

## CHAPTER VII

### TEST PROGRAM

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#### SECTION I - INTRODUCTION

A test program was set up early in the Explorer XIII program to simulate as nearly as possible all expected environmental conditions to which the spacecraft would be subjected during the ascending phase of the trajectory and while in orbit. These tests included spin, acceleration, thermal-vacuum, vibration, and shock. Functional tests were performed on individual modules of the telemeter system.

The test procedure started with breadboard construction of telemeter components. Circuits were evaluated in this phase, and components chosen for prototype construction. Prior to the prototype environmental tests, it was found desirable to construct a mock-up structure for preliminary evaluation of the spacecraft hardware.

#### SECTION II - ENVIRONMENTAL TEST SPECIFICATIONS

##### A. Flight Qualification (Prototype Payload).-

1. Spin: The prototype payload shall be subjected to a spin rate of 225 rpm, which is  $1\frac{1}{4}$  times the expected flight value. This spin rate shall be achieved in 1 second and held for 15 minutes. The data telemeters shall be operated and checked before and after this test. They shall also be operated for 10 repeated duty cycles after the payload has achieved the specified spin rate.

##### 2. Acceleration:

(a) Thrust axis - The payload shall be subjected to a static acceleration of 24g at its center of gravity, which is equal to  $1\frac{1}{2}$  times the expected value. The variation along the thrust axis shall not exceed -15 percent at the forward end and 10 percent at the aft end of the payload. This force shall be held for 3 minutes. The payload telemeter shall be in an operational status and shall be checked before and after this test.

(b) Lateral (normal to thrust axis) - The payload shall be subjected to a static acceleration of 10.5g, which is equal to  $1\frac{1}{2}$  times the expected

value, in the direction most likely to produce damage. A second test shall be conducted with the force at 90° to this direction. This force shall be held for 3 minutes in each direction. The payload telemeter shall be in an operational status and shall be checked before and after this test.

3. Temperature: With the payload mounted in a suitable test chamber, it shall be subjected to a temperature environment at atmospheric pressure in accordance with the following schedule:

(a) Lower the test-chamber temperature to -22° F and hold for 6 hours. Operate the data telemeters for a normal duty cycle during the holding period.

(b) Raise the test-chamber temperature to 140° F (18° F above the maximum expected flight temperature) and hold for 6 hours. Operate the data telemeters for a normal duty cycle during holding period.

(c) Lower the test-chamber temperature to 10° F (18° F below the minimum expected flight temperature) and operate the data telemeters.

(d) Raise the test-chamber temperature to 140° F and operate the data telemeters for several duty cycles.

(e) Lower the test-chamber temperature to room temperature and operate the data telemeters.

4. Thermal vacuum: The prototype shall be subjected to a thermal-vacuum test as outlined below. The maximum and minimum temperatures of the components as tabulated below shall not be exceeded.

Component	Temperature, °F	
	Maximum	Minimum
Solar cells	250	-50
Pressurized cells	200	-25
Grid detectors	150	-50
Telemeters	140	-30

(a) Vacuum - The test chamber shall be evacuated to  $1 \times 10^{-4}$  mm Hg or less with the payload remaining at ambient temperature. The telemeters shall be operated while the chamber is being evacuated.

(b) Hot Soak - While maintaining the aforementioned vacuum, raise the chamber temperature until the temperatures of the telemeters reach 140° F. Hold the vacuum and temperature for at least 7 days. Operate the data telemeters for a normal duty cycle.

(c) Cold Soak - While still maintaining the vacuum, lower the test-chamber temperature until the temperature of the telemeters reaches 10° F.

Hold the vacuum and temperature for at least 3 days. Operate the data telemeters for a normal duty cycle.

(d) Cycle - While maintaining the vacuum, the temperature of the forward shell shall be varied to duplicate the temperature time profile of a typical orbit. The test shall be conducted for 14 typical orbits (approximately 24 hours). Operate the data telemeters for a normal duty cycle.

(e) Ambient Conditions - Bring the test chamber back to atmospheric pressure and the payload to ambient temperature and operate the data telemeters a few cycles.

5. Vibration: The payload shall be in an operational status during the vibration test and checked electrically and mechanically after completion. A detailed examination for evidence of cracks, wear, loose parts, and the like shall be made.

The payload shall be subjected to a low-level vibration of  $\pm 2g$  for 90 seconds by using a sine-wave logarithmic sweep from 15 to 2000 cps. In addition, it shall be subjected to vibrations in three orthogonal directions in accordance with the following schedule. The prototype test levels and durations are 50 percent more than those expected in flight.

(a) Sinusoidal

Frequency range, cps	Amplitude, g-rms along -	
	Thrust axis	Transverse axis
5 to 50	2	1
50 to 500	5	2
500 to 2000	12	3

The sweep rate shall be approximately 2 octaves per minute with a duration of approximately  $1\frac{1}{2}$  minutes per frequency range.

(b) Random

Direction	Frequency range, cps	Amplitude, g-rms
Thrust axis	20 to 2000	13
Transverse axis	20 to 2000	3

The duration shall be 5 minutes in each direction.

(c) Resonance

Direction	Frequency range, cps	Amplitude, g-rms
Thrust axis	550 to 650	45
Transverse axis	550 to 650	10

The duration shall be 12 seconds in each direction.

6. Shock: The payload shall be subjected to a 45g shock load parallel to and in the direction of the thrust of the rocket motor. The duration of the shock pulse shall be not less than 5 milliseconds. This shock load is  $1\frac{1}{2}$  times greater than that expected in flight. The telemeters shall be operational during this test. The payload shall be checked electrically and mechanically after this test.

B. Flight Acceptance.-

1. Thermal vacuum: The payload shall be subjected to a thermal-vacuum test as outlined below. The temperature of the components as tabulated below shall not be exceeded.

Component	Temperature, °F	
	Maximum	Minimum
Solar cells	250	-50
Pressurized cells	*160	-25
Grid detectors	*125	-50
Telemeters	140	-30

\*The temperature limits were reduced due to experience gained in the flight qualification tests.

(a) Vacuum - The test chamber shall be evacuated to  $1 \times 10^{-4}$  mm Hg or less with the payload remaining at ambient temperature. The telemeters shall be operated while the chamber is being evacuated.

(b) Hot Soak - While maintaining the aforementioned vacuum, raise the chamber temperature until the temperatures of the telemeters reach 110° F. (The temperatures presented represent the expected minimum and maximum flight temperatures.) Hold the vacuum and temperature for at least 3 days. Operate the data telemeters for a normal duty cycle.

(c) Cold Soak - While still maintaining the vacuum, lower the test-chamber temperature until the temperature of the telemeters reaches 25° F. Hold the vacuum and temperature for at least 2 days. Operate the data telemeters for a normal duty cycle.

(d) Cycle - While maintaining the vacuum, the temperature of the forward shell shall be varied to duplicate the temperature time profile of a typical orbit. The test shall be conducted for 14 typical orbits (approximately 24 hours). Operate the data telemeters for a normal duty cycle.

(e) Ambient Conditions - Bring the test chamber back to atmospheric pressure and the payload to ambient temperature and operate the data telemeters a few cycles.

2. Vibration: The payload shall be in an operational status during this test and checked electrically and mechanically after completion. A detailed examination for evidence of cracks, wear, loose parts, and the like shall be made.

The payload shall be subjected to a low-level vibration of 1g for 60 seconds by using a sine-wave logarithmic sweep from 15 to 2000 cps. In addition, it shall be subjected to vibrations in three orthogonal directions in accordance with the schedule listed below. The payload test levels and durations are the same as those expected in flight.

(a) Sinusoidal

Frequency range, cps	Amplitude, g-rms along -	
	Thrust axis	Transverse axis
5 to 50	1	5
50 to 500	3	1
500 to 2000	10	2

The sweep rate shall be approximately 4 octaves per minute, with a duration of approximately 1 minute per frequency range. The amplitudes on the vibration table shall be limited to 0.5 inch peak to peak.

(b) Random

Direction	Frequency range, cps	Amplitude, g-rms
Thrust axis	20 to 2000	8
Transverse axis	20 to 2000	2

Duration shall be 4 minutes in each direction.

(c) Resonance

Direction	Frequency range, cps	Amplitude, g-rms
Thrust axis	550 to 650	40
Transverse axis	550 to 650	6

Duration shall be 10 seconds in each direction.

3. Shock: The payload shall be subjected to a 30g shock load parallel to and in the direction of the thrust of the rocket motor. The duration of the shock pulse shall be not less than 5 milliseconds. This shock load is equal to that expected in flight. The telemeters shall be operational during this test. The payload shall be checked electrically and mechanically after this test.

SECTION III - FUNCTIONAL AND DESIGN TESTS

Subsystem.- Early in the project, a program of evaluations of the electrical system of each subassembly was devised for the telemeter prototype. Before any telemeter module was incorporated into the telemeter stack, it had to be operated in a temperature chamber from 32° F to 140° F without deterioration of its operation. At the same time, electrical performance was evaluated under varying voltage conditions. Each telemeter was enclosed by a pressurized container which was mounted on the telemeter bulkhead. Early results led to several changes in the d-c voltage converter. Complete calibrations to determine temperature effects were made on the subcarrier oscillator module before it was incorporated into the telemeter. After each module had been qualified under these tests, it became a part of the telemeter.

Systems.- The bulkhead, with its two telemeters, underwent extensive electrical tests before flight qualification.

Command rf signals were transmitted to the telemeters. Proper operation of the command receivers would allow the satellite transmitters to operate for approximately 1 minute and then return to the off condition. During these tests, trouble developed in the time-delay turn-on circuit which required that the circuit be redesigned. Also, because modulation trouble was experienced during the electrical tests, the push-pull modulator output circuitry was redesigned.

An operational test was performed on the telemeter assembly in which an rf signal was sent to the telemeter once every 3 minutes. After 500 of these cycle tests, a calibration check was made; the past calibration agreed with the present calibration and indicated that no change occurred during the test.

The telemeters were then submitted to a 10-hour hot and cold test with stabilized temperatures ranging from 20° F to 120° F. Power was supplied to the telemetry system from an external source, and operation of the system was

commanded through an rf link. Over this 10-hour period, the telemeter was operated on a 1-minute "on" and 1-minute "off" basis. This hot and cold test was accomplished without a telemeter failure.

Next, the telemeter was connected to its sensors. Each sensor input was individually shorted while each telemeter channel was interrogated. Interference experienced between the impact-detector and the pressurized-cell-detector inputs was located in the pressurized-cell mounting-structure wiring tunnels. The trouble was apparently caused by proximity of lead wires in the wiring tunnels. It was discovered that the pressurized-cell-detector leads had an induced voltage pulse ranging from 3 to 5 volts (this pulse came from the encoder) and had caused crosstalk. Additional shielding of the impact-detector leads and a complete wire rerouting reduced the interference voltage to an acceptable level.

At this stage of the test program, the battery supply modules were installed in the telemeter system. Electrical checks were performed to ensure the proper operation between the power supply and the solar-cell charging system. All prototype tests were conducted by using this internal power system.

Mock-up Vibration Test.- Vibration tests were also made early in the program on a complete satellite payload with mock-up instrumentation. The telemeter canisters were filled with a potting compound to simulate the actual weight. The telemeter bulkhead was completely assembled, including the external-internal power relay, battery supply, and the support spider. The mock-up was first vibrated in two sections and finally as a complete unit. This test was conducted to qualify the structural worthiness of the payload hardware. A vigorous vibration test program was initiated using an electronic shaker located at Langley. Readout equipment necessary for this test consisted of the following:

1. 12 accelerometers
2. Data-write equipment for oscillograph
3. Tape recorder and assorted playback equipment

Mock-up Payload Adaptor, Telemeter Bulkhead, and Forward Shell.- The first tests were made on the forward shell, telemeter bulkhead, and payload adaptor that mounts on the Scout fourth-stage X-248 rocket motor. Vibrometers were located at the center top of the forward shell to check the vibration expected at the base of the solar cells. Vibration pickups were located at the base and top of the telemeter canisters, and at the input to the payload adaptor. A pickup was located on the support bracket located under the bulkhead.

Input g-levels were applied to the base longitudinally from 2 to 20g in three separate steps and over the frequency range from 20 to 2000 cps; each test lasted about 3 minutes. From past experience gained from X-248 rocket motor firing, a random vibration test over the range from 550 to 650 cps at 11 g-rms for 5 minutes was conducted.

Amplification factors approached 100 on the telemeter bulkhead, while the X-, Y-, and Z-axis vibrometers mounted on top of the telemeter canisters

exceeded this amplification. Three of the four antennas broke off during the test at a frequency of 125 cps on the 12g run.

The bulkhead support was redesigned, stiffened, and also made lighter. X-ray pictures were taken of the old antennas and revealed their structural flaws; new antennas were designed and made. The vibration tests were repeated and the structure withstood each test without damage.

Pressurized-Cell Support Structure.- The pressurized-cell support structure with pressurized cells installed was mounted on the vibration table and vibrated in the same manner as the forward shell. Only 40 pressurized cells were mounted on the structure for this test. Small vibrometers were mounted on several of the cells along with monitors on the structure itself and on the shaker table. During preliminary tests, the lower support foot showed fatigue on 20 percent of the pressurized cells.

The tests were halted at this point and a method of supporting the pressure cell by the addition of a rubber shock mount was tried. In this new configuration 10 percent of the cells were lost. This method of support was abandoned. Modifications were made to the support foot through the addition of a gusset, and the tests were rerun and completed without structural damage.

Complete Mock-Up Payload Assembly.- After the modifications had been made to the payload hardware and detectors, the vibration tests were rerun successfully.

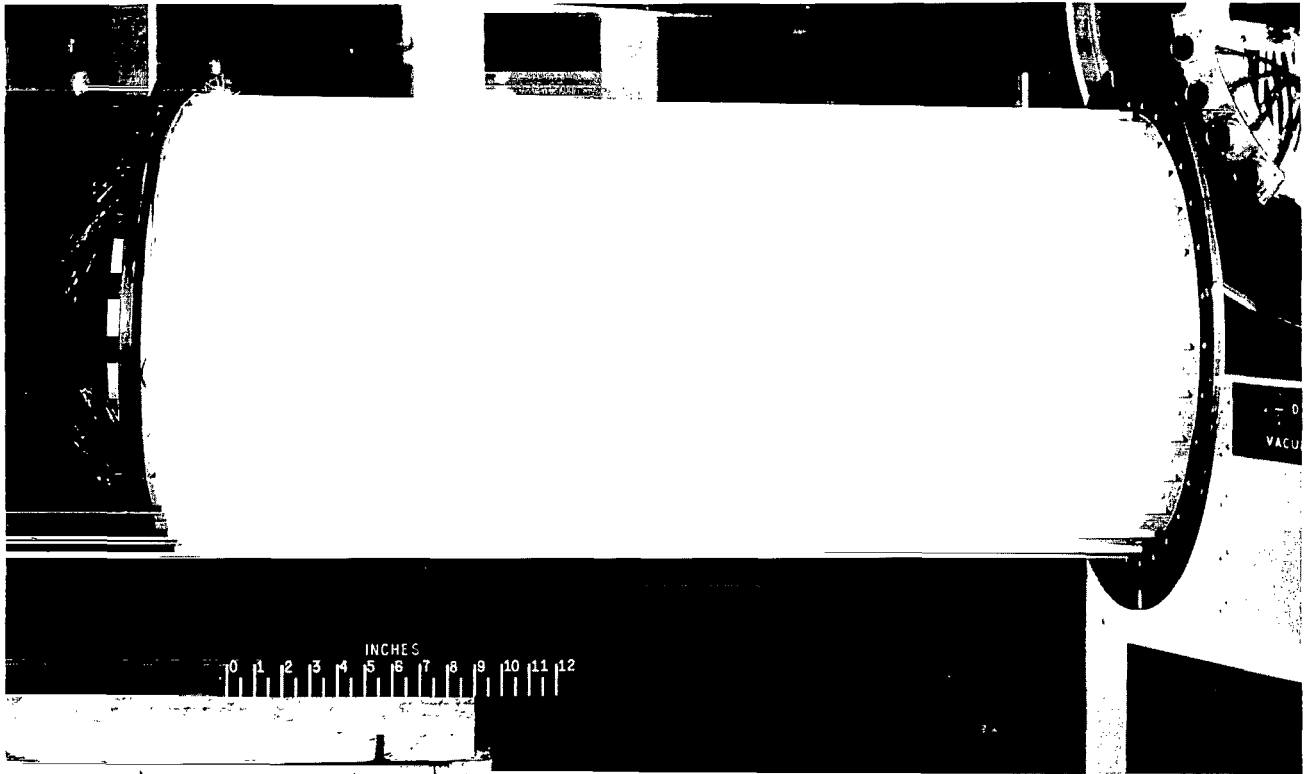
Fourth-Stage Rocket-Motor Heat-Transfer Test.- As pointed out in chapter VI, it was necessary to keep the temperature of the satellite detectors below 180° F. The fourth-stage rocket-motor case is a major source of heat immediately after burnout. In order to determine the heat transfer from the motor case to the payload, a series of tests were conducted in a vacuum chamber at low pressures which simulated the altitude at which the motor burnout occurs.

Temperatures measured during actual rocket-motor test firings were fed into a programmer which used a point on the outside surface of the motor case midway between the rocket-motor mounting flanges as the control point. The case was heated by six 2,500-watt quartz lamps inserted through the nozzle end of the rocket



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Figure VII-1.- Rocket-motor heat-transfer test; vacuum chamber and test hardware.



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Figure VII-2.- Rocket-motor heat-transfer test; painted pressurized-cell mounting structure and monitor thermocouple.

motor. Figure VII-1 shows the assembly in the vacuum chamber. The tests were conducted until the outside of the pressurized-cell mounting structure (fig. VII-2) reached peak temperature and started to decline. Generally, peak temperature was not reached in less than 30 minutes.

It was found that by covering the outside of the X-248 rocket motor, inside and outside of the payload adapter and inside of the pressurized-cell mounting structure with aluminum foil and painting the outside of the pressurized-cell mounting structure with acrylic white lacquer that the temperature of the outside of the pressurized-cell mounting structure did not exceed 145° F 45 minutes after simulated motor burnout. Figure VII-3 shows the spent X-248 motor case covered with aluminum foil and the array of six 2,500-watt quartz lamps, mounting hardware, and test equipment. Figure VII-4 shows the payload adapter and pressurized-cell mounting structure with foil applied. Figure VII-2 shows the pressurized-cell mounting structure painted.

#### SECTION IV - FLIGHT QUALIFICATION TESTS

After the component and mock-up test outlined in section III, the assembled prototype payload was subjected to a series of mechanical, electrical, temperature, and thermal-vacuum tests. These tests were conducted to ascertain the soundness of design both mechanically and electrically. The test specifications are outlined in section II.

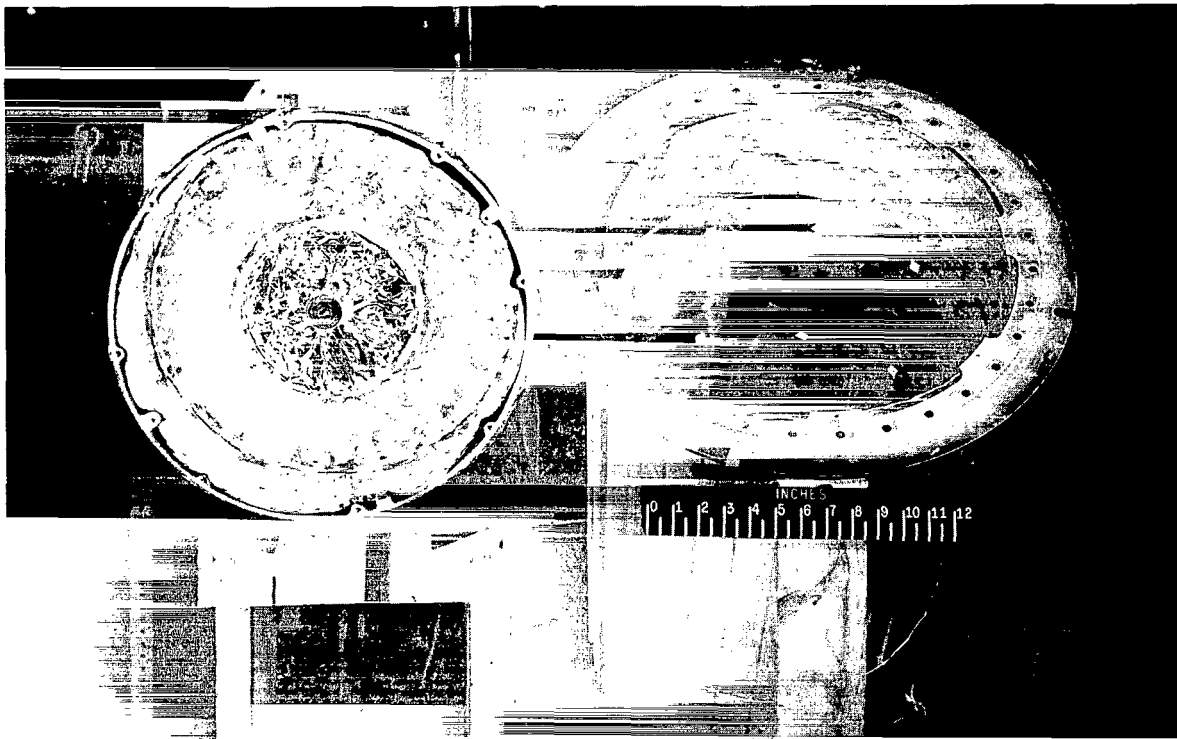
Spin Test.- The prototype was first subjected to a spin test to approximate the spin imparted to the satellite (180 rpm) by the spinning fourth-stage motor of the Scout launch vehicle. The prototype payload was mounted in a lathe (fig. VII-5) and minor adjustments made to minimize eccentricities. The telemeters were operated and sensors examined; all were found to be in satisfactory condition. A slow-speed run was made to check for unbalance; balance weights were not required. The payload was then spun up to 154 rpm, the maximum rpm of the lathe. This spin rate was held for 15 minutes while the telemeters were operated satisfactorily for 10 duty cycles. After despinning, the telemeters were operated again; the sensors inspected; and antennas checked. All were found in satisfactory condition.

Acceleration Test.- The prototype was completely enclosed in a windshield and only mounted in the plane of the vehicle spin bearing. The antennas had to be in the folded condition thereby rendering the telemeters inoperative.

The sensors were examined and telemeters checked and found to be in satisfactory condition before the payload was installed in the windshield. The windshield was then mounted on a centrifuge and the payload subjected to 24g acceleration in the longitudinal thrust direction (fig. VII-6) and  $10\frac{1}{2}g$  in the normal and transverse directions (fig. VII-7). These g-levels were held for 3 minutes in each condition. When the test was completed, the telemeters were checked and found in satisfactory condition. One pressurized-cell detector and the front pressurized-cell mounting-structure ring were damaged during this test; all other sensors were in satisfactory condition. It is believed that this damage was incurred during handling, when the wind shield bent under one of the lifting rig supports.



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Figure VII-3.- Rocket-motor heat-transfer test;  
test hardware and instrumentation.



L-60-2960

Figure VII-4.- Rocket-motor heat-transfer test; pressurized-cell mounting structure with aluminum foil on inside surface on the right and dome (forward end) of the X-248 rocket motor covered with aluminum foil.

**Thermal-Vacuum Test.**- A complete telemeter receiving station was set up for this test (fig. VII-8). This station consisted of a tape recorder and quick-look readout equipment. Thirty-six additional thermocouples were located on the prototype and recorded through auxiliary equipment.

Performance checks prior to the thermal-vacuum test indicated a loss in command-receiver sensitivity, on internal power only, for telemeter B. The sensitivity loss was not present when the telemeter was operated from an external power source. Telemeter B contained a radio-tracking beacon that was to operate continuously for the first 3 weeks of the orbital flight. This beacon signal was blocking the command transmission to telemeter B. (A method of rf shielding within the canister reduced the problem to an extent such as to allow the test to be continued.) Normal receiver sensitivity for both telemeters was between -96 to -98 dbm, and power output of the transmitters was 92 mw.

Before the thermal-vacuum test was begun, a short 12-hour temperature test was conducted. This test covered excursions of telemeter temperatures from -22° F to 140° F. The purpose of the test was to study heat balance and the only instrumentation mandatory were recorders and the 36 thermocouples mounted on the structure of the prototype. Prior knowledge of the telemeter indicated it would not operate at temperatures below 0° F. Although the telemeter could

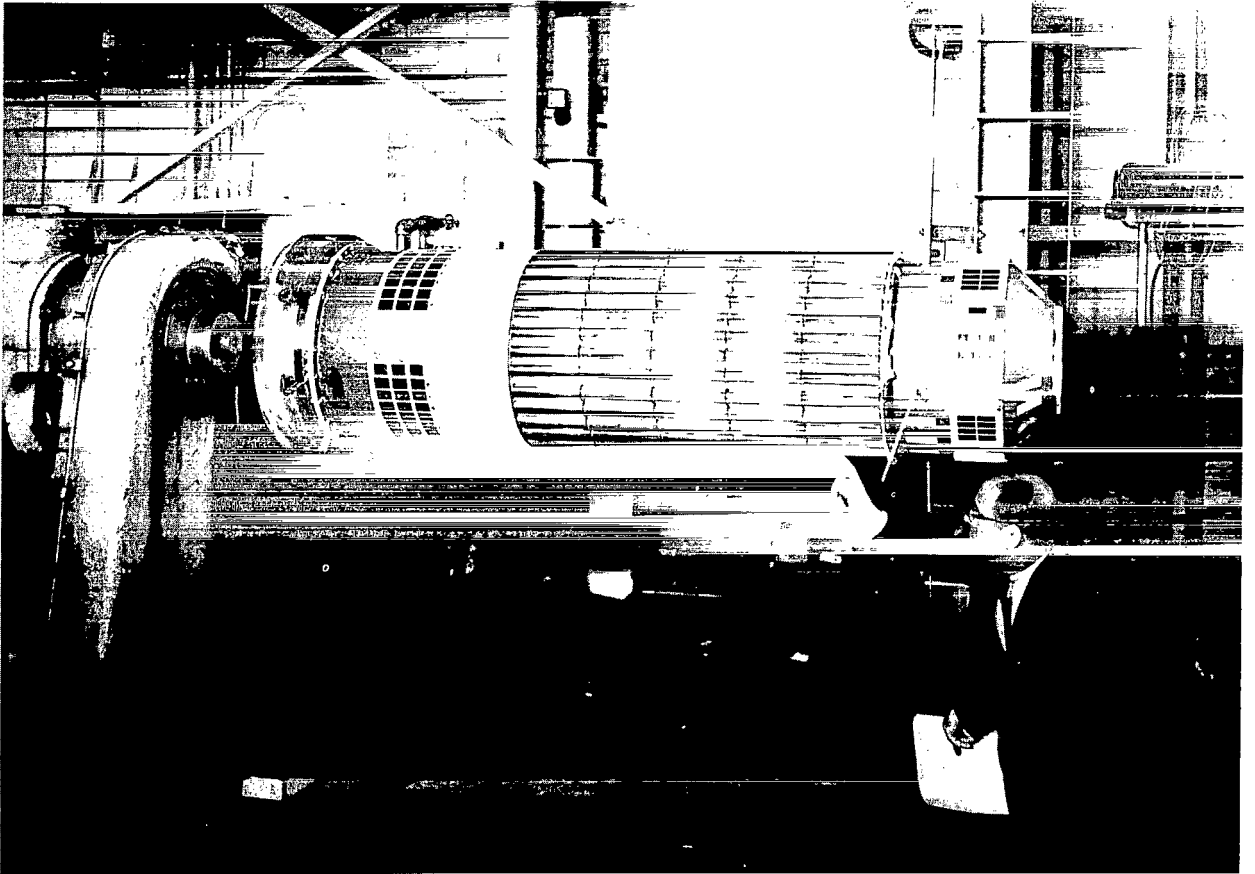


Figure VII-5.- Spin test; prototype payload mounted on a lathe.

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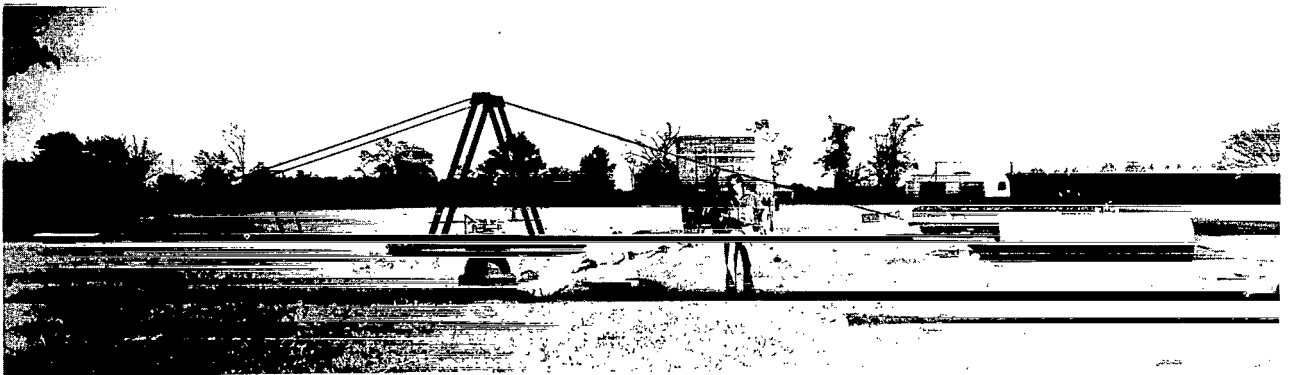


Figure VII-6.- Acceleration test (thrust axis); prototype payload (in windshield) mounted on centrifuge.

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L-60-7267

Figure VII-7.- Acceleration test (transverse axis); prototype payload (in windshield) mounted on centrifuge.

not be expected to operate under these conditions, this test afforded an opportunity to check the lower limit of the telemeter operation.

A complete set of telemeter zero references was made for each of the 96 channels at room temperature with the prototype mounted in the chamber. Test procedures required that the telemeter be commanded on every 30 minutes and complete tape records made. The temperature was dropped to  $-22^{\circ}$  F for the 6-hour cold test. At the end of the first  $1\frac{1}{2}$  hours of the test, the modulating tone necessary to operate the telemeter B receiver had drifted 50 cycles. After 3 hours of testing, the internal telemeter temperature had reached  $5^{\circ}$  F and telemeter B could not be commanded on. Four hours after the test began, the base of the telemeter canisters was reading  $-30^{\circ}$  F and neither telemeter could be commanded. The cold test was continued for 2 more hours. It was still impossible to operate either telemeter.

Heat was applied and the prototype underwent a 6-hour hot soak. The prototype was not removed from the chamber and was still in an inoperative condition. Four hours after the hot soak began, when the base of the telemeter had reached  $55^{\circ}$  F, it was possible to operate telemeter B. Five hours after the test began, both telemeters were operating normally, but the frequency shift that occurred during the cold soak in telemeter B had not rectified itself.

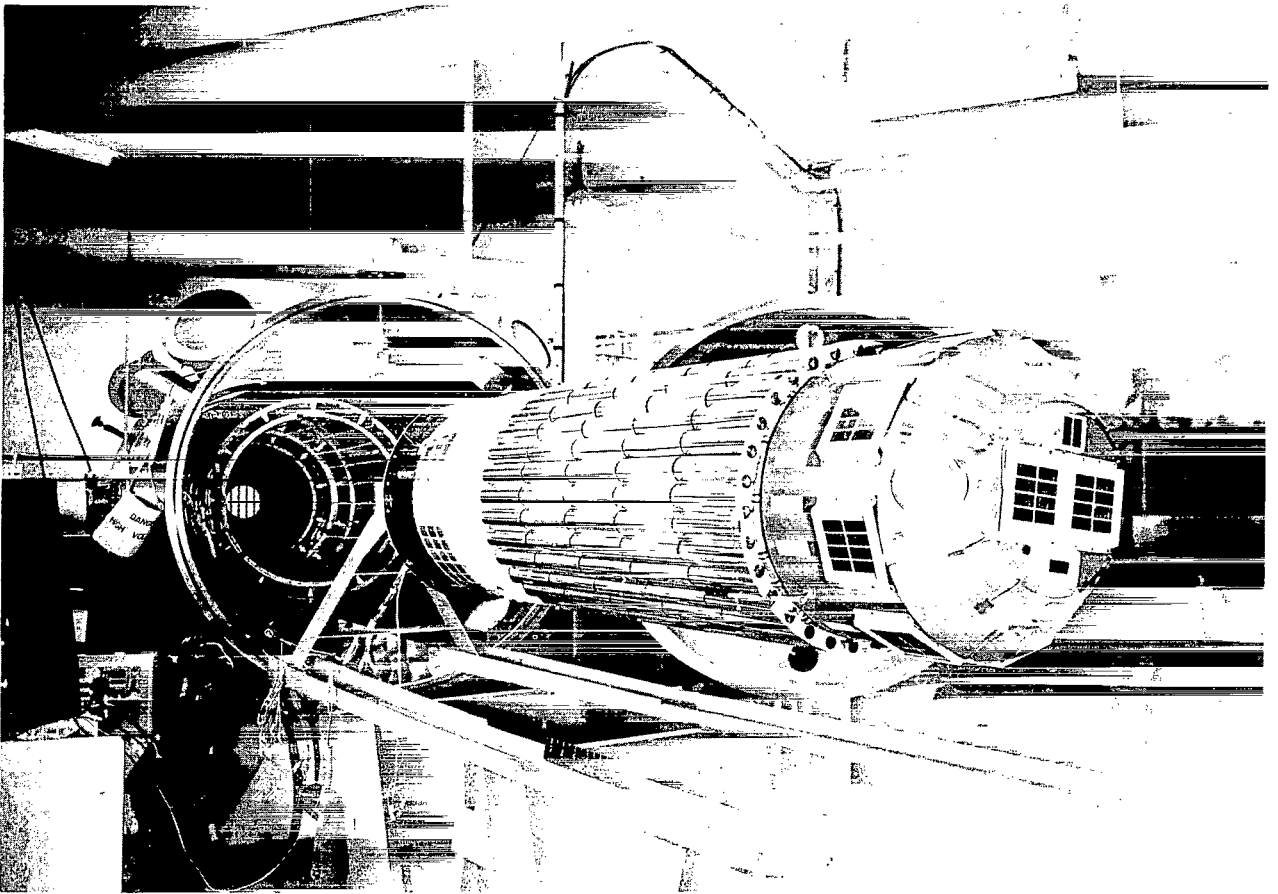


Figure VII-8.- Temperature and thermal-vacuum test; prototype payload readied for test. L-60-8225

Immediately following the temperature tests, the prototype was subjected to thermal-vacuum tests which simulated conditions in the space environment. The following tests were performed while under a vacuum of  $10^{-5}$  mm of mercury, as outlined in section II.

- (1) The temperature was raised to  $140^{\circ}$  F and held at this point for 7 days.
- (2) Still holding the vacuum, the temperature was lowered to  $10^{\circ}$  F and held for 3 days.
- (3) The final test consisted of a cycle every 98 minutes, between  $10^{\circ}$  F and  $140^{\circ}$  F for 14 cycles.

The telemeters were operating continuously during these tests.

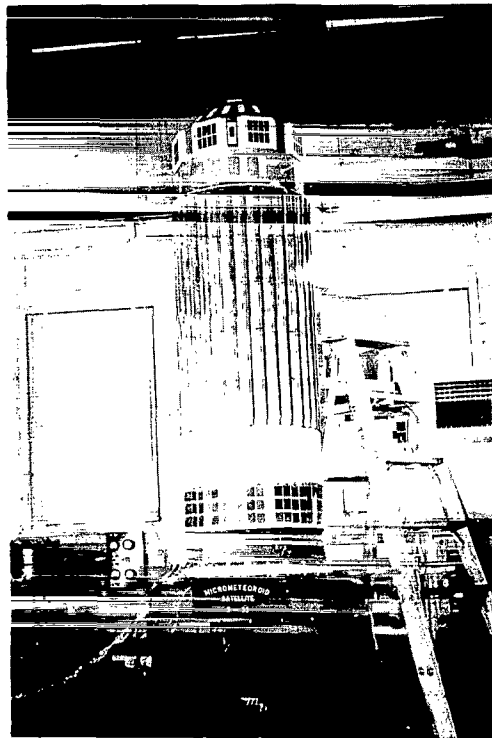
A complete set of zero calibrations was made at room temperature, and preparations were made to make a 7-day hot test at  $140^{\circ}$  F. After determining

that the telemeters were working properly, heat was applied. Interrogations were made every 30 minutes, by using a signal generator and an audio generator for the modulating tone. Complete magnetic tape records were made in addition to the quick-look setup which used a counter and paper tape. Internal package temperature, channel D-4, was telemetered and monitored through the rf link along with internal battery voltage, channel F-12. The third day of the test, the 1-minute transmitting time of telemeter B after interrogation had changed from 58 seconds to 15 seconds. Before the final day of the test, telemeter B began operating properly upon interrogation and worked for the full 58 seconds. The complete 7-day hot and 3-day cold tests were completed.

Upon completion of these tests, the telemeter deck was removed from the prototype, and an extensive program of modification was begun on the telemeters. The oscillator circuit in the receiver modules was reworked and the bandwidth of the interrogation tone increased by 50 cycles. The internal batteries were examined and their charge circuitry modified.

Vibration Test.- The vibration-test program conducted on the prototype ranged from 1g to 4.5g in amplitude with a frequency excursion from 20 to 2000 cps as outlined in section II (fig. VII-9). Operation of the telemeter was required throughout this test. A complete telemeter station was used to monitor the prototype during vibration.

Operation of the telemeter rf transmitter into the antenna system interfered with the vibration-table level-indication system to such an extent as to eliminate the rf type of test. Therefore, the transmitter output was directly coupled through coaxial cable to the receiving station. Satisfactory operation of the vibration table was then accomplished. Tape recordings were made of the telemeter operation during the vibration test. An analysis was made of the tape and showed that operation was normal during vibration. Vibration records were obtained from the numerous vibrometers located on the structure. These vibration records are shown in figures VII-10 and VII-11. The prototype was vibrated without mechanical or electrical failure to approximately  $1\frac{1}{2}$  times the expected flight environmental conditions. Experience gained from the vibration test of the mock-up payload contributed to the success of this test.



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Figure VII-9.- Vibration test; prototype payload mounted for test.

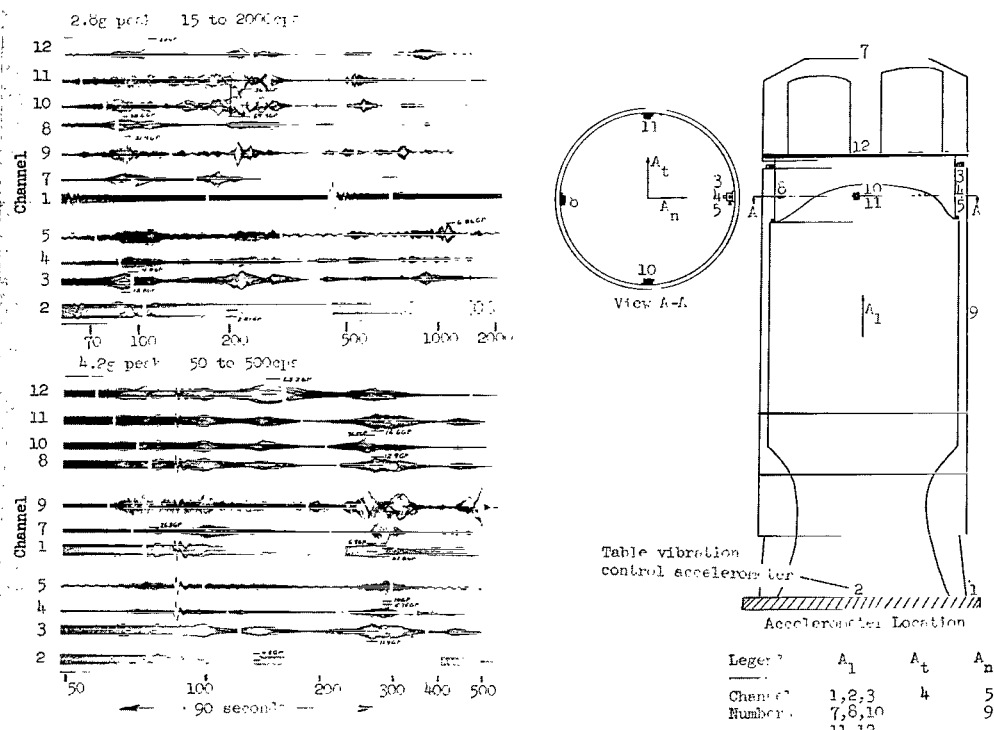


Figure VII-10.- Vibration test; prototype payload accelerometer locations and test records.

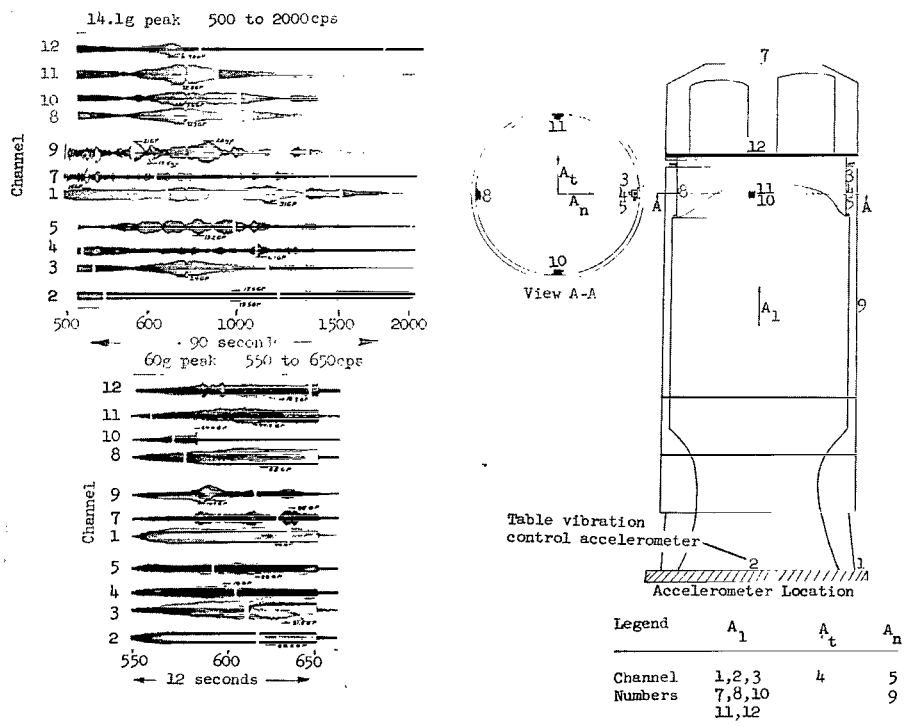
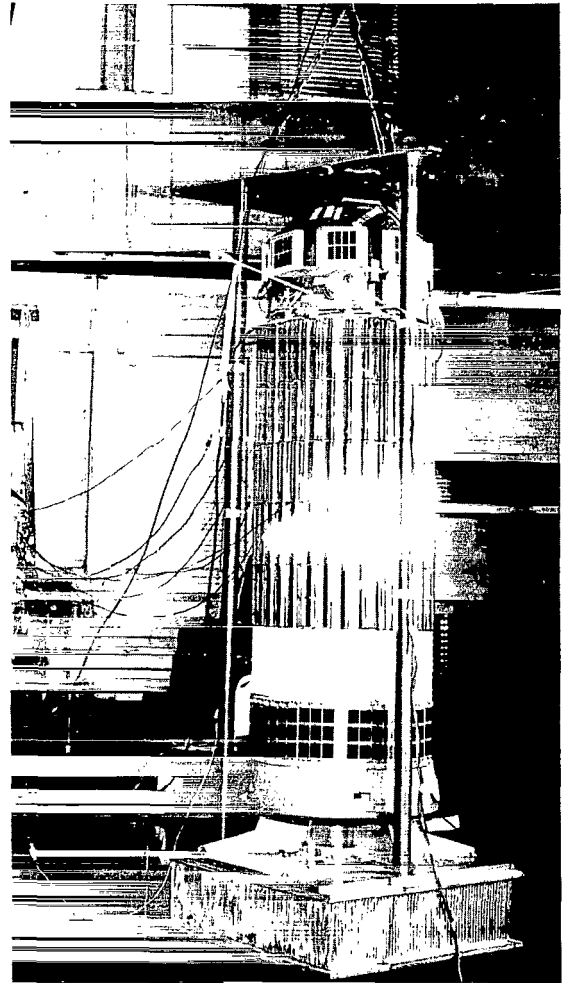


Figure VII-11.- Vibration test; prototype payload accelerometer locations and test records.

Shock Test.- The prototype was assembled on a test fixture (fig. VII-12) which had an aluminum honeycomb pad 31 inches square by 8 inches thick attached to its base. The prototype and test fixture assembly were raised 10 inches off the floor and dropped. Figure VII-13 is a record of this test showing an excursion of 45g and a duration of 10 milliseconds. The telemeters were operating on their internal power during this shock test. The signals transmitted by the telemeters were picked up and recorded on magnetic tape at a receiving station near where the test was conducted.

Telemeter zero calibrations were made before and after the shock test and indicated that the telemeter and detectors were not affected by this test. Because of the short duration of the shock pulse, it was impossible to detect any telemeter changes during that period. A visual inspection showed no damage to the hardware or detectors.



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Figure VII-12.- Shock test; prototype payload mounted in shock fixture.

## SECTION V - FLIGHT ACCEPTANCE TESTS

The flight payload was subjected to thermal-vacuum, vibration, and shock tests simulating the environment expected during launch and orbital flight. The tests were conducted in accordance with the specifications outlined in section II.

Thermal-Vacuum Test.- This test consisted of the 3-day hot soak at 110° F, 2-day soak at 25° F, and 14 simulated orbital cycles with temperature variations from 110° F to 25° F. All conditions were carried out at a pressure of 10<sup>-4</sup> mm Hg.

A complete telemeter station was used to readout the data. The quick-look counter and printer used in the prototype test were again employed. The rf coupling to the telemeters was direct through the coaxial cable, and additional battery monitoring equipment was used in this test.

The flight model was subjected to 180 hours of continuous testing as described previously. The malfunctions associated with prototype thermal-vacuum

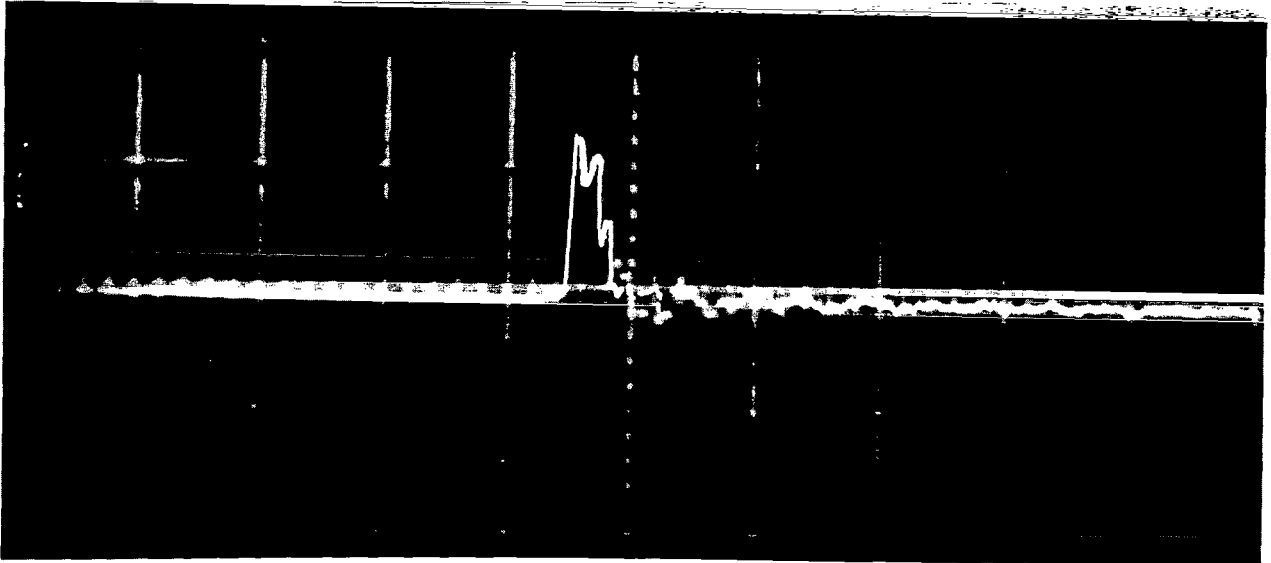


Figure VII-13.- Shock-test record; prototype payload.

tests were not present during the flight-acceptance tests. Figures VII-14 to VII-16 show telemetered data recorded through the rf link and compared with data from thermocouples located in the same area and recorded by a strip-chart recorder. Figures VII-17 to VII-22 indicate internal battery voltage, package temperature, full-scale and zero calibrations as telemetered through the rf link. Figure VII-23 shows the variation in the vacuum throughout the tests as recorded by gages located in the chamber (ion gage 1), and at the inlet to the diffusion pump of the vacuum chamber (ion gage 2).

Mechanical and Electrical Tests.- The complete flight satellite was assembled and each sensor input tested and recorded through an rf link. Operation of the system was performed for each module as previously done in the prototype telemeter construction. Complete calibration of each of the 96 telemeter channels was accomplished before and after every major test. The flight satellite was shock tested in the same manner as described for the prototype. The telemeter was operating on its flight batteries for this test and complete tape records were made of the operation during this test. Calibration of each channel was made before and after the test. Figure VII-24 is a record of the shock test indicating 35g peak for 10 milliseconds. No structural or telemeter damage was experienced during this test.

Test results of the flight-acceptance tests are shown in figures VII-25 and VII-26. Locations of monitoring vibrometers are also indicated in these figures. Some desirable vibrometer locations were deleted because of the possibility of contaminating the gold-plated surface of the telemeters and the flight bulkhead. Location of vibrometers on the forward-shell surface was impossible because of the heat-transfer coating. The vibration tests were performed with the telemeter operating on its internal battery supply. Monitoring of the telemeters was accomplished through the rf link, and recorded on magnetic tape. The complete test was carried out without mechanical or electrical failure.

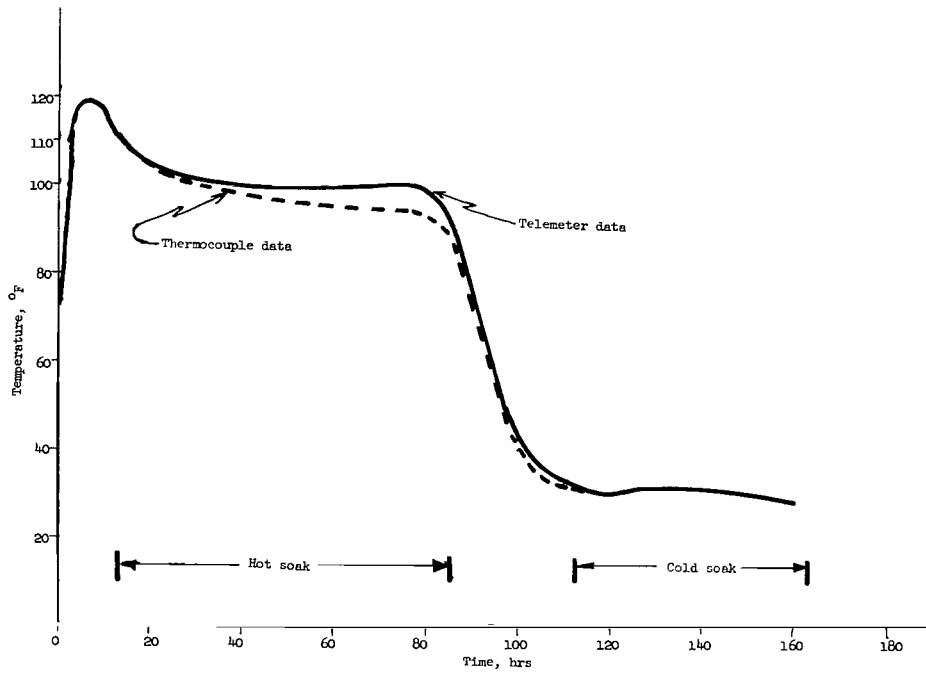


Figure VII-14.- Thermal-vacuum test record; flight-payload grid detector; channel S-11; telemeter "A".

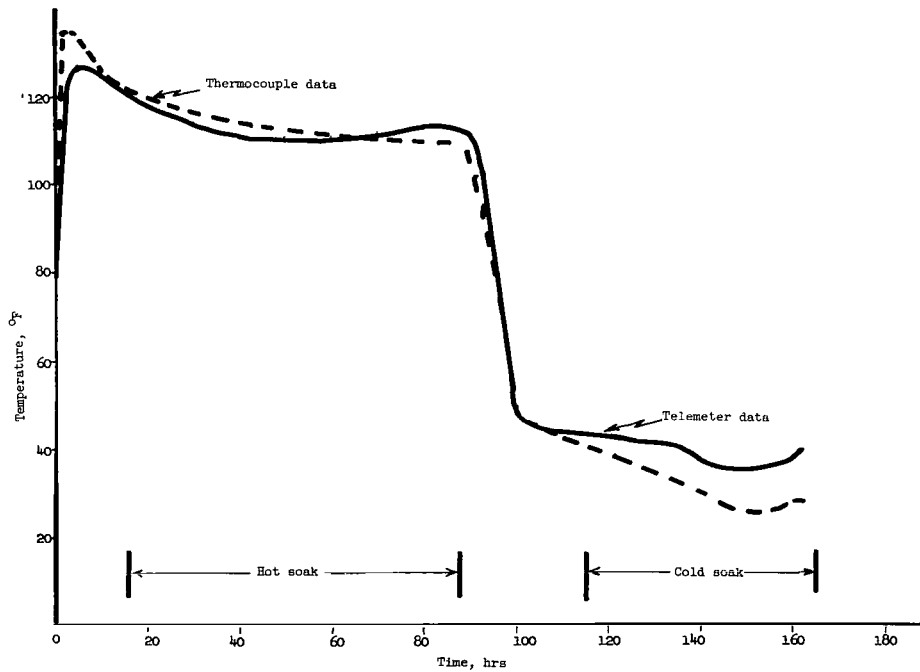


Figure VII-15.- Thermal-vacuum test record; flight-payload pressurized-cell detector; channel S-10; telemeter "A".

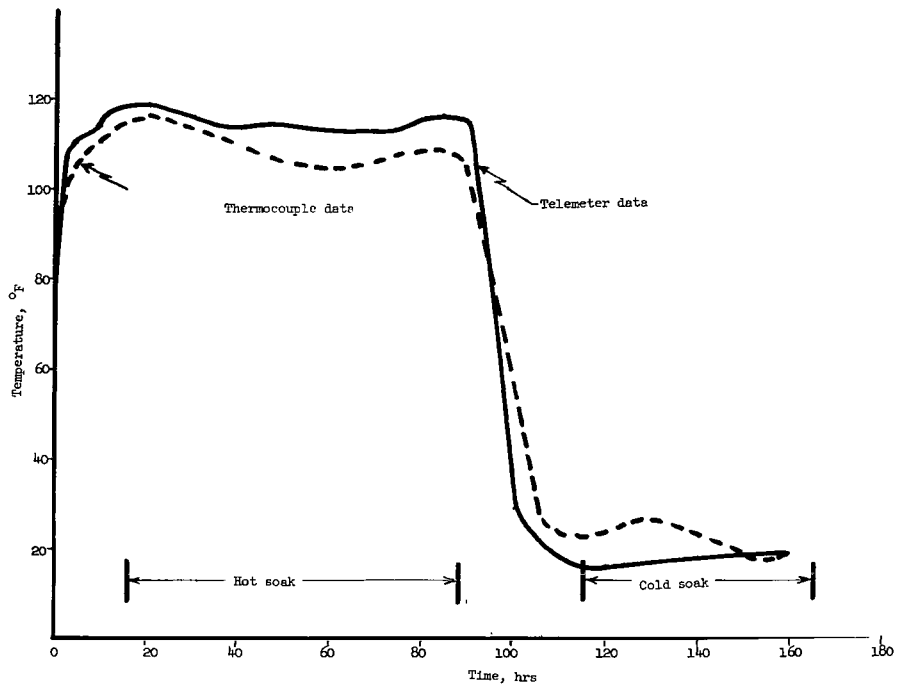


Figure VII-16.- Thermal-vacuum test record; flight-payload solar cell; channel D-11; telemeter "A".

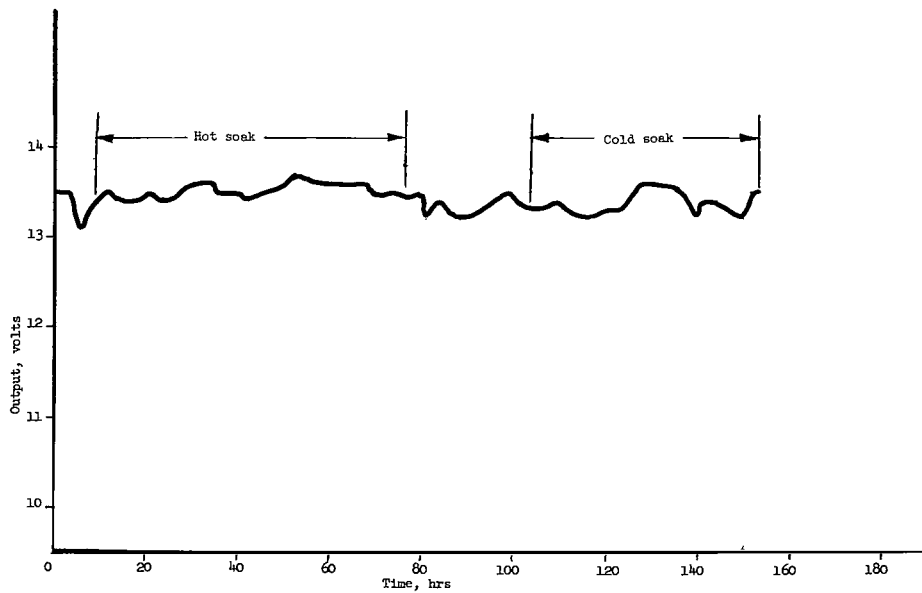


Figure VII-17.- Thermal-vacuum test record; flight payload; "B" battery voltage; channel F-12; telemeter "A".

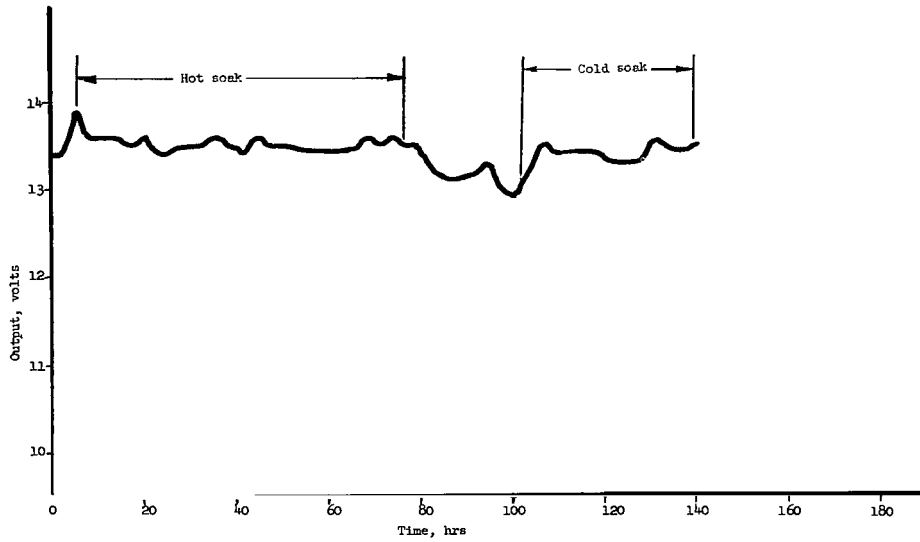


Figure VII-18.- Thermal-vacuum test record; flight payload; "B" battery voltage; channel F-12; telemeter B.

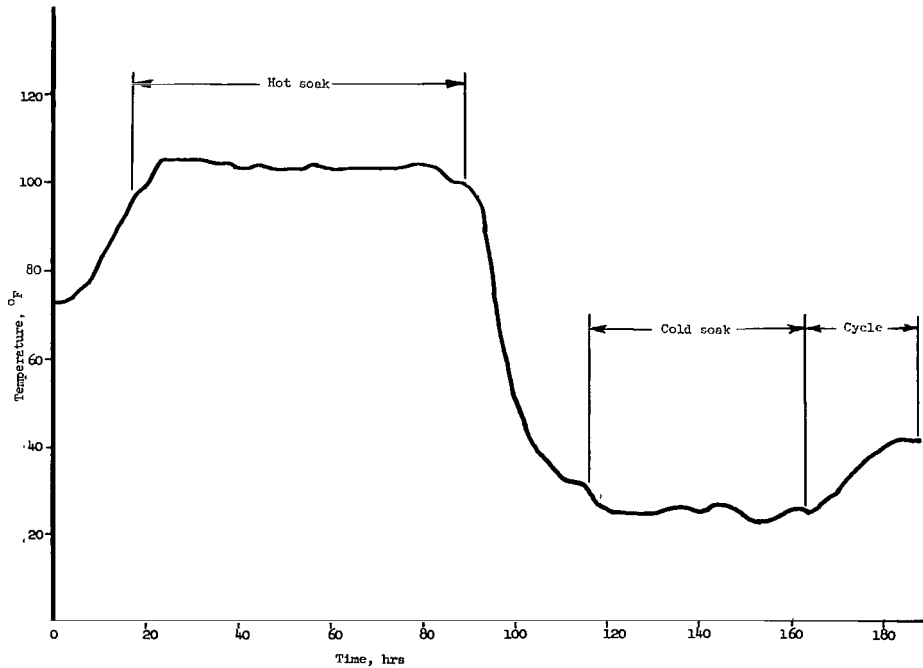


Figure VII-19.- Thermal-vacuum test record; flight-payload telemeter temperature; channel D-4; telemeter "A".

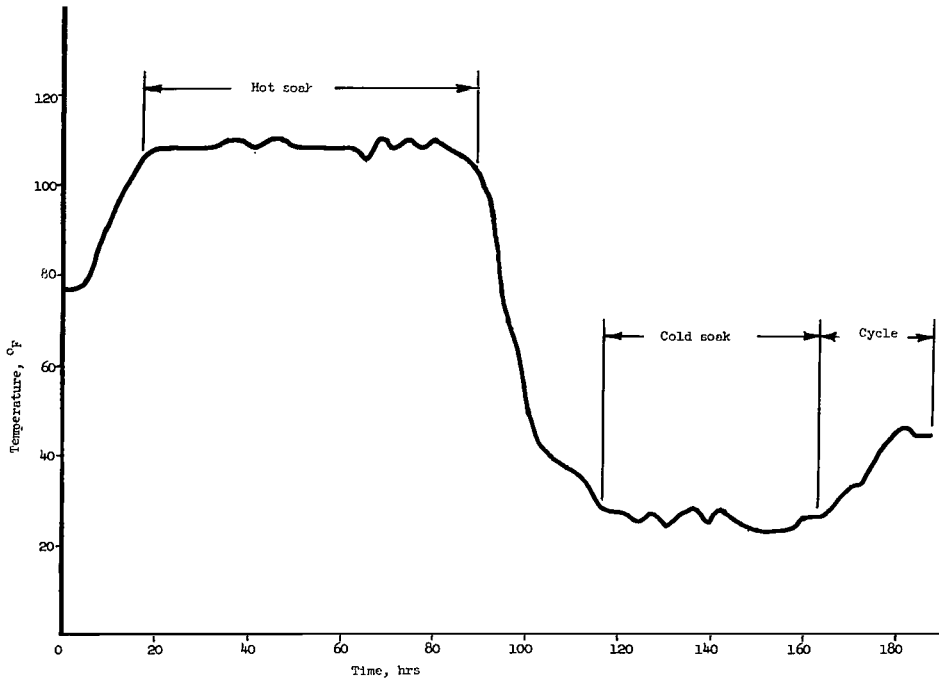


Figure VII-20.- Thermal-vacuum test record; flight-payload telemeter temperature; channel D-4; telemeter "B".

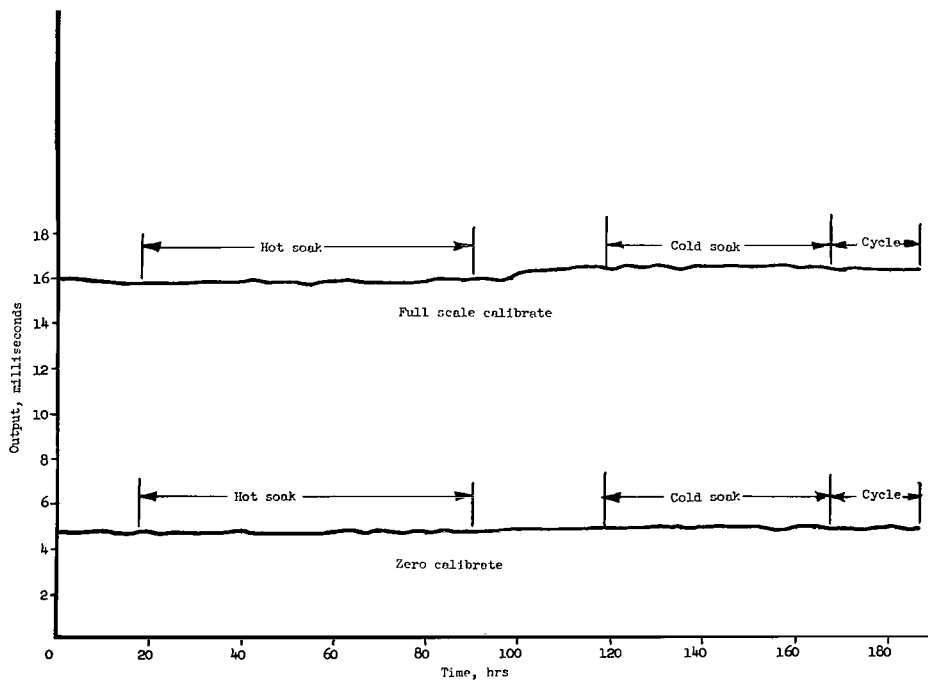


Figure VII-21.- Thermal-vacuum test record; flight payload; full scale and zero calibrate; channels 18 and 19; telemeter "A".

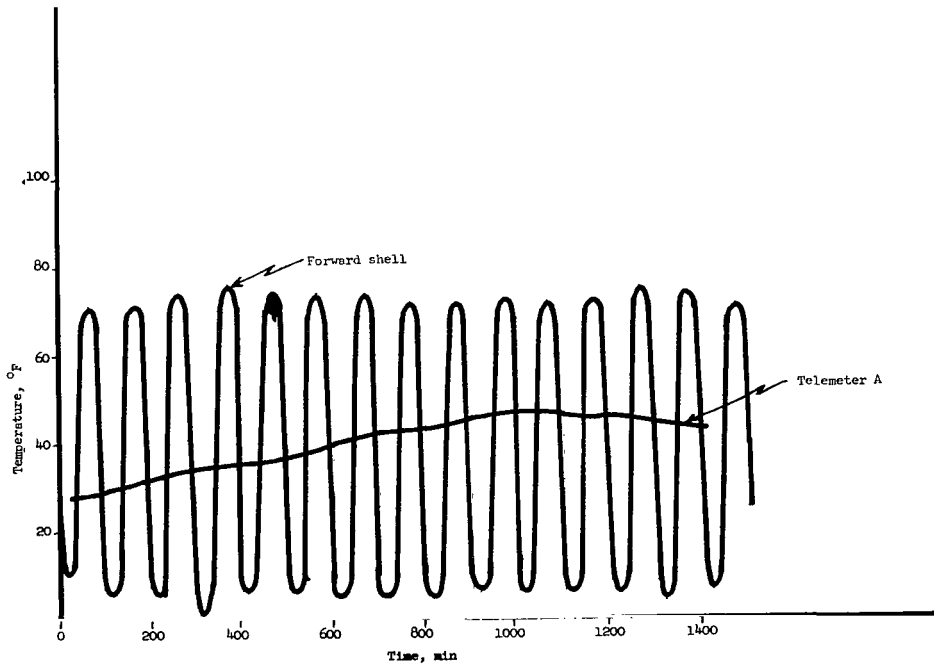


Figure VII-22.- Thermal-vacuum cyclic test record; flight-payload telemeter temperature; channel D-4; telemeter "A".

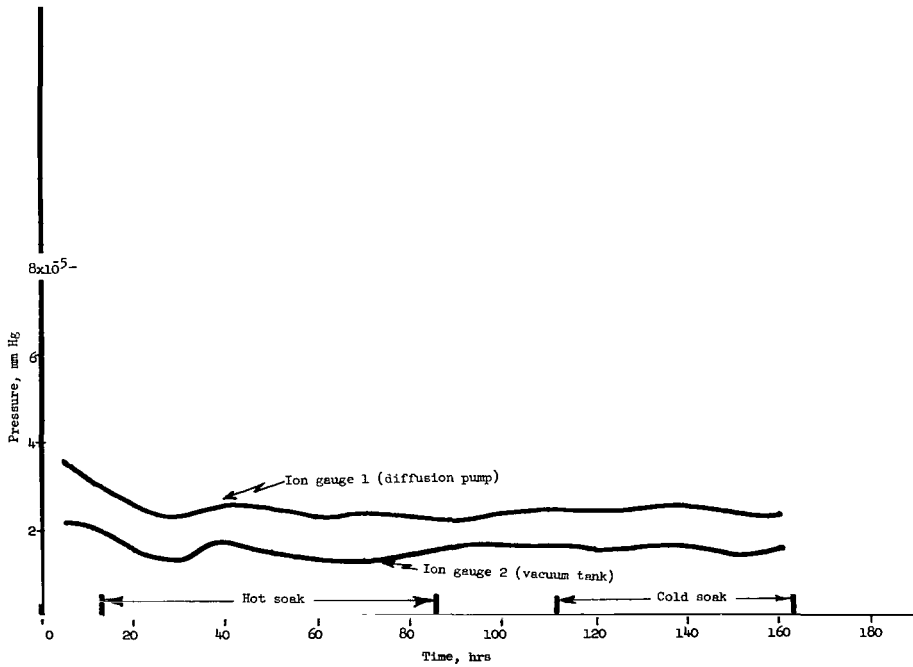


Figure VII-23.- Thermal-vacuum test record; flight payload; test-chamber pressure.

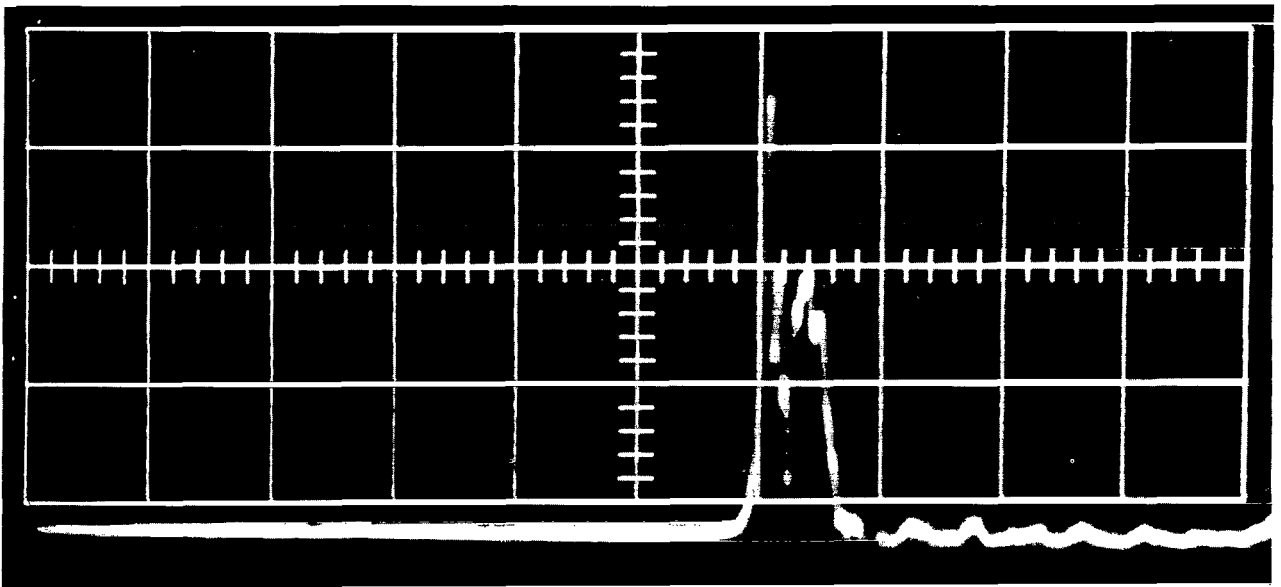


Figure VII-24.- Shock-test record; flight payload.

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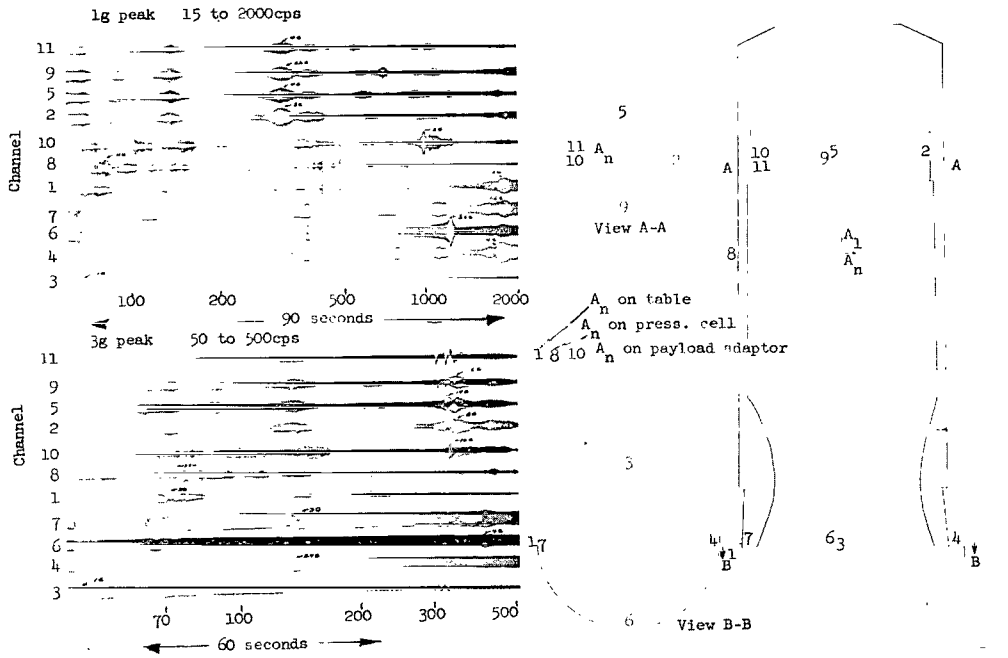


Figure VII-25.- Vibration test; flight payload; accelerometer locations and test records.

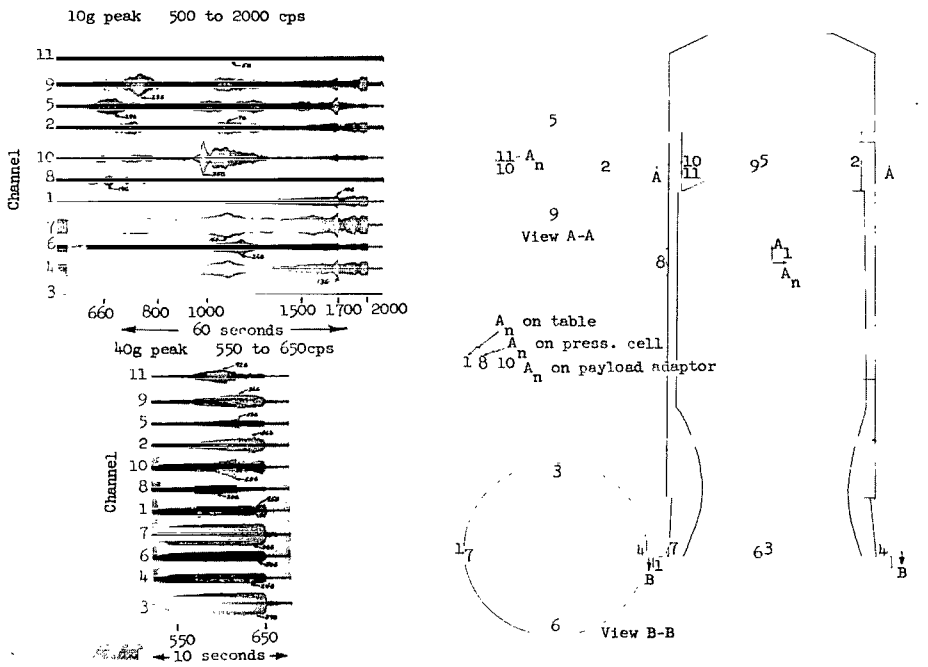


Figure VII-26.- Vibration test; flight payload; accelerometer locations and test records.