The Spacecraft Test and Evaluation Program

By T. B. DELCHAMPS, G. C. JONASSON and R. A. SWIFT

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Considerations guiding the planning and execution of environmental tests in the development, design qualification and flight acceptance phases of the Telstar satellite program are discussed. Specific test procedures are covered and highlights of test results involving mechanical, thermal and magnetic properties of the spacecraft are reviewed.

I. INTRODUCTION

Essential to the success of any development program are: (a) sound basic design; (b) discriminate selection of materials and components; and (c) careful fabrication. The function of test evaluation, as a fourth phase in the development plan, is to determine the degree to which the three basic requirements listed have been satisfied. It might be suggested that a unique feature of the Telstar satellite program has been the joint achievement of *depth* and *concurrency* in all four development phases. Such an approach has been dictated both by program urgency and the extraordinary cost of failure in an undertaking of this nature. In the process, each phase of the spacecraft development effort has served to reinforce collaterally every other phase, to a degree not normally realized in a sequentially structured program.

The spacecraft design is largely derived from existing systems of proven reliability. Maximum safety margins permitted by size and weight limitations have been utilized. Components and materials have been selected wherever possible on the basis of successful prior use in related applications. Manufacture and assembly operations have been carried out under carefully controlled conditions. The spacecraft test evaluation has served to demonstrate the effectiveness of these measures in providing a design which would survive the launch and operate satisfactorily in the orbital environment. This article will cover both general and detailed aspects of the spacecraft test evaluation, as the program progressed through the basic-development, design-qualification and flight-acceptance phases.

In the development phase, testing effort was directed toward the evaluation and qualification of spacecraft components and subassemblies, and the study of full-scale models. These tests were conducted under the mechanical and thermal conditions associated with launch and operation in orbit. It was during this phase, through a process of selection and elimination involving available alternatives, that the final design was crystallized. This was followed by design-qualification tests on the prototype spacecraft. In this phase, the controlling philosophy was to achieve maximum assurance, in the limited time available, that the design was capable of surviving launch and performing the intended function under conditions anticipated in orbit. To fulfil this objective, test conditions were selected in a manner which introduced a margin of severity beyond the specific environments predicted for the operational satellite. Finally, flight-acceptance tests performed on operational models were designed to reveal defects which may have been introduced in the manufacturing process. In this phase, the selected test conditions reflected the best estimate of the actual launch environment, but retained, in the thermal-vacuum portion, the added margin previously included in the prototype evaluation. This latter consideration recognized an element of uncertainty in predicting long-term thermal response of the satellite in space.

II. PRELIMINARY MODEL AND SUBASSEMBLY TESTING

The purpose of the preliminary test program was to evaluate the mechanical and electrical design of parts and subassemblies prior to their installation into an actual spacecraft. The tests on structural models with dummy electronics packages provided the necessary data to formulate the subassembly vibration test levels and durations. The levels for temperature tests were based on design objectives and calculated values expected in the space environment.[1] Once a subassembly design had been successfully qualified, there was reasonable assurance that it would survive the dynamic environment of powered flight on the launch vehicle and that it would operate properly in the temperature range predicted for orbital flight.

2.1 *Tests Performed on Development Models*

To conserve time and allow parallel testing, five satellite development models were assembled and used to provide data on system responses to environmental conditions. The use of models provided information required for subassembly design and pinpointed the possible trouble areas prior to the prototype phase of the project. This allowed any necessary changes to be made before assembly of the prototype. The models used were two structural models (referred to as the vibration and mechanical models), two thermal models, and one electrical development model. The latter, as the name implies, was assembled primarily to evaluate the spacecraft electronics as a complete system, but it also served to provide data on electrical performance under certain environmental conditions. A tabulation of the test models appears in Table I.

2.1.1 *Vibration and Shock Tests*

Vibration tests were performed on the two structural models to evaluate the mechanical design. Vibration levels were determined at various points in the structure when the entire system was subjected to sinusoidal and random vibration over a wide band of frequencies. The areas to which the most attention was directed were the nylon lacing supporting the electronics canister, the solar cells and their method of attachment, and the structure itself. Results of these early tests proved that from the standpoint of vibration: (a) the mechanical design was basically sound, (b) the solar cells would survive a vibration environment more severe than that expected during launch, and (c) that the nylon lacing provided sufficient isolation of the electronics canister for the anticipated environment. On the basis of these results, it was possible to predict the vibration levels which would be experienced by any of the various subassemblies located either inside the electronics canister or on the outer structure of the spacecraft. The vibration response of the electronics canister is shown in Fig. 1. It is evident that

TABLE I — SPACECRAFT TEST MODELS

Model	Panels	Antennas	Electronic Chassis
Vibration	Brass modules to simulate solar cells	No VHF antenna. Brass dummy for microwave	Dummy-weighted to simulate actual chassis
Mechanical	Full complement of actual solar cells	All antennas installed	Dummy-weighted to simulate actual chassis
Thermal model 1	Full complement of actual solar cells	All antennas installed	Dummy—not weighted
Thermal model 2	No solar cells	Microwave only	Dummy—not weighted
Electrical development	Full complement of actual solar cells	All antennas installed	Actual chassis, but not foamed

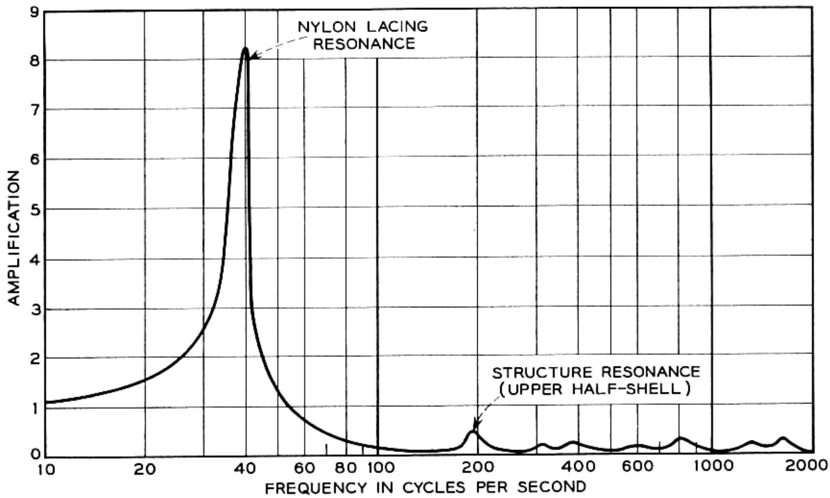

Fig. 1 — Amplification vs frequency, electronics chassis.

at frequencies between 54 and 2000 cycles per second the isolation provided by the lacing is very good.

Figs. 2 and 3 illustrate typical responses at two points on the outer structure. The peaks on these curves represent the resonant frequencies of various parts of the spacecraft. The frequency and amplification factor

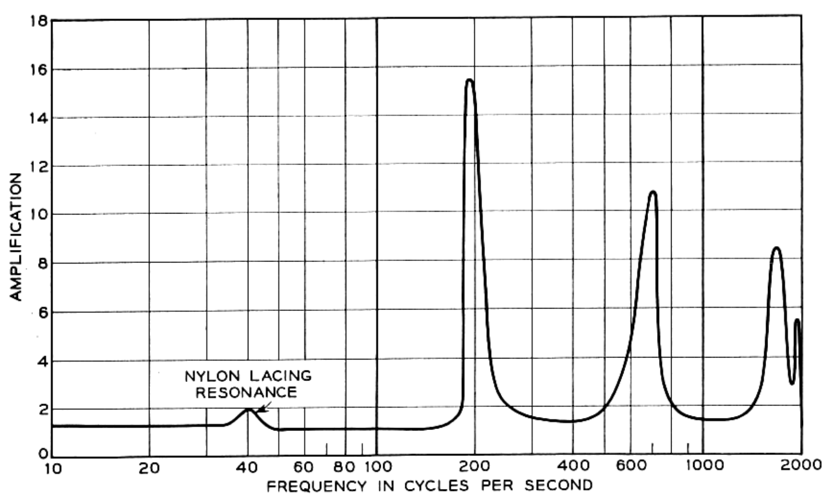

Fig. 2 — Amplification vs frequency, typical solar cell panel.

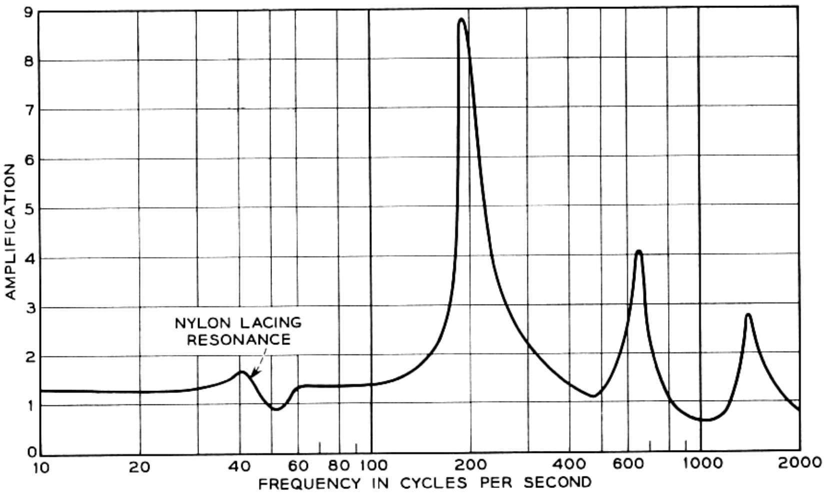

Fig. 3 — Amplication vs frequency, typical structural member (upper half-shell).

associated with each response provided the data necessary to determine the levels for qualification and acceptance vibration of subassemblies, prior to their installation in the spacecraft.

The vibration model was also subjected to several shock tests for the purpose of calibrating the shock test machine and determining the effect of shock on the satellite structure. These tests are summarized in Table II. Pulse shapes from four tests are presented in Fig. 4. The pulse shown in Fig. 4(a) was chosen for the subsequent prototype shock test. No deterioration of the structure occurred as a result of this test.

Table II — Preliminary Shock Tests

Drop Number	Drop Height (Inches)	Pulse Time (Milliseconds)	Maximum Acceleration (g's)
1	6.5	—	—
2	6.5	20	21
3	7.5	20	22.7
4	8.25	20	27
5	8.5	19.2	29.4
6	8.0	16.8	33.6
7	7.5	14.4	34.2
8	7.0	15.2	34.5
9	6.0	15.2	31.5

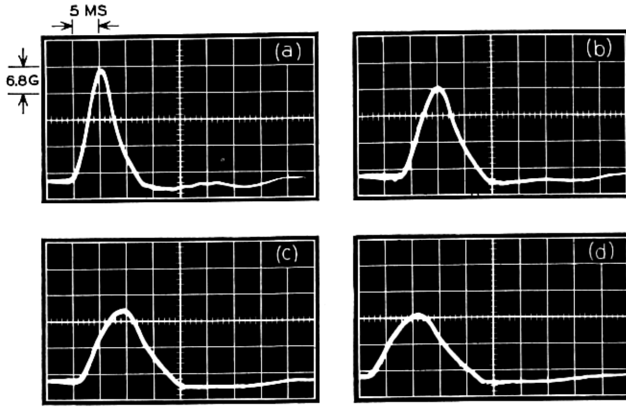

Fig. 4 — Pulse shapes formed using various arrestor rubber thicknesses with Barry 15000 vertical-drop machine: (a) 30-g peak — 13-millisecond pulse obtained with 1-inch rubber arrestor; (b) 24-g peak — 14-millisecond pulse obtained with 1.5-inch rubber arrestor; (c) 20-g peak — 18-millisecond pulse obtained with 2-inch rubber arrestor; (d) 17-g peak — 19-millisecond pulse obtained with 2.5-inch rubber arrestor.

2.1.2 *Preliminary Thermal-Vacuum Tests*

Several tests were performed with the two thermal models in the space simulator for calibration purposes and for verification of the predicted satellite temperatures under extreme orbital conditions. The results of these tests are summarized in Table III. The data obtained from the thermal models show that the chassis temperatures were between 15°F and 90°F. Further tests were performed to observe the thermal-shutter operation and the consequence of a failure of one or both shutter mechanisms. The results of these tests indicated that with the shutters closed in a fully sunlit orbit or open in a maximum eclipse orbit, canister temperatures of 92°F and −20°F, respectively, would prevail. The latter type of failure is the most critical with respect to the operation of the electronic circuits, but, by virtue of the fail-safe nature of the shutter mechanism design, this failure is also the least likely to occur. The shutters have been designed to avoid bearing surfaces which might seize. This leaves loss of fluid in the bellows assembly as the most likely mode of failure. Should this occur, spring loading will force the shutter to return to the closed position.

2.2 *Subassembly Qualification and Acceptance*

Each spacecraft subassembly was subjected to several vibration and temperature cycles prior to installation in the spacecraft. These tests

TABLE III — THERMAL-VACUUM TEST RESULTS

Spin Axis Parallel to Arc Lamps

Chassis Power	Simulated Orbital Condition	Canister Wall Temperature
16.0 watts	Fully sunlit (plus 18°F margin on skin)	74°F
7.0 watts	Maximum eclipse (minus 18°F margin on skin)	14°F

Spin Axis Perpendicular to Arc Lamps

Chassis Power	Simulated Orbital Condition	Canister Wall Temperature
16.0 watts	Fully sunlit (plus 18°F margin on skin)	87°F
7.0 watts	Maximum eclipse (minus 18°F margin on skin)	32°F

were performed to detect marginal components which may have slipped by quality control and to pinpoint manufacturing defects. A prototype of each subassembly was vibrated at qualification levels, determined from the results of the vibration tests performed on the structural models. Once a particular design was so qualified, the subsequent flyable units were subjected to acceptance level tests which were somewhat less stringent than those required for initial qualification. The vibration test levels and durations for qualification and acceptance of subassemblies located either inside the electronics canister or on the outer structure, are given in Table IV.

Most flyable subassemblies were temperature cycled over the range of 0°F to 125°F, stabilized at each extreme for a period of six hours and electrically tested at the maximum and minimum conditions. This test represented an extension of 15°F below and 35°F above the temperature range expected during the satellite lifetime.

III. PROTOTYPE QUALIFICATION TESTS

The design qualification program included a complete sequence of tests intended to subject the prototype model to environmental rigors more stringent than those expected from transportation, handling, test, pre-launch, launch, injection, and orbit. The prototype was deliberately over-tested to assure that the basic design possessed a margin of safety which would allow for variations in subsequent systems. The purpose of the design qualification tests was to demonstrate the ability of the design to meet all performance requirements without degradation due

TABLE IV

(a) Vibration Inputs for Inside-the-Canister Assemblies: Qualification Levels

Vibration Axis	Frequency Range (cps)	Sweep Duration (Minutes)	Acceleration (g's Vector)
Thrust	10–30	1.00	2.3*
	30–60	1.00	21.0
	60–2000	2.7	14.0
	550–650	0.5	21.0
Lateral	5–30	1.25	0.9*
	30–60	1.00	4.8
	60–2000	2.7	2.8
	550–650	0.5	4.8

(b) Vibration Inputs for Inside-the-Canister Assemblies: Acceptance Levels

Vibration Axis	Frequency Range (cps)	Sweep Duration (Minutes)	Acceleration (g's Vector)
Thrust	5–30	1.2	1.5*
	30–60	0.5	14.0
	60–2000	1.1	7.1
	550–650	0.5	14.0
Lateral	5–30	1.2	0.6*
	30–60	0.5	4.2
	60–2000	1.1	1.5
	550–650	0.5	4.2

(c) Vibration Inputs for On-the-Frame Assemblies: Qualification Levels (Typical)

Vibration Axis	Frequency Range (cps)	Sweep Duration (Minutes)	Acceleration (g's Vector)
Thrust	10–2000	1.7	2.3†
	150–300	0.5	46.0
	300–2000	1.5	20.0
	550–650	0.5	27.0

(d) Vibration Inputs for On-the-Frame Assemblies Acceptance Levels (Typical)

Vibration Axis	Frequency Range (cps)	Sweep Duration (Minutes)	Acceleration (g's Vector)
Thrust	10–2000	1.7	2.3
	150–300	0.5	28.0
	300–2000	1.5	14.0
	550–650	0.5	21.0

* Maximum table excursion — 0.5 inch double amplitude.
† This run was to observe possible fixture resonances.

to exposure to these more stringent environments. The spacecraft test sequence is tabulated in Table V, and the levels required for design qualification in Table VI. The specific test requirements were jointly agreed upon by Bell Telephone Laboratories, Incorporated, and the National Aeronautics and Space Administration.

3.1 *Balance, Moments of Inertia and Center of Gravity Determination*

To insure correct performance during launch and orbit, the mechanical parameters of weight, center of gravity location, moments of inertia, and degree of unbalance must be accurately measured, and corrected if out of established limits. Briefly, the part each parameter plays in the over-all performance of the satellite is as follows: The permissible weight is determined by the desired orbit and the type of launch vehicle used; for the Telstar satellite, the chosen orbit and Thor-Delta vehicle limited the final weight to a maximum of 175 pounds. The center of gravity location and moments of inertia affect the satellite attitude stabilization in orbit, and also influence corrections required during the guided portion of powered flight. The moments of inertia also determine the size of the rocket motors to provide spin for stabilization of the third stage. The static and dynamic balance of the satellite determine the degree of stability during third-stage flight and orbit.

Since the Telstar satellite is spin-stabilized, it is necessary that the major moment of inertia: (a) be greater than the remaining principal moments of inertia, and (b) have its axis coincident with the desired spin axis. The first requirement may be met by designing for correct

Table V — Spacecraft Test Sequence

Qualification Tests	Acceptance Tests
Leak	Leak
Electrical Acceptance	Electrical Acceptance
Static Balance	Static Balance
Spin	
Dynamic Balance	Dynamic Balance
Weight	Weight
Moments of Inertia	Moments of Inertia
Temperature	
Humidity	
Sustained Acceleration	
Shock	
Vibration	Vibration
Dynamic Balance	Dynamic Balance
Thermal Vacuum	Thermal Vacuum
Leak	Leak
System Check	System Check
Electrical Acceptance	Electrical Acceptance

TABLE VI — SPACECRAFT QUALIFICATION TEST LEVELS

Test	Levels		Total Time
	Thrust	Transverse	
Vibration			
5–50 cps	2.3 g peak	0.9 g peak	3.3 min.
50–500 cps	10.7 g peak	2.1 g peak	3.3 min.
500–2000 cps	21.0 g peak	4.2 g peak	2.0 min.
550–650 cps	40.0 g peak	14.0 g peak	1.0 min.
Random (20–2000 cps)	11.5 g rms	11.5 g rms	8.0 min.
Temperature			
Low temperature soak	0°F		6 hours
High temperature soak	140°F		6 hours
Low temperature operate	15°F		6 hours
High temperature operate	90°F		6 hours
Humidity	86°F–95%		24 hours
Thermal vacuum	pressure $<10^{-5}$ mm Hg		
High temperature	max. predicted MRT +18°F		6 days
Low temperature	min. predicted MRT −18°F		3 days
Sustained acceleration			
Thrust	+25 g		3 min.
Transverse (4 orientations)	±3 g		4 min.
Shock			
Thrust (3 drops)	30-g 10–15 millisec half-sine pulse		—
Spin	225 rpm		3 hours

mass distribution and the second by fine adjustment of the static and dynamic balance of the satellite. The design objective for the ratio of maximum transverse moment to the major moment of inertia was a ratio of less than 0.95.

The balance requirements imposed on the satellite by the launch vehicle were a center of gravity offset of less than 0.005 inch and a principal axis shift of less than 0.008 radian.

The prototype spacecraft was balanced to achieve a center of gravity offset of less than 0.003 inch and a principal axis shift of less than 0.002 radian. The moment of inertia ratio was found to be 0.956, which was slightly above the design objective; however, it was known at the time that the flyable satellites would have lower ratios, so no effort was made to optimize the prototype.

3.2 *Temperature and Humidity Tests*

The purpose of the temperature and humidity tests was to show that the spacecraft would survive temperature and humidity extremes that might be encountered during storage or operation in an uncontrolled environment for short periods. The levels employed for these tests were those which might result from a failure of either heating or air-conditioning equipment in the satellite storage and testing areas.

The tests were performed as follows: storage for a period of six hours at an ambient temperature of 140°F; storage for six hours at a temperature of 0°F; operation at a chassis temperature of 15°F; and operation at a chassis temperature of 90°F. The humidity test consisted of a 24-hour storage period at a temperature of 86°F and a relative humidity of 95 per cent.

During the temperature test, it was found that the telemetry failed to operate below 60°F. The failure was isolated to a switching transistor in the voltage-controlled oscillator and the circuit was redesigned for later spacecraft.

Following the temperature and humidity tests, physical examination of the spacecraft revealed that the oxide coating had flaked off in some areas, and it was noted that several gray stains, which were later found to be caused by oil, had appeared during the test. As a result of these findings, steps were taken to prevent exposure of subsequent satellites to an oil-contaminated atmosphere.

3.3 *Vibration Tests*

The prototype model was subjected to the series of vibration tests outlined in Table VI. The following failures were noted during the vibration tests: (a) radiation detectors P1, P2, E1 became inoperative, (b) one of the static balance weights was torn loose from the framework and damaged a solar cell panel, (c) two helix wires on the VHF antenna were broken, and (d) several short circuits developed in the solar power plant.

It was found that two of the radiation detector failures had been caused by broken wiring at the canister header and the third by a defective diode. In order to prevent failures of the same type, the wiring harness was modified and a more severe diode screening process was instituted. The improved wiring harness and more reliable diodes were installed in all subsequent spacecraft.

The loss of the static balance weight was attributed to local stress concentrations in the satellite frame at the point of attachment. The stress was relieved by distributing the weight over a larger area, allowing more space between the screws used to attach each weight, and by providing some degree of mechanical damping through the use of epoxy between the frame member and the balance weight. As a result of these modifications, no further difficulty has been encountered with the balance weight mounting.

Close inspection of the breaks in the helix antenna revealed that in both cases the fracture had been started during fabrication. In subsequent units the possibility of breakage was minimized by increasing the radii of the various bends, inspecting the wire more closely before forming, and inspecting the completed antenna assembly for cracks by means of a dye penetrant.

Inspection of the solar power plant wiring revealed that the insulation had ruptured at points where it crossed sharp corners of the spacecraft framework and at cable tie points, resulting in short circuits. The wire was replaced with a wire having a higher density insulation and wrapped with Mylar tape at tie points and areas where the cabling crossed the framework.

3.4 *Shock Test*

The shock test consisted of four half-sine pulses, 30 g's in amplitude and 13 milliseconds in duration, applied along the thrust axis of the spacecraft. Performance of the satellite following the test was satisfactory and gave no evidence of deterioration as a result of the test. During impacts, however, the telemetry dropped out of synchronization on the second, third and fourth shocks. Investigation failed to indicate any reason for the loss of synchronization, and it did not recur during any of the later environmental tests.

3.5 *Thermal-Vacuum Tests*[1]

The thermal-vacuum tests were a simulation of the extreme thermal conditions expected in orbit.[1] For the qualification test, it was required that the satellite be exposed to simulated solar illumination at a pressure of less than 1.0×10^{-5} mm Hg, with a chamber-wall temperature of less than $-280°F$, for a period of nine days. A fully sunlit orbit was simulated for six days with power dissipation within the electronics canister corresponding to maximum efficiency of the solar power plant. This was immediately followed by simulation of the maximum eclipse orbit for three days with a dissipation within the canister corresponding to a solar plant efficiency of 68 per cent of its initial value, the condition expected near the end of Telstar's two-year life. During the six-day test, the simulated solar input was controlled to give a mean radiant temperature[1] 18°F above the value predicted for the fully sunlit orbit, and the power input to the electronics chassis was maintained at 16 watts. The mean radiant temperature during the three-day test was maintained 18°F below that expected in the maximum eclipse orbit and the power input was limited to 7 watts.

Electrical operation represented maximum use of the satellite in the two extreme conditions and was conducted on the following schedule: the communications experiment and telemetry was turned on for three half-hour periods each day, beginning at approximately 0900, 1200 and 1500 hours, and the telemetry was operated for the first five minutes of each remaining half-hour period.

Shortly after the start of the prototype test, telemetry data showed that the canister pressure had decreased, indicating that a leak had developed. An electrical check indicated that a failure had occurred in the microwave circuitry. The test was terminated at that time and the satellite returned to the Hillside laboratory for disassembly and inspection.

It was discovered that the material around the top shutter mounting stud had failed, causing a leak in the dome of the canister. Since the failure had apparently occurred during either the vibration or shock tests, it served to emphasize the importance of performing the thermal-vacuum test after completion of the mechanical tests. Further investigation revealed the cause of the microwave failure to be destruction of the output transistors in the 255 and 277-mc transistor multipliers caused by voltage breakdown in the power supply resulting from loss of canister pressure. Laboratory experiment confirmed that the failure was a direct result of the corona effect present in the power supply when operated at reduced pressures.

The prototype unit was repaired and returned to the Whippany environmental test laboratory for completion of the qualification tests. The thermal-vacuum test was completed without further incident. The test results are shown in Table VII.

3.6 *Leak Tests*

Since the spacecraft electronics are contained within a hermetically-sealed canister and are designed to operate at a pressure of approximately 10 psia, it was necessary that the canister leak rate be evaluated. The

Table VII — Thermal-Vacuum Qualification Test Results

Test	Canister Temperature		Mean Skin Temperature
	Predicted	Actual	
Full sunlight	73°F	73.3°F	21°F
Maximum eclipse	14°F	17.3°F	−47°F

leak test also served as a means of determining if the canister had ruptured or if a weld had failed since sealing. The canister was backfilled with argon gas and the leak rate determined in an evacuated chamber, using a mass spectrometer. Leak tests were performed in this manner prior to and after the environmental test series.

Before testing, the prototype leak rate was less than the measurement capability of the detector. Following the test program, a leak rate of 5×10^{-4} std cc/sec of argon was indicated. Investigation revealed that the chamber, the Mylar insulating blankets around the canister, and the satellite itself, had become contaminated with argon at the time canister pressure was lost during the initial thermal-vacuum test. These findings raised serious doubt as to the significance of this final leak reading. Subsequent tests on flyable spacecraft have revealed that the leak rates are well below the specified 8×10^{-5} std cc/sec of argon.

3.7 *Spin Test*

The prototype was subjected to a spin rate of 225 rpm for a period of three hours, the last hour of which was under electrical load simulating operation in orbit. The 225 rpm spin rate was a 25 per cent increase over that expected in actual flight (the satellite was injected into orbit with an initial spin rate of 177.7 rpm). There were no spacecraft malfunctions during this test.

3.8 *Sustained Acceleration*

The prototype was mounted on a horizontal centrifuge and sustained acceleration, measured at the spacecraft center of gravity, was applied along each of its three coordinate axes. The first test consisted of one run with 25 g's applied in the thrust direction for a period of three minutes. The second test consisted of four runs in the lateral plane, with the prototype spacecraft being rotated 90 degrees after each run. The levels during the latter runs were 3 g's, maintained for a period of one minute in each position. The electrical performance of the spacecraft was monitored during the periods of constant acceleration, and electrical checks were performed following each run. There was no evidence that satellite performance was in any way degraded either during or as a result of the applied accelerations.

3.9 *Magnetic Drag and Moment Measurements*

Since the Telstar satellite is spin-stabilized and orbits within the magnetic field of the earth, it was necessary that the effect of that field on the spin rate and precession of the satellite be determined. For this reason, the torque due to eddy currents, and the residual magnetic moment along the spin axis of the spacecraft were measured. The former effect is the cause of spin decay and the latter causes precession.

The magnetic drag constant was found to be 1355 ± 15 per cent meters[4] per ohm. The drag torque due to a given field strength was calculated from the relationship $T_0 = p\omega B^2$, where p is the magnetic drag constant, ω the angular velocity of the satellite and B the field strength normal to the satellite spin axis. The flyable spacecraft were essentially the same as the prototype with respect to eddy current generation, and subsequent models were not drag tested.

The magnetic moment of the prototype was measured, and a coarse correction was made to investigate the feasibility of compensating for the effect of the traveling-wave-tube field.

IV. SPACECRAFT ACCEPTANCE TESTS

The spacecraft acceptance test program, illustrated in Table VIII, was similar to the qualification program but consisted of only vibration, thermal-vacuum, balancing, and measurement of weight, center of gravity and the principle moments of inertia. The vibration levels and durations closely approximated those expected during launch. The thermal-vacuum test was shortened to three days maximum sunlight and two days maximum eclipse, at the same levels required for qualification. The same requirements and objectives were met for balance, weight, center of gravity and moments of inertia as for the prototype. The remaining tests (spin, temperature, humidity, sustained acceleration and shock) which had been performed on the prototype model were for

TABLE VIII — SPACECRAFT ACCEPTANCE TESTS

Test	Levels		Total Time
Vibration	Thrust	Transverse	
5–50 cps	1.5 g peak	0.6 g peak	1.7 min.
50–500 cps	7.1 g peak	1.4 g peak	1.7 min.
500–2000 cps	14.0 g peak	2.8 g peak	1.0 min.
550–650 cps	40.0 g peak	9.4 g peak	0.5 min.
Random (20–2000 cps)	7.7 g rms	7.7 g rms	4.0 min.
Thermal Vacuum	Pressure 10^{-5} mm Hg		
High Temperature	max. predicted MRT + 18°F		3 days
Low Temperature	min. predicted MRT − 18°F		2 days

the purpose of design qualification and were not required for flyable spacecraft.

4.1 *Balance, Moments of Inertia and Center of Gravity Determination*

The flyable model spacecraft were subject to the same requirements for static and dynamic balance, moments of inertia, and center of gravity as the prototype model. The results of these measurements are shown in Table IX. All five spacecraft were well within all specification requirements and design objectives.

4.2 *Vibration Tests*

The vibration levels for acceptance testing of flyable spacecraft are shown in Table VIII. The acceptance tests for five flyable spacecraft were completed with no failures of in-line systems or serious malfunction of secondary systems. The two malfunctions which did occur were: flyable 2 — the radiation damage data assigned to telemetry channel 49 gave faulty information following sinusoidal vibration in the thrust direction (the same data are carried on channel 47, so no effort

TABLE IX — BALANCE, WEIGHT, C.G. DETERMINATION, AND MOMENTS OF INERTIA MEASUREMENTS

Parameter	NASA Requirement	Design Objective	Fly 1	Fly 2	Fly 3	Fly 4	Fly 5
Weight (lbs)	None	175.0 max	172.00	170.94	172.30	173.81	174.28
Spin-axis moment of inertia, slug-ft^2	None	Not specfied	4.164	4.141	4.207	4.202	4.1961
Maximum transverse moments of inertia, slug-ft^2	None	Not specfied	3.822	3.825	3.872	3.888	3.9184
Moment of inertia ratio	None	0.95	0.918	0.924	0.920	0.925	0.934
C.G. location (inches)							
From separation plane	None	16.5	16.48	16.46	16.36	16.4	16.26
Spin-axis offset	Less than 0.005	—	Less than 0.003*	Less than 0.003*	Less than 0.003*	Less than 0.003*	Less than 0.003*
Principle axis shift (radius)	Less than 0.008	—	Less than 0.002*	Less than 0.002*	Less than 0.002*	Less than 0.002*	Less than 0.002*

* Lower limit of reliable measurement.

was made to isolate the trouble); flyable 5 — following sinusoidal vibration in the thrust direction, telemetry channels 63 and 64 had no output, indicating a failure in one of the radiation detectors located on the skin of the spacecraft. It was agreed that the remaining two channels provided adequate coverage for the radiation experiment. Since complete isolation or repair would have been impossible without opening the canister, no corrective action was taken.

4.3 *Thermal-Vacuum Tests*

The flyable model spacecraft were exposed to the same thermal-vacuum conditions as the prototype, but for a shorter duration. The acceptance tests ran for five days (three days under maximum sunlight conditions and two days under maximum eclipse conditions). The flyable satellites were operated on the same duty cycle as outlined for the prototype.

Performance of the flyable spacecraft during thermal-vacuum tests was satisfactory in all respects. A tabulation of temperatures attained and chamber pressures for each test is presented in Table X. A graph illustrating the range of electronics chassis temperatures for five flyable satellites is shown on Fig. 5. Results from flyable model 2, which is the Telstar satellite, have been plotted for reference.

4.4 *Leak Tests*

The sealed electronics canister of each flyable model spacecraft was leak tested in the same manner as the prototype, before and after the acceptance test program. The results of these tests are shown in Table XI.

Table X — Thermal-Vacuum Acceptance Test Results

Satellite	Canister Temperature*		Mean Skin Temperature†		Chamber Pressure	
	Maximum Eclipse	Full Sunlight	Maximum Eclipse	Full Sunlight	Minimum	Maximum
Fly 1	43°F	88.5°F	−34°F	35°F	1.4×10^{-6} mm Hg	2.5×10^{-6} mm Hg
Fly 2	34°F	80.5°F	−40°F	27°F	1.0×10^{-6} mm Hg	2.1×10^{-6} mm Hg
Fly 3	43°F	83°F	−40°F	32°F	1.4×10^{-6} mm Hg	2.7×10^{-6} mm Hg
Fly 4	35°F	83°F	−35°F	36°F	4.8×10^{-7} mm Hg	9.1×10^{-7} mm Hg
Fly 5	38°F	78°F	−41°F	28°F	7.7×10^{-7} mm Hg	1.2×10^{-6} mm Hg

* Average temperature of four battery groups.
† Average of six points on skin.

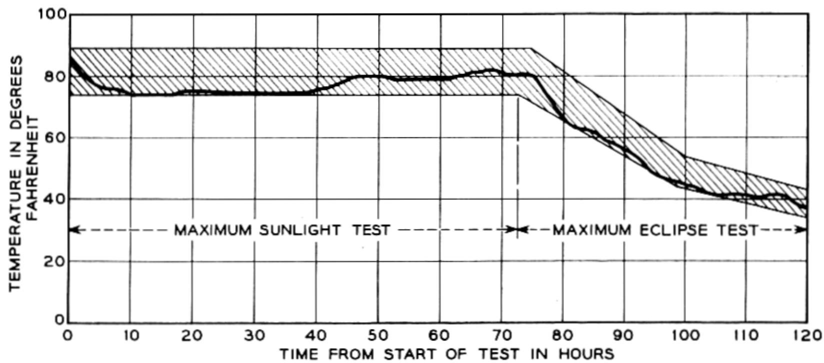

Fig. 5 — Electronic chassis temperature for five flyable satellites; flyable model 2 plotted for reference.

4.5 *Magnetic Moment Measurement*

The magnetic moment along the spin axis of the flyable spacecraft was measured and compensated to achieve the values indicated in Table XII. The plus and minus signs refer to the direction of the magnetic moment vector, the positive direction being toward the top of the satellite.

TABLE XI — LEAK TEST RESULTS

	Before Test	After Test
Fly 1	1×10^{-7} cc/sec argon	2×10^{-6} cc/sec argon
Fly 2	2×10^{-6} cc/sec argon	3×10^{-6} cc/sec argon
Fly 3	2.3×10^{-7} cc/sec argon	1.25×10^{-5} cc/sec argon
Fly 4	2.5×10^{-7} cc/sec argon	1.8×10^{-6} cc/sec argon
Fly 5	3×10^{-7} cc/sec argon	4×10^{-5} cc/sec argon

TABLE XII — MAGNETIC MOMENT MEASUREMENT (FLYABLE SPACECRAFT)

Satellite	Equivalent Torque in a 0.01-Oersted Field
Fly 1	1.9×10^{-6} ft-lbs
Fly 2	-3.3×10^{-6} ft-lbs
Fly 3	Less than measurement accuracy
Fly 4	-2.0×10^{-6} ft-lbs
Fly 5	Less than measurement accuracy

V. CONCLUSION

Components and materials selected for use in the Telstar satellite program were those which reflected a history of reliable usage in previous successful programs. The system design included the maximum practical margin of safety, commensurate with weight and space limitations of satellite systems. In addition to designing with a maximum safety factor, all manufacturing was accomplished under the closest possible control. Extensive testing of subassemblies and satellite models, carried out in parallel with the design and assembly of actual spacecraft, enhanced reliability of the finished product, and permitted timely modifications where required.

The environmental test program was designed to provide the most comprehensive evaluation of the spacecraft design in the time available. The qualification tests performed on the prototype provided assurance that the Telstar satellite was compatible with the environments expected during powered and orbital flight. The acceptance tests served to uncover manufacturing defects which may have developed during assembly of the various subassemblies into a completed system. Five flyable spacecraft have successfully completed the acceptance test program with no major failures or malfunctions in the in-line systems.

REFERENCE

1. Hrycak, P,. Koontz, D., Maggs, C., Stafford, J. W., Unger, B., and Wittenburg, A. M., The Spacecraft Structure and Thermal Design Considerations, B.S.T.J., this issue, p. 973.