanguard vehicles SLV-3 and SLV-4 required a spin rate of 55 r.p.m. when entering orbit. Since the third-stage rocket was spin-stabilized in flight, and because other considerations required that the satellite remain attached long enough to acquire more than the desired 55 r.p.m., a satellite spin-reduction mechanism was developed. Although the mechanisms functioned properly in both flights, the desired spin rate was not achieved owing to uncontrollable flight effects. These effects make the prediction of satellite spin rates after a long pre-separation coasting period extremely difficult. To meet future requirements a control system is needed which can orient a payload according to a predetermined scheme and maintain that orientation for the desired period.

CONTENTS

Summary i

INTRODUCTION 1

REDUCTION OF SPIN 2

PERFORMANCE OF THE MECHANISMS 2

  Satellite Launching Vehicle Three (SLV-3) 2
  Satellite Launching Vehicle Four (SLV-4). 4

CONCLUSIONS AND RECOMMENDATIONS 6

ACKNOWLEDGMENTS 7

Appendix A - Weights and Moments of Inertia of Cloud Cover Satellites. 18

Appendix B - Weights of Spin-Reduction Mechanism Components. 19

Appendix C - Tabulation of SRM Data for Vehicles SLV-3 and SLV-4. 20

VANGUARD SATELLITE SPIN-REDUCTION MECHANISM

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INTRODUCTION

During the Vanguard satellite launching program, tests performed on the third-stage rocket* at Tallahoma, Tennessee, indicated that residual burning of the solid propellant might be expected to last as long as 147 seconds after the nominal full-thrust burning time of 30 seconds. This residual burning would produce a small erratic thrust from the motor. The differential (separation) velocity imparted between satellite and third-stage rocket by the separation mechanism† was a nominal 3 feet per second. Since residual burning was capable of increasing the third-stage velocity by 10 to 30 times that amount, a post-separation collision between third stage and satellite was obviously possible if separation were to be effected 30 seconds after burnout as planned. To insure against such a collision, a longer "coasting time" was necessary before separation.

Accordingly, the satellite separation devices were modified to delay separation until approximately 5 minutes after third-stage ignition.† The third-stage forward bearing system had been designed to provide a satellite rotation rate of approximately 55 to 60 r.p.m. at the time of normal separation (30 seconds after burnout). The USASRDL Cloud Cover satellite (Figure 1 and Appendix A) required an initial spin rate of approximately 55 r.p.m. in order to perform its mission properly. Since the satellite would now remain attached longer than 30 seconds after burnout, the spin rate would increase as a function of the no-load friction torque in the bearing assembly. From test data the no-load friction torque was determined to be 0.028 inch-pound for the bearing assembly used in launch vehicle SLV-3, and 0.042 inch-pound for the assembly used in SLV-4. Calculations based on these friction torques indicated that the satellites would reach much higher pre-separation spin rates than those desired. It was therefore necessary to reduce the spin rate to a value within the desired limits.

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*Manufactured by the Grand Central Rocket Company. Another version of the third-stage rocket, manufactured by the Allegany Ballistics Laboratory, was employed in some of the later Vanguard launchings.

† Baumann, R. C., "Vanguard Satellite Separation Mechanisms," NASA Technical Note D-497, 1960

REDUCTION OF SPIN

A spin-reduction mechanism (SRM) was designed utilizing the principle parts of the existing Vanguard satellite separation mechanisms for actuation (Figures 2, 3, and 4; and Appendix B). (This mechanism can also be used to increase the spin rate of a satellite.) Actuation of the SRM is accomplished in a manner identical to that of the standard short-time Vanguard separation mechanism, as follows: If an acceleration of approximately 12 g or more is applied for 2 seconds, the timer runs for 10 seconds more to a stop on the g-weight arm; but if 12 g or more is applied for *less* than 2 seconds, the unit resets itself. After the acceleration falls below the 12-g level at third-stage burnout, the timer starts and runs for approximately 30 seconds. At the end of this period the timer arm closes circuits to the wire-wound squibs (containing a small powder charge) in the caterpillar motors. The powder ignites, expanding the bellows of the caterpillar motors, which in turn rotate a circular T-shaped part located in the center of the spin-retarding mechanism. A slot in the T is thus aligned with a pin press-fitted through the piston shaft; the shaft contains an O-ring that seals the high-pressure sphere. With the restraint removed, the piston is forced out by compressed nitrogen in the sphere; nitrogen then flows through two external tubes perpendicular to the spin axis and 180 degrees apart, and expands through two small jet nozzles located 16 inches apart. The nozzles are so oriented that the resulting torque opposes the rotational torque imparted to the satellite through the bearing assembly, thus reducing the spin rate.

PERFORMANCE OF THE MECHANISMS

Satellite Launching Vehicle Three (SLV-3)

Predicted Flight Performance

Prediction of the satellite spin rate was based on bearing-friction data supplied by The Martin Company. Using these data in conjunction with other parameters made it possible to calculate the satellite spin rates. The information shown in Figure 5 was utilized to predict the necessary spin reduction. Separation mechanism no. 15 and the "best" bearing were used for the SLV-3 flight; the anticipated total spin was 104.4 r.p.m. Therefore it should have been necessary to remove approximately 50 r.p.m. to achieve the desired 55 r.p.m. at satellite separation.

The pressure sphere of SRM flight unit no. 1 was charged with 6.7 grams of dry nitrogen gas. A laboratory calibration curve (Figure 6) showed that with this charge the satellite spin rate should be reduced by 49 ± 2 r.p.m. This mechanism was installed directly to the forward portion of the third-stage forward spin bearing. On top of the SRM were mounted a 20-inch-diameter radiation shield (peculiar to the Cloud Cover satellite), the satellite separation mechanism (long-delay), and the satellite (Figure 7).

Actual Flight Performance

To evaluate the performance of the spin-reduction mechanism, it was necessary to obtain the SLV-3 flight data records from the Minitrack stations at the Air Force Missile Test Center (Florida), Grand Turk and Antigua (B.W.I.), and South Africa. Careful study and interpretation of the Minitrack AGC records permitted a proper determination of the satellite spin rate as a function of time, based upon the number of nulls per unit time in the AGC trace. The physical position of the satellite with respect to the tracking stations determines the null rate obtained; e.g., from the Air Force Missile Test Center there were two nulls per revolution, while from Grand Turk and Antigua there were four nulls per revolution.

The data obtained from the Minitrack record are shown in Figure 8. Superimposed on this plot is the preflight predicted curve of spin rate with time. It is readily seen that the spin rate at the time of satellite separation was much higher than was anticipated. The first portion of the curve (through SRM actuation) follows the predicted curve reasonably closely. In the final coasting period the spin rate increases more rapidly.

The increase in coasting-time spin rate over the predicted rate might be attributed to some combination of the following: (1) Thermal expansion of bearing assembly parts (due to heat conduction from the third stage) causing the built-in clearances in the bearing assembly to be taken up, hence increasing the no-load friction torque; (2) loading of the bearing by acceleration due to residual burning of the third-stage rocket; (3) loading of the bearing due to wobbling of the third-stage rocket-satellite combination; and (4) friction variation with rotation rate. Of these four possibilities the largest variation would probably be caused by thermal expansion of the bearing assembly components and by wobbling. As can be seen in Figure 8, the no-load friction torque μ_{NL} during the final coasting period went from 0.038 to approximately 0.05 inch-pound, considerably higher than the predicted 0.028 inch-pound.

The spin-reduction mechanism was actuated at the proper time and reduced the satellite spin rate by approximately 56 r.p.m. This excess of some 11 percent over the predicted spin reduction did not result from any malfunction of the SRM: It was found subsequently that the vacuum chamber exployed in the laboratory had caused the SRM to operate at reduced efficiency during calibration. This will be explained in the discussion of the SLV-4 flight which follows.

Although the final satellite spin rate exceeded the desired 55-r.p.m., valuable information was obtained for evaluating the spin-reduction mechanism, the satellite separation mechanism, and the forward bearing assembly. This information makes possible a much closer prediction of the final satellite spin rate under normal flight conditions.

Satellite Launching Vehicle Four (SLV-4)

Predicted Flight Performance

After a careful study of all available data on bearing friction (Figure 9), SRM performance, and vehicle performance, a proposed curve for the SLV-4 flight was formulated (Figure 10). The bearing assembly to be used was recalibrated in the field prior to flight and an increase in friction parameters was evidenced (Figure 9). The SLV-4 flight curve was again formulated with these field measurements, and the results are superimposed in Figure 10. This curve indicated that it was necessary to remove about 78 r.p.m. in order to obtain the desired 55 r.p.m. at separation. The time to satellite separation was shortened from 315 seconds to 244 seconds to reduce the overall satellite spin-rate increase to a level that could be handled by the existing SRM. Spin reduction for the SLV-4 flight was based on the field version of the predicted spin rate curve.

Actual Flight Performance

The SRM for the SLV-4 flight was charged with 11.2 grams of nitrogen gas (approx. 2500 p.s.i.). With this charge, the laboratory calibration curve indicated a reduction of the satellite spin rate by 78 r.p.m. Flight data indicate, however, that a reduction of 94 r.p.m. actually occurred. A series of tests was conducted to explain the large difference.

It was found that a loss of about 5 percent had occurred during the laboratory calibration of the SRM because of retarding forces exerted on the nozzle-arms by gas that was deflected back by the bell jar used as a vacuum chamber An additional 3- to 4-percent loss in calibration could be attributed to the axial load of the inertia disc used for calibration; this axial load increased the friction of the bearing as compared with the no-load condition. A third unknown that could cause an increase in SRM performance is heating of the SRM nitrogen during flight by conduction and radiation. An increase from 70° to 150°F would produce approximately 7 percent more impulse.

A flight curve has been added to the calibration data (Figure 6); this curve will be used to guide future reduction predictions. Although the SRM reduced the spin 16 r.p.m. more than was intended in the SLV-4 flight, it is believed that this error can be accounted for in the future and that the necessary allowances can be made. The SRM was actuated properly at the prescribed time.

A composite curve (Figure 11) has been plotted to show predicted and actual results in the SLV-3 and SLV-4 flights. The SLV-4 curves indicate that the predicted and actual results are in good agreement up to 75 seconds (SRM actuation), after which the flight curve departs from the predicted curve in several ways. The predicted slope has been added for comparison purposes. Even with the -28 r.p.m. at 75 seconds, a satellite spin rate of approximately +27 r.p.m. would be expected at the time of satellite separation under normal conditions. Such was not the case; the satellite spin rate at separation was smaller than -1 r.p.m. about the spin axis. At or shortly after separation, however, the satellite received sufficient impulse to cause an additional rotation of approximately 15 r.p.m. about an axis believed to be through the equator.

The following theory provides a possible explanation of the causes and effects of what happened prior to, during, or shortly after satellite separation:

As was pointed out earlier, one of the possible causes of frictional variation is the variation of rotational rate: As the rotation rate decreases, the friction torque also decreases. Since extremely low values of friction torque are the concern here, it is reasonable to assert that the torque could have decreased from the predicted average no-load value of 0.048 inch-pound to an average no-load value of 0.025 inch-pound after SRM actuation and prior to satellite separation. If this did occur, the low spin rate about the x-x (spin) axis at separation is explained.

The source supplying this friction torque, the empty third stage, was rotating in the direction opposite that of the satellite. The satellite, therefore, had to decrease in spin rate, pass through zero, and pick up spin rate in the same direction as the third stage. If the satellite had remained attached for an additional 170 seconds, it probably would have reached a spin rate about the x-x axis of approximately +28 r.p.m.

The satellite, owing to its low spin rate, had little or no spin stability about the x-x axis at the time of separation from the third stage. The satellite had to travel approximately 4-3/8 inches on the separation sleeve to become free. With a small wobble angle of the third-stage rocket-satellite combination (in the order of 2 or 3 degrees) and the comparatively high third-stage spin rate (in the order of 3 r.p.s.), a tipping impulse at separation (only approx. 0.25 lb-sec was needed) could have caused the 15-r.p.m. satellite rotation about another axis, probably y-y. With this additional tipping impulse, the satellite could have gone into a complex motion consisting of spin, precession, and perturbation.

The most likely cause of the spurious rotation, however, is a collision between the third stage and the satellite after the initial separation. Optical tracking data conclusively show a third-stage orbital velocity exceeding that of the satellite by some 200 feet per second. Thus, clearly, residual burning occurred in the rocket after satellite separation.

CONCLUSIONS AND RECOMMENDATIONS

With the SRM system described, pertinent flight data, and accurate bearing friction data, under normal conditions of flight, it is now possible to obtain a desired spin rate within 15 percent, by use of the long-delay separation approach. By proper care and calibration of the bearing assemblies, the friction parameters can be defined. Bearing loading after third-stage burnout can be attributed to a number of uncontrollable effects such as wobble, temperature changes, and residual burning. If the satellite spin-axis moment of inertia is low and the third stage and satellite reach an equilibrium rotation condition, other unknowns are introduced that further complicate the situation: For example, the third-stage spin rate in the Vanguard TV-4 flight increased approximately 30 r.p.m. during the burning period.

It can therefore be concluded that reasonably accurate satellite spin-rate predictions can be made only to the end of the third-stage burning. During the final coasting phases (prior to separation), the uncontrollable unknowns cause wide variations from the predicted values. The SRM itself has functioned reliably at the prescribed times, and has reduced the satellite spin rate by predictable amounts in Vanguard flights SLV-3 and SLV-4. A tabulation of spin-reduction data from these two flights is given in Appendix C.

To assure a proper satellite spin rate about a preselected axis in the future, use of the following system is recommended: (1) After separation, use a yoyo-type device such as is used by JPL* to reduce the spin rate to as near zero as possible; (2) use a system of jets located so as to impart rotation about the desired axis; and (3) design inertial stability into the payload about the desired spin axis.

For payloads of reasonable size, the SRM system described herein could give reliable and accurate spin rates. There is of course no substitute for a control system which can orient the vehicle or payload with respect to a predetermined scheme and maintain this orientation for the required period. Pending the development of such a system, a lightweight reliable system must be provided to establish certain basically required motions such as rotation, or the lack of it. By means of integrating centrifugal switches, a power supply, and proper valving to a device operating on the SRM principle, it is possible at present to maintain spin-rate control about a preselected axis for payloads of moderate size, provided the mass-inertia distribution is proper.

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*Wells, W. H., and McDonald, W. S., "Satellite Spin Reduction," Jet Propulsion Laboratory inter-office memo dated July 1, 1958

ACKNOWLEDGMENTS

The assistance and support of the individuals and groups which made the rapid development, use, and evaluation of the SRM possible, is gratefully acknowledged: The U.S. Naval Research Laboratory, the Diamond Ordnance Fuze Laboratories, The Raymond Engineering Laboratories, and many personnel of the Goddard Space Flight Center. Mr. N.C. Schaller prepared the SLV-3 spin-bearing data. Messrs. J. T. Shea, J. E. Bush, F. T. Martin, J. C. MacFarlane, and LTJG A. Simkovitch greatly assisted in the preparation of this report.

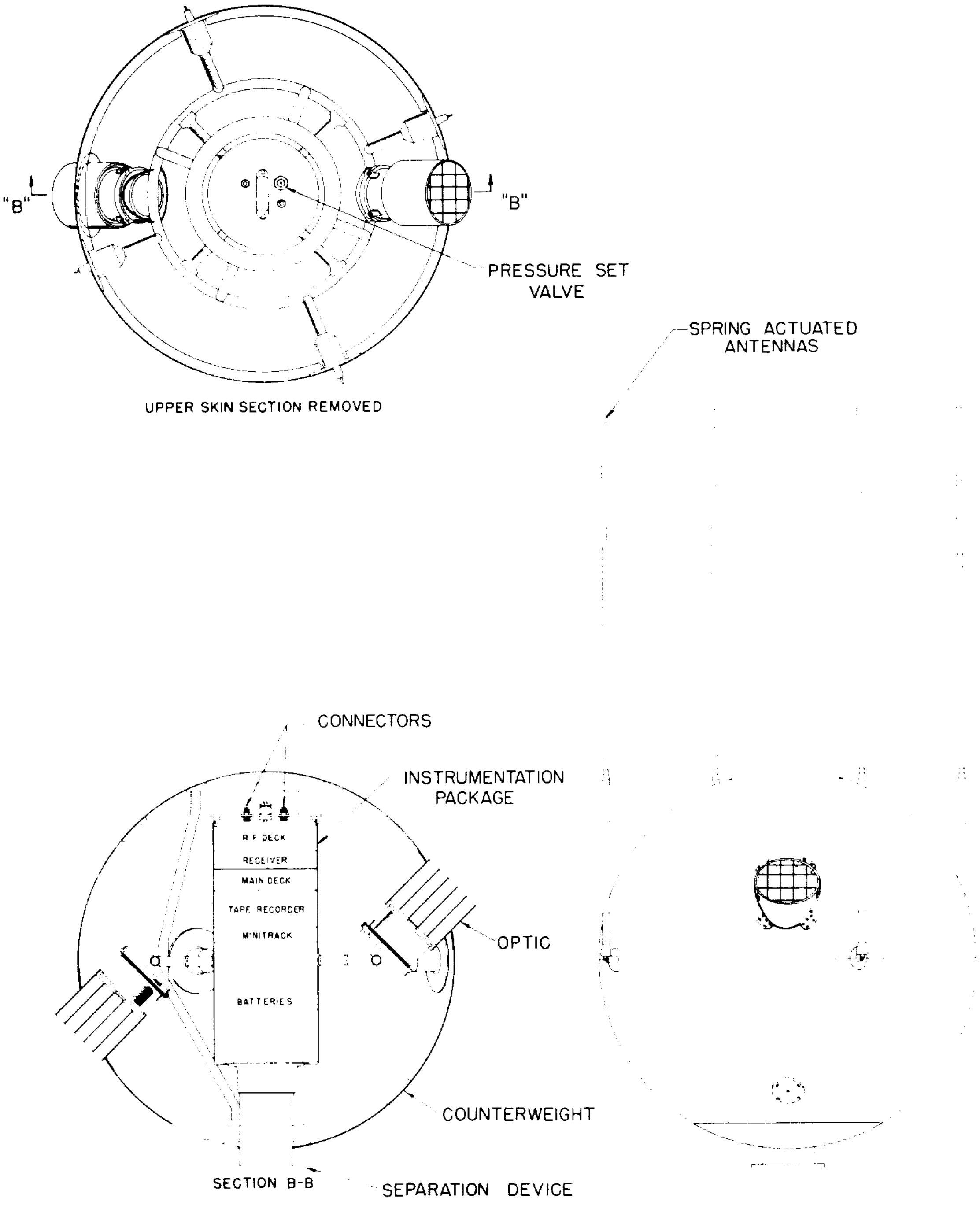
"B"
"B"
PRESSURE SET VALVE
UPPER SKIN SECTION REMOVED
SPRING ACTUATED ANTENNAS
CONNECTORS
INSTRUMENTATION PACKAGE
R F DECK
RECEIVER
MAIN DECK
TAPE RECORDER
MINITRACK
BATTERIES
OPTIC
COUNTERWEIGHT
SECTION B-B
SEPARATION DEVICE

Figure 1 - 20-Inch-diameter Cloud Cover Satellite

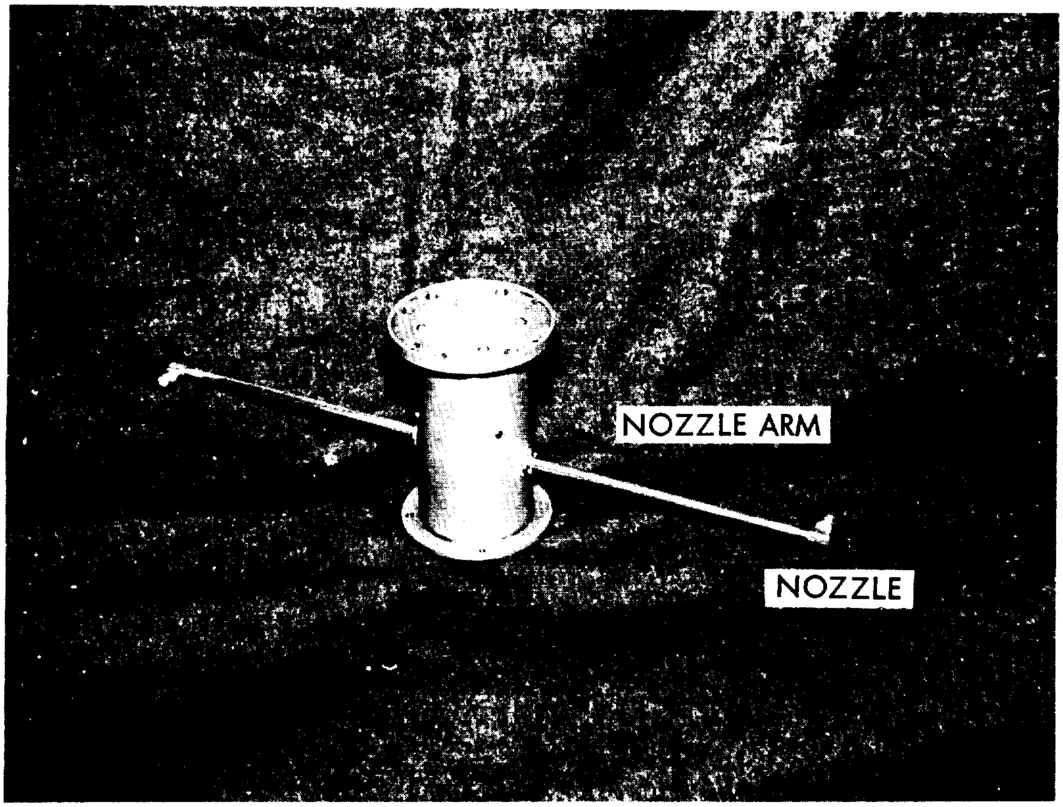
NOZZLE ARM
NOZZLE

Figure 2 - Spin-reduction-mechanism assembly

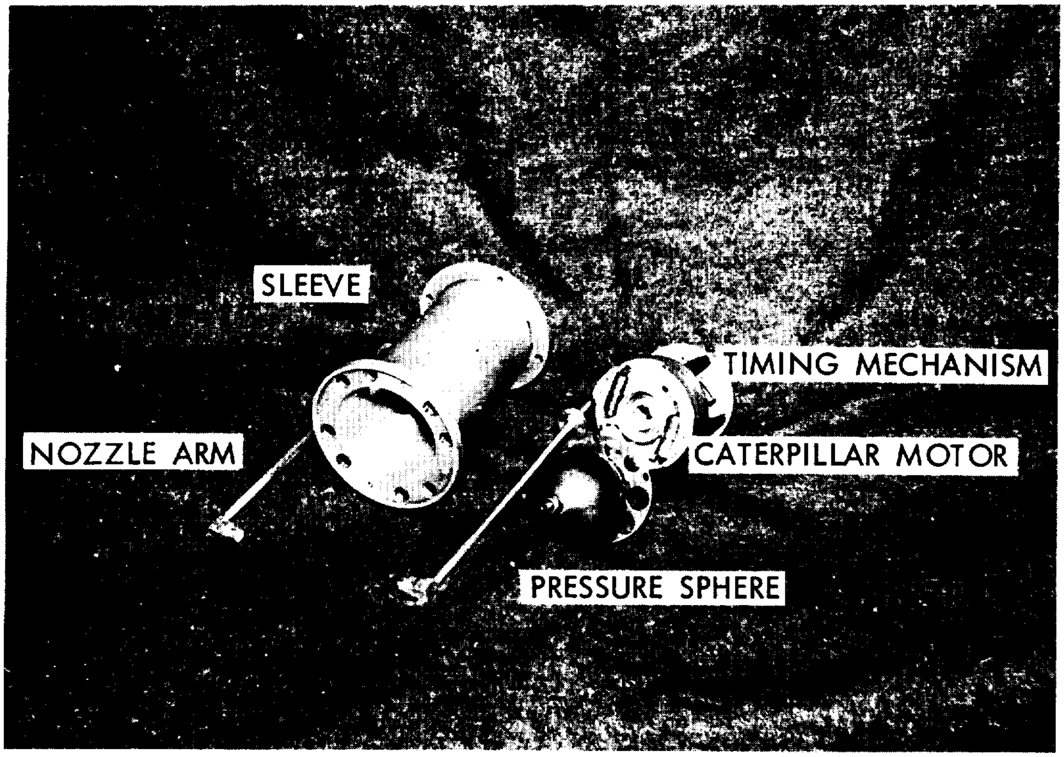
SLEEVE
TIMING MECHANISM
NOZZLE ARM
CATERPILLAR MOTOR
PRESSURE SPHERE

Figure 3 - Spin-reduction-mechanism components

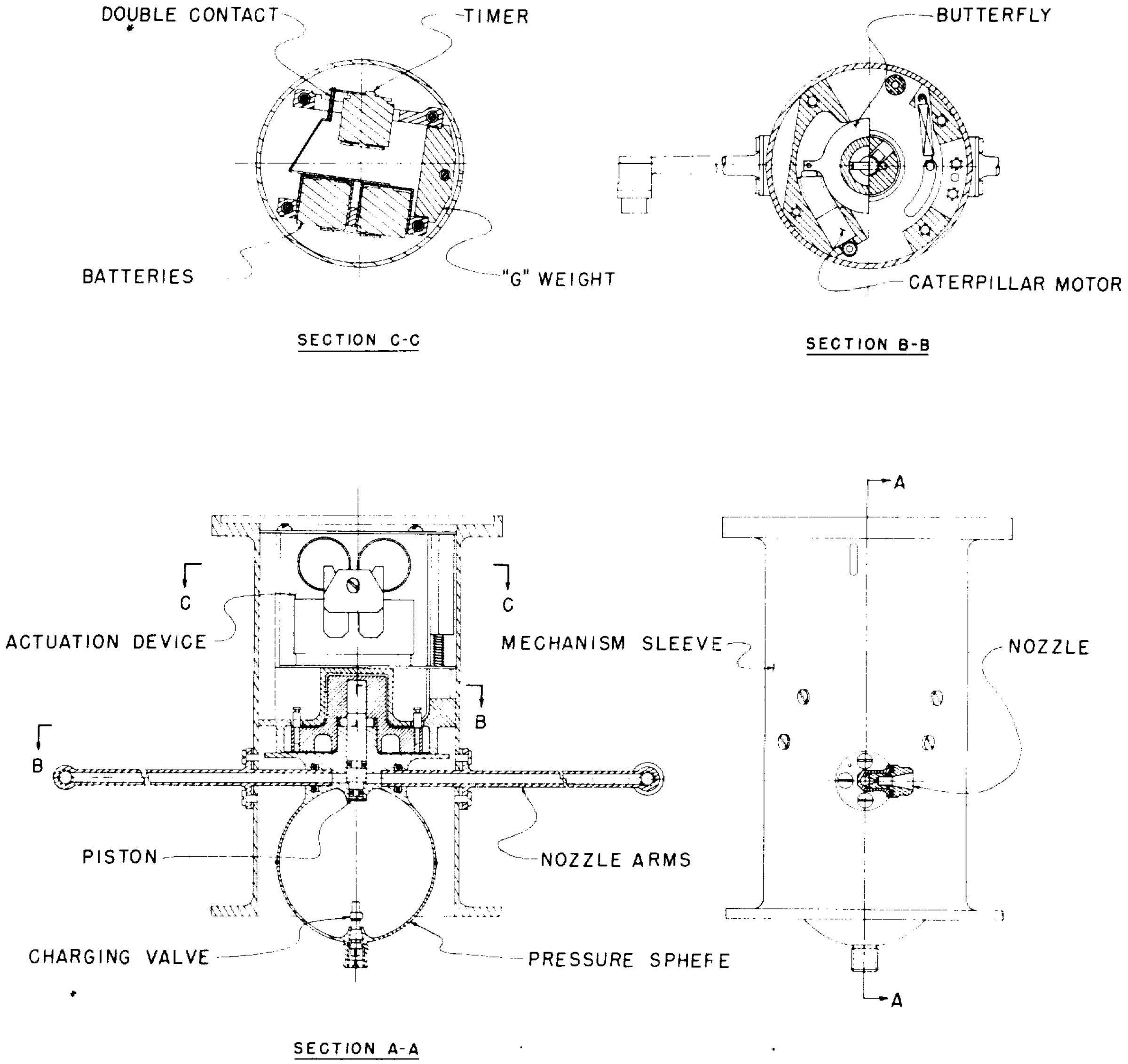
DOUBLE CONTACT
TIMER
BATTERIES
"G" WEIGHT
SECTION C-C
BUTTERFLY
CATERPILLAR MOTOR
SECTION B-B
C
C
ACTUATION DEVICE
MECHANISM SLEEVE
B
B
PISTON
NOZZLE ARMS
CHARGING VALVE
PRESSURE SPHERE
SECTION A-A
A
A
NOZZLE

Figure 4 - Spin-reduction mechanism

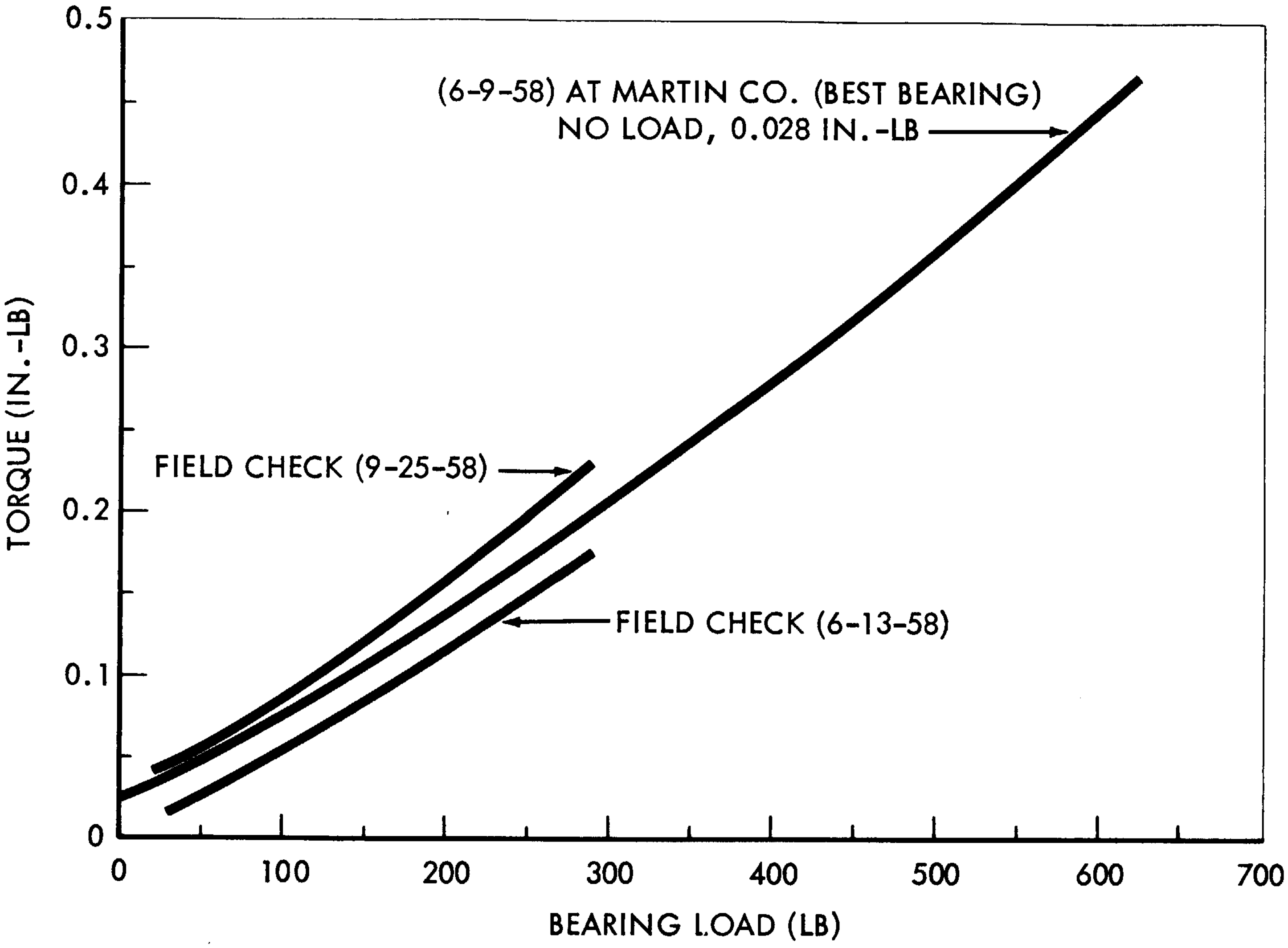

Figure 5 - SLV-3 bearing data

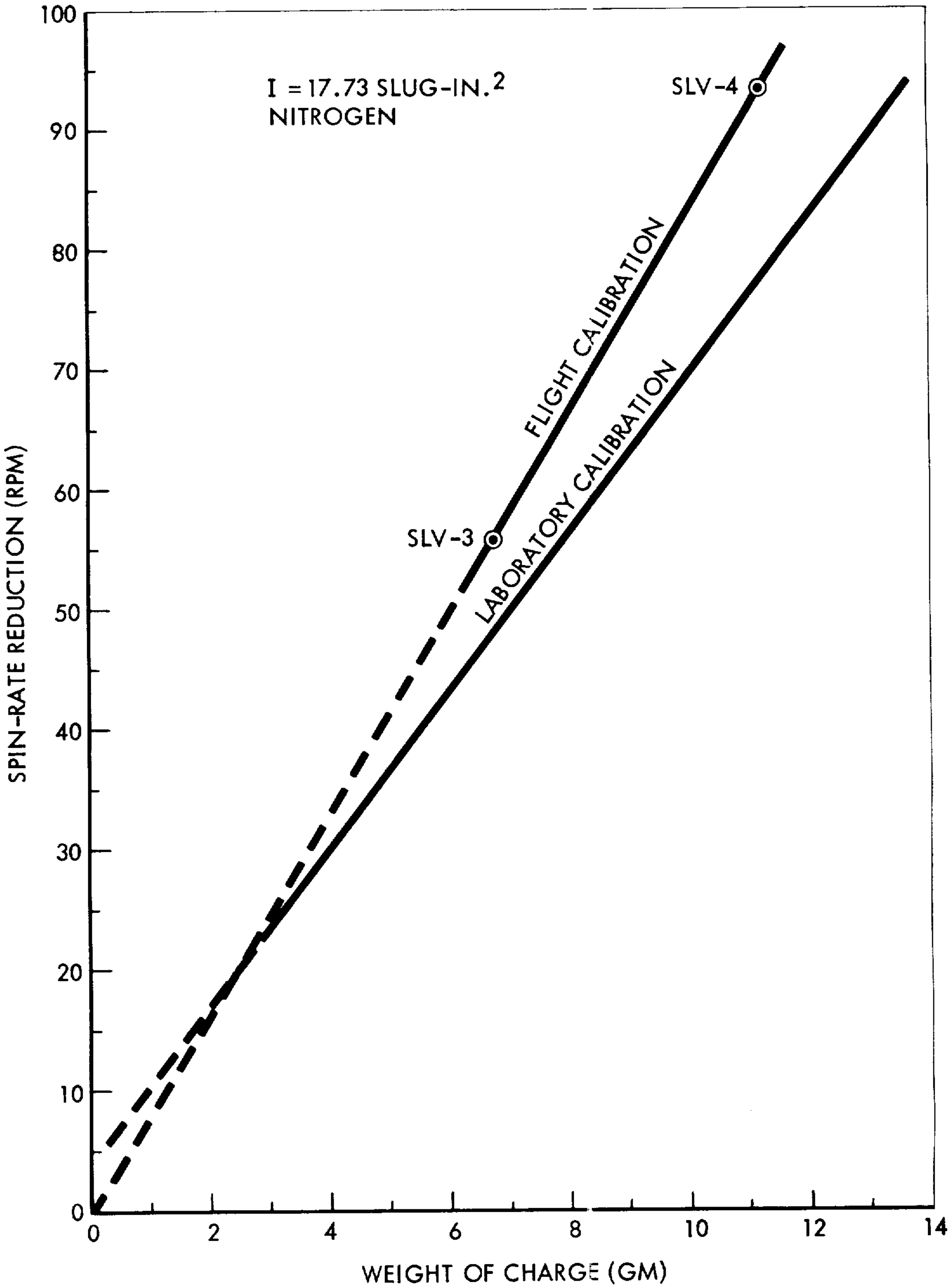

Figure 6 - SRM calibration curves

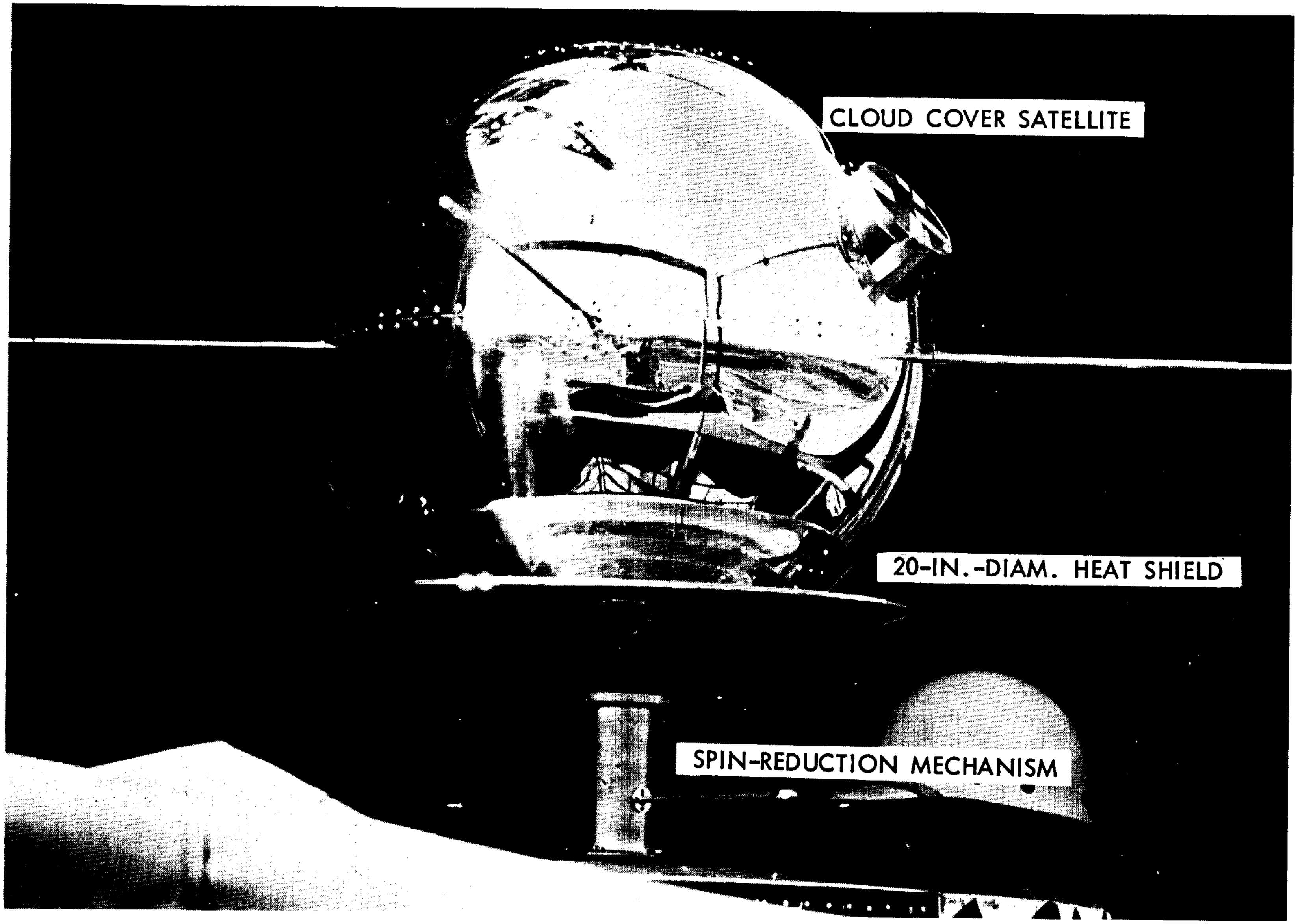
CLOUD COVER SATELLITE
20-IN.-DIAM. HEAT SHIELD
SPIN-REDUCTION MECHANISM

Figure 7 - Assembly of spin-reduction mechanism to SLV-3

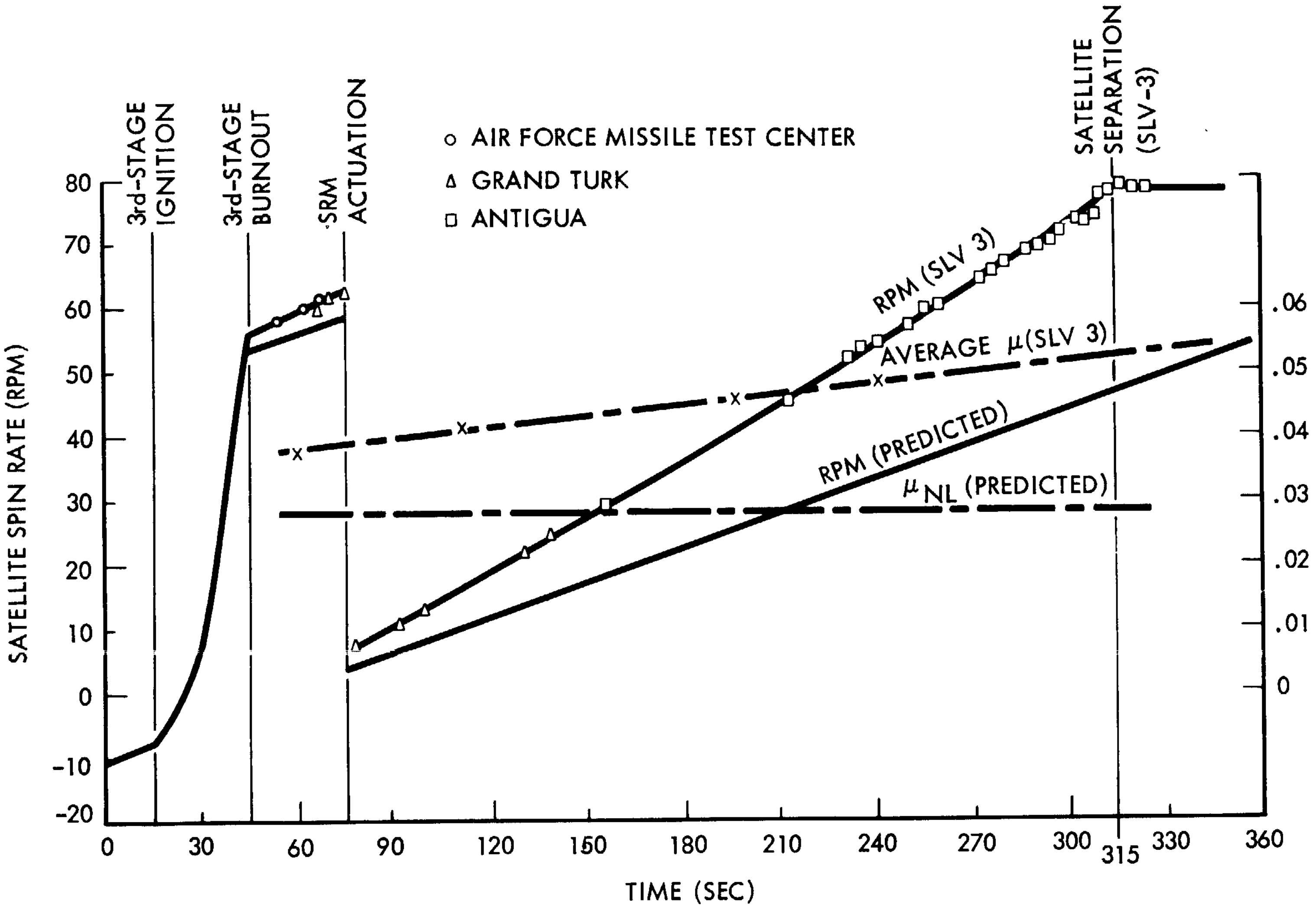

Figure 8 - SLV-3 flight data

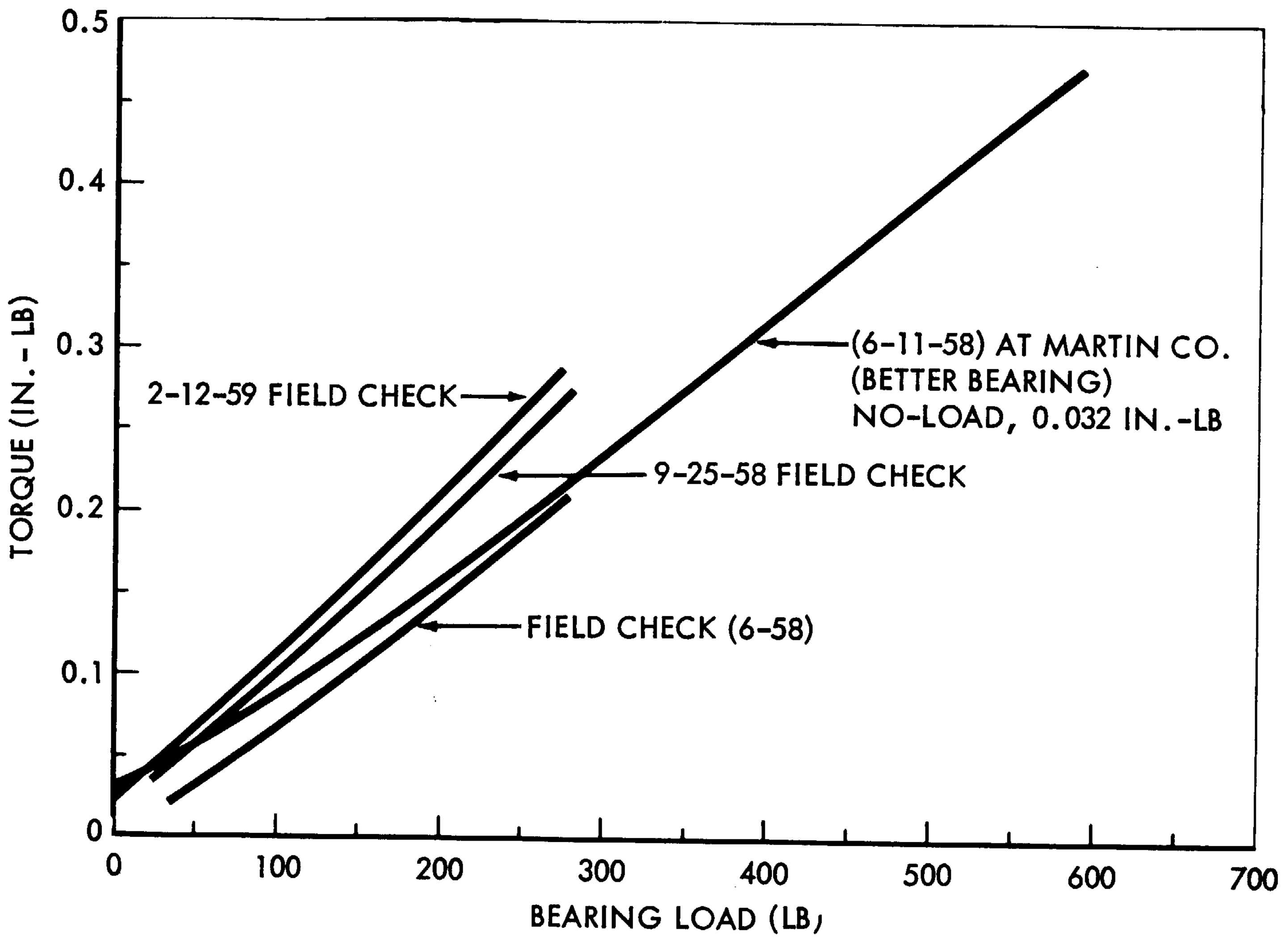

Figure 9 - SLV-4 bearing data

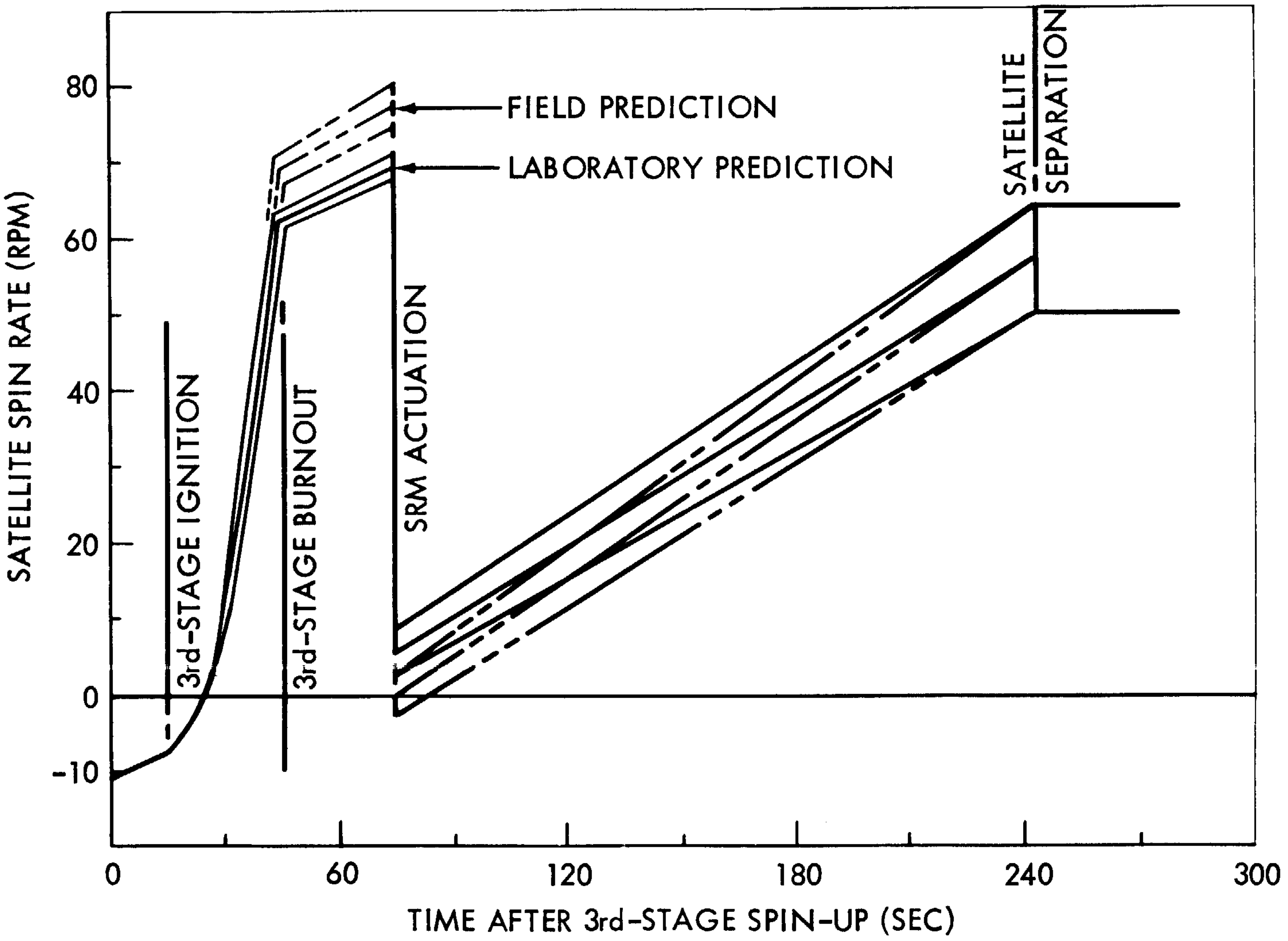

Figure 10 - Predicted SLV-4 curves

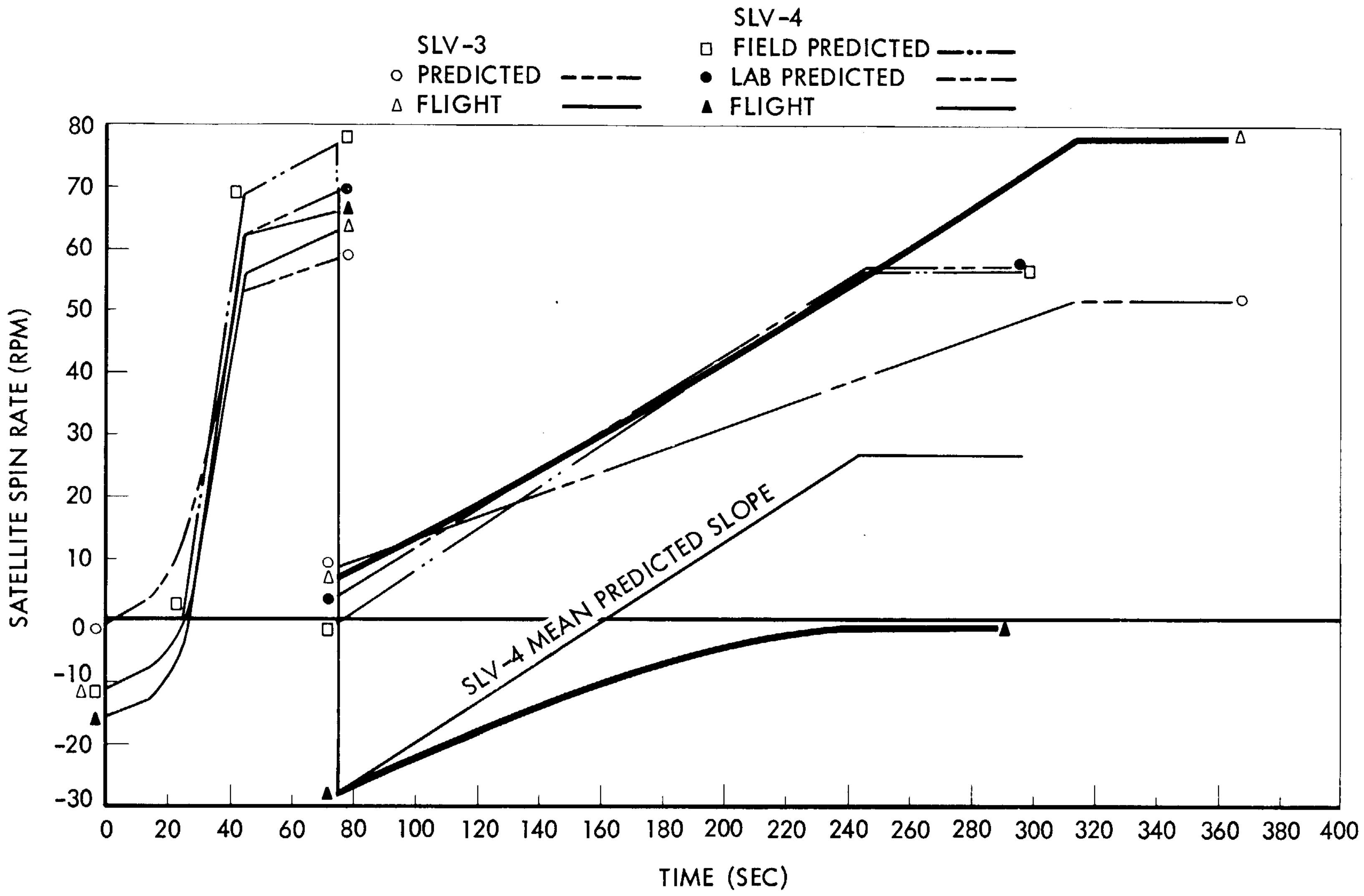

Figure 11 - SLV-3 and SLV-4 predictions and flight data

Appendix A

WEIGHTS AND MOMENTS OF INERTIA OF CLOUD COVER SATELLITES

Item	Moment of inertia about spin axis (slug-in.^{2})	Weight (lb.)
Satellite Flight Units:		
No. 1 (SLV-4)	16.96 (18.94*) (20.18†)	20.74
No. 2 (SLV-3)	–	20.9
No. 3 (spare)	–	20.77
Spin Reduction Mechanism	0.10	1.10
Satellite Separation Mechanism (long-delay)	0.02	0.90
20" Diameter Heat Shield	0.65	0.40
Total Contributed by Satellite and Attached Components:		
SLV-3	17.73	23.3 ± .03
SLV-4	17.73	23.17 ± .03

*Moment of inertia about equator 1-3.
†Moment of inertia about equator 2-4.

Appendix B

WEIGHTS OF SPIN-REDUCTION-MECHANISM COMPONENTS

The following weight breakdown is based on actual weights of representative flight-unit components. The prototype-unit weights are approximately 1.25 pounds (less balancing weight), or 0.15 pound heavier than the flight unit.

Component	Number required	Weight (gm.)	Weight (lb.)
Sleeve	1	103.0	0.2271
Squib retainer	1	35.9	0.0791
Bottle	1	119.5	0.2634
Butterfly	1	15.7	0.0346
Stem	2	21.4	0.0471
Orifice	2	2.0	0.0044
Saddles	2	1.4	0.0031
Studs for spring	2	0.2	0.0004
Springs	2	0.4	0.0009
Piston and pin	1	2.3	0.0051
High-pressure valve stem	1	1.0	0.0022
"O" Ring 2-8	2	0.2	0.0004
"O" Ring 2-6	2	0.2	0.0004
"O" Ring 2-4	2	0.2	0.0004
No. 2-56 Screw and washer	8	0.2	0.0004
No. 6-32 Screw and washer	4	1.1	0.0024
2 Batteries, 2 squibs, and timer mechanism	1	182.0	0.4012
Nitrogen charge	2000 p.s.i.g.	8.5	0.0187
TOTAL WEIGHT			1.0913

Appendix C

TABULATION OF SRM DATA FOR VEHICLES SLV-3 AND SLV-4

Function	Average μ_{NL} (in.-lb)	Average μ_L (in.-lb)	Duration (sec)	Satellite spin rate (r.p.m.)	Variation from predicted (r.p.m.)
2nd Stage					
SLV-3 Predicted	—	—	—	0	—
SLV-3 Flight	—	—	—	-11	-11
SLV-4 Lab. predicted	—	—	—	-11	—
SLV-4 Field predicted	—	—	—	-11	—
SLV-4 Flight	—	—	—	-15.3	-4.3
1st Coast[†]					
SLV-3 Predicted	0.028	—	15	+3	—
SLV-3 Flight	0.038	—	15	-8	-11
SLV-4 Lab. predicted	0.038	—	15	-8	—
SLV-4 Field predicted	0.042	—	15	-7	—
SLV-4 Flight	0.0206	—	15	-13	-5** -6***
3rd-Stage burning					
SLV-3 Predicted	—	0.257	30	53	—
SLV-3 Flight	—	0.329	30	56	+3
SLV-4 Lab. predicted	—	0.35	30	62 ±1.5	—
SLV-4 Field predicted	—	0.38	30	69 ±2	—
SLV-4 Flight	—	0.386	30-37.5	62	0** -7***
2nd Coast[‡]					
SLV-3 Predicted	0.028	—	30	58	—
SLV-3 Flight	0.045	—	30	62.5	+4.5
SLV-4 Lab. predicted	0.038	—	29	69 ±1.5	—
SLV-4 Field predicted	0.042	—	29	77 ±2	—
SLV-4 Flight	0.0206	—	30	66	-3** -11***
Spin reduction					
SLV-3 Predicted	—	—	—	49.5*	—
SLV-3 Flight	—	—	—	56.0*	+6.5
SLV-4 Lab. predicted	—	—	—	65*	—
SLV-4 Field predicted	—	—	—	78*	—
SLV-4 Flight	—	—	—	94*	— +16***
3rd Coast[§]					
SLV-3 Predicted	0.028	—	240	47	—
SLV-3 Flight	0.045	—	240	78	+31
SLV-4 Lab. predicted	0.048	—	163	57 ±7	—
SLV-4 Field predicted	0.053	—	168	56 ±7.5	-58**
SLV-4 Flight	0.0247	—	169	-1	-57.5***

[†] The period from initial third-stage spinup to third-stage ignition.
[‡] The period from third-stage burnout to SRM actuation.
[§] The period from SRM actuation to satellite separation.
*Amount of actual spin reduction by the SRM.
**Variation from laboratory prediction.
***Variation from field prediction.

NASA - Langley Field, Va.