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AEROBEE 350 ROCKET INSTRUMENTATION

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SUMMARY

This report describes the Aerobee 350 rocket instrumentation and telemetry systems. To date there have been a total of three rocket launches. The first launching, designated as Flight 12.02 Goddard Test and Support (GT), was an experimental rocket to evaluate the compatibility of the Aerobee 350 rocket and the rocket launcher at Wallops Island, Virginia. The last two rocket launches, designated as Flights 17.01 GT and 17.02 Goddard Test and Support, Goddard Ionospheric Physics, Goddard Solar Physics, University Ionospheric Physics (GT, GI, GS, UI), carried heavily instrumented payloads.

A description is given of the instrumentation and telemetry systems used for these rockets, as well as of the ground equipment used at the Wallops Island ground stations to check out, receive, and record the telemetry data for the launches.

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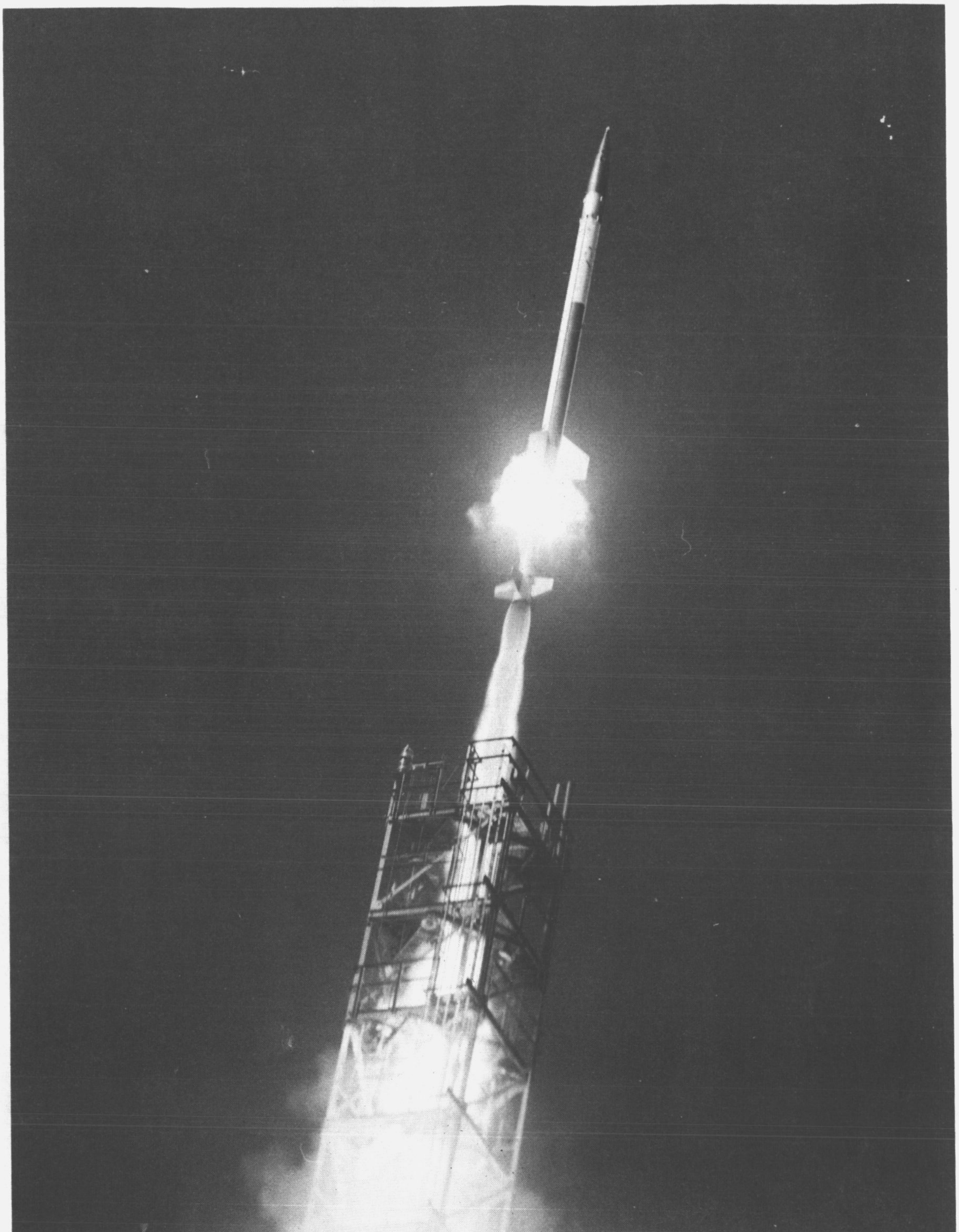


Frontispiece—Aerobee 350 Launch of Flight 17.02, at Wallops Island, Virginia

AEROBEE 350 ROCKET INSTRUMENTATION

INTRODUCTION

The beginning of the AEROBEE 350 program dates back to late 1961 when a proposal was submitted to Goddard Space Flight Center (GSFC) by Space General Corporation (SGC), prime contractor for AEROBEE sounding rocket vehicles. Under terms of this proposal, SGC would design, analyze, test, and document a new sounding rocket system, designated as AEROBEE 350. Studies of this proposal at GSFC were completed in March 1962, and a contract was awarded to SGC. The new system, consisting of an AEROBEE 350 sounding rocket vehicle, and attitude control system, would be capable of carrying a 150-pound payload (nominal weight) to an altitude of 290 statute miles, and would control the positioning of the payload in space.

Concurrent with the award of the contract to SGC, NASA/GSFC established the AEROBEE 350 Project Office within its Sounding Rocket Flight Performance Section. The Project Office, in turn, requested that a working group be established, within the Sounding Rocket Branch, in support of the program. This working group was composed of a mechanical engineer, an instrumentation engineer, and a flight performance engineer. Representing the Sounding Rocket Instrumentation Section, the instrumentation engineer was requested, by the Project Office, to design and develop instrumentation and telemetry systems capable of measuring and transmitting vehicle acceleration, attitude and motion dynamics, propulsion pressure, and temperature of both payload and vehicle.

AEROBEE 350 INSTRUMENTATION

FLIGHT 12.02

While the Aerobee 350 program was progressing toward a first complete evaluation, questions arose as to the compatibility between the new vehicle and the launch tower at Wallops Island, Virginia. (See Figure 1.) As a result, the Project Office decided to launch an experimental vehicle using a dummy sustainer atop the Nike booster. This experimental flight was designated as Flight 12.02.

The purpose of the instrumentation system for this flight was to obtain telemetry data which would be used to determine launch tower-vehicle compatibility, the effect on the sustainer during Nike boost when fully loaded with inert liquids (which simulated the weight of the propellants), and sustainer propellant

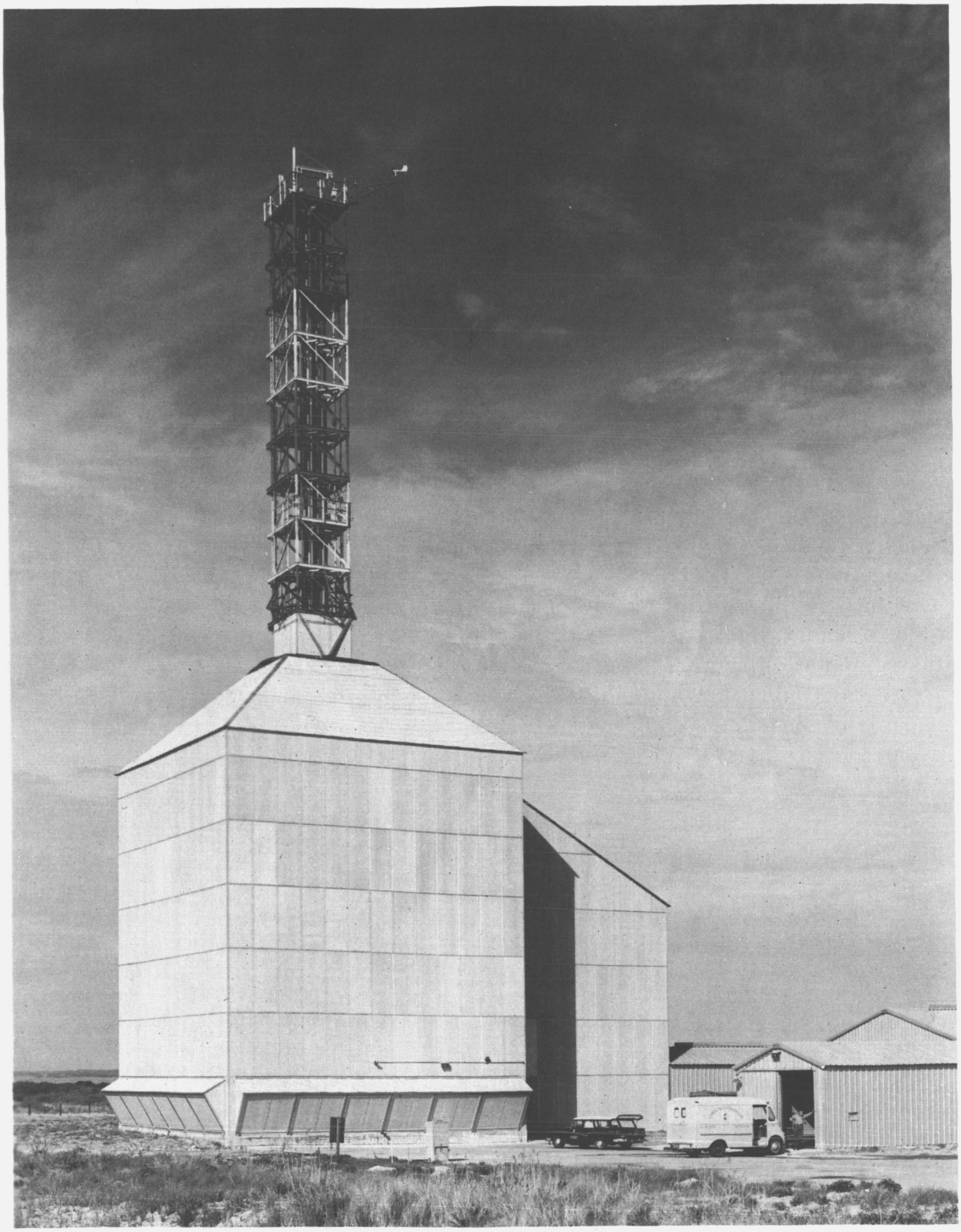


Figure 1. Launch Tower at Wallops Island, Virginia

tank pressurization characteristics under accelerations produced by first stage burning.

During the design of the basic Aerobee 350 tank assemblies, a question arose as to the placement of antennas on the sustainer, as it was desirable to standardize placement of the telemetry and command antennas. GSFC selected the small quadraloop antennas for use with the Aerobee 350, and indicated initially that two pairs of antennas were to be mounted on the sustainer regulator section as standard equipment. A revision was found necessary because of the excessive number of doors and access cutouts located in this region. Therefore, placement of the standard antennas was changed from the regulator section to a position forward of the helium tank, on the 8-inch cylindrical forward flange of the sustainer.

Flight 12.02 instrumentation included the following equipment.

(a) Four riding shoes with load cells incorporated, mounted in place of the standard riding shoes, to determine the tower-vehicle compatibility when correlated with tower-rail linearity, vehicle position versus time, and shoe-to-rail load measurements.

(b) A gyro roll-stabilized platform, mounted on the payload rack, to measure dynamic motion of the vehicle in roll, pitch, and yaw attitudes.

(c) An ogive sensor, mounted in the nose cone fairing, to provide aspect data on the rocket angle of attack (pitch and yaw), during the boosted portion of the flight.

(d) A three axis payload accelerometer, mounted on the payload rack to provide thrust data, pitch data, and yaw data.

(e) Two thermocouples, one mounted at the regulator inlet, in line with Fin IV, and one mounted at the regulator outlet, at the helium manifold. These were used to measure the gas bottle temperature at the regulator inlet, and the gas regulator temperature at the regulator outlet. Both thermocouples were of the type to measure temperatures within the range of -200 to +125 degrees F (-129 to +51.5 degrees C).

(f) Five pressure transducers, mounted in the tail section at the base of the propellant tank, and under the thrust structure blast deflector. Four of these were used to measure the propellant system pressurization during booster acceleration and one was used to measure the Nike booster chamber pressure.

(g) A three axis payload vibration sensor, mounted on the base of the forward payload, to provide tower-transit vibration data during launch.

(h) Eight strain gauges, installed on two struts of the booster thrust structure, to measure any bending moments of the structure during flight.

The instrumentation equipment installed in Rocket 12.02 is shown in Figure 2. In addition, this dummy flight included a vehicle recovery system. Recovery of the rocket after flight was desirable to analyze the tower-vehicle strike data. This subsystem, was also intended to illustrate the techniques to be used in an Aerobee 350 command safety system.

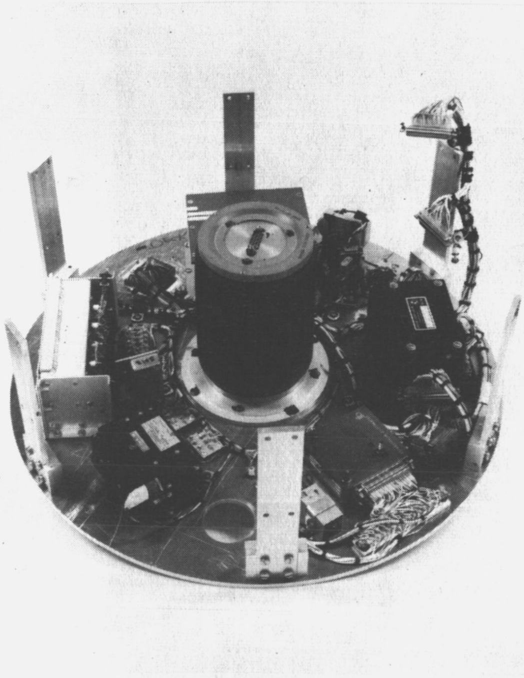
Flight 12.02 was successfully launched on December 11, 1964 at 1800 Z, from Wallops Island, Virginia. (See Figure 3.) Telemetry systems functioned normally, and good data were received throughout the flight. As a result of the flight of 12.02, the Wallops Island Range Safety Branch accepted the basic technique for the Aerobee 350 command safety system, and the launch tower - Aerobee 350 vehicle combination was considered as satisfactory.

FLIGHT 17.01

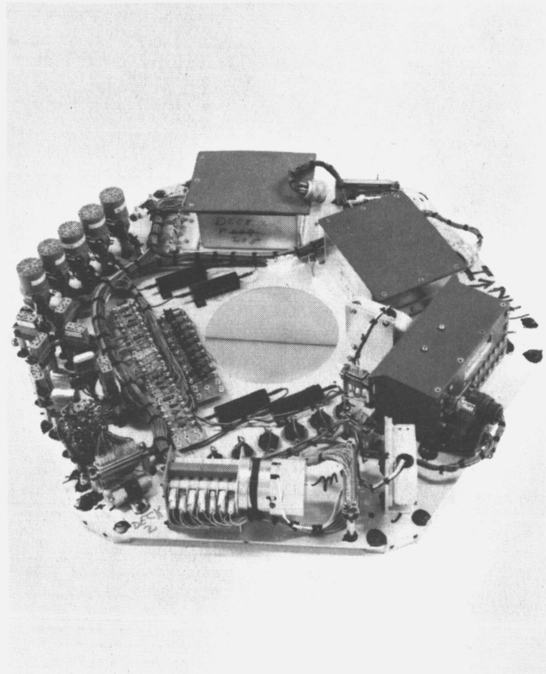
The instrumentation task for Flight 17.01 was far greater than that for Flight 12.02. Little comparison existed between most of the instrumentation designs for these two missions, however similarity did exist in those systems used for measuring temperature, vibration, magnetics, and pressure. See Figure 4 for instrumentation package carried on Flight 17.01. Figure 5 shows 17.01 being prepared for flight.

The instrumentation system on Flight 17.01 included four FM/FM telemetry systems, employing a total of fifty-four standard IRIG bands, and a sixteen-channel PPM telemetry system used to transmit performance data. Selected data were redundantly monitored by the PPM system to qualify this system for use on future Aerobee 350 payloads. For the FM/FM telemetry systems, signal conditioning components were utilized to condition the raw data, so that proper voltage levels of zero to +5 volts dc would be provided to modulate the telemetry systems. For data allocations of the telemetry systems, including the commutators, see Appendix A.

A problem developed in the design of the tail can instrumentation. Electrically, it was desirable to locate signal conditioning units for tail section performance instrumentation in the tail section, so as to limit the number of shroud line connectors and reduce noise. This meant that propellant cutoff components,



Deck No. 1



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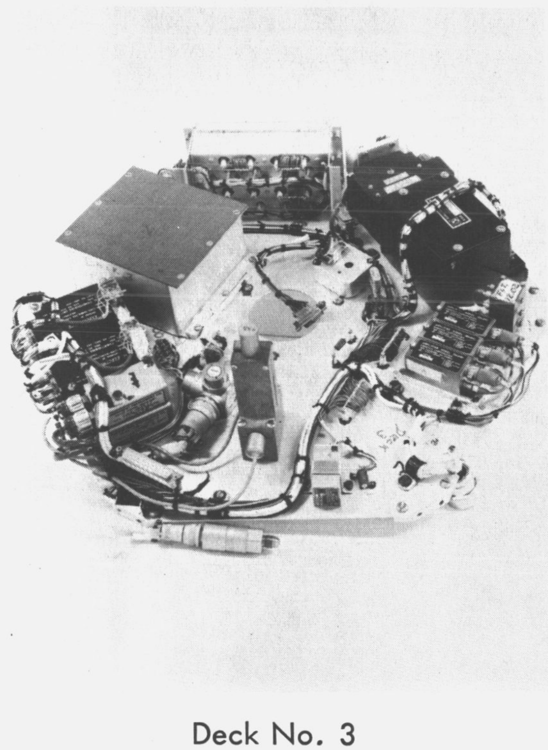
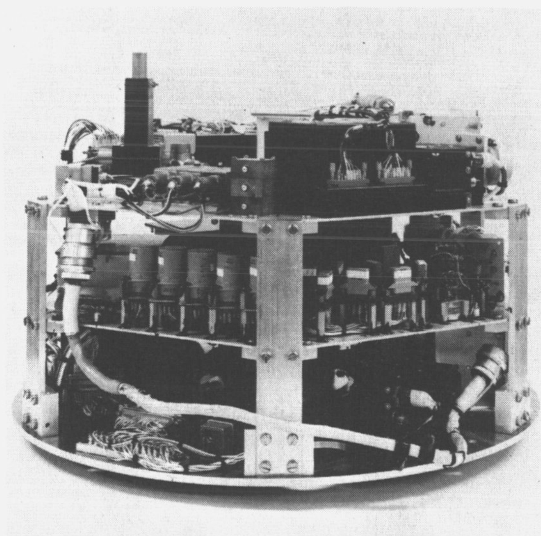




Decks No. 1-2-3 Assembled

Deck No. 3

Figure 2. Flight 12.02 Instrumentation Packaging

Figure 3. Launch of Flight 12.02

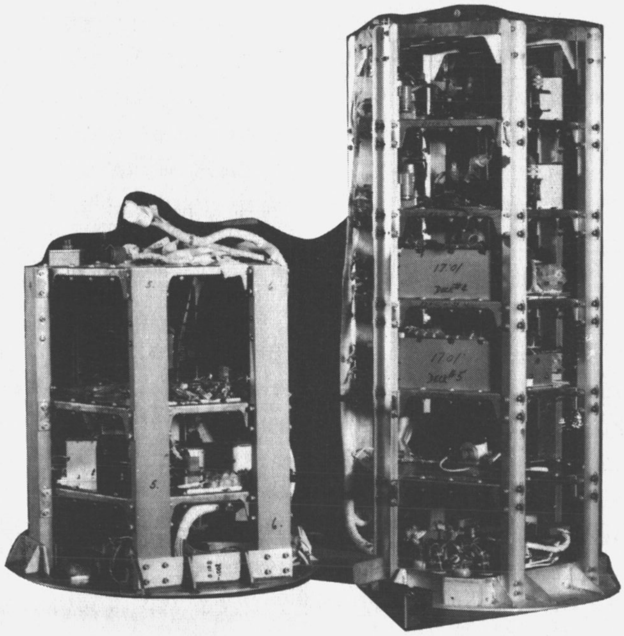


Figure 4. Instrumentation for Flight 17.01

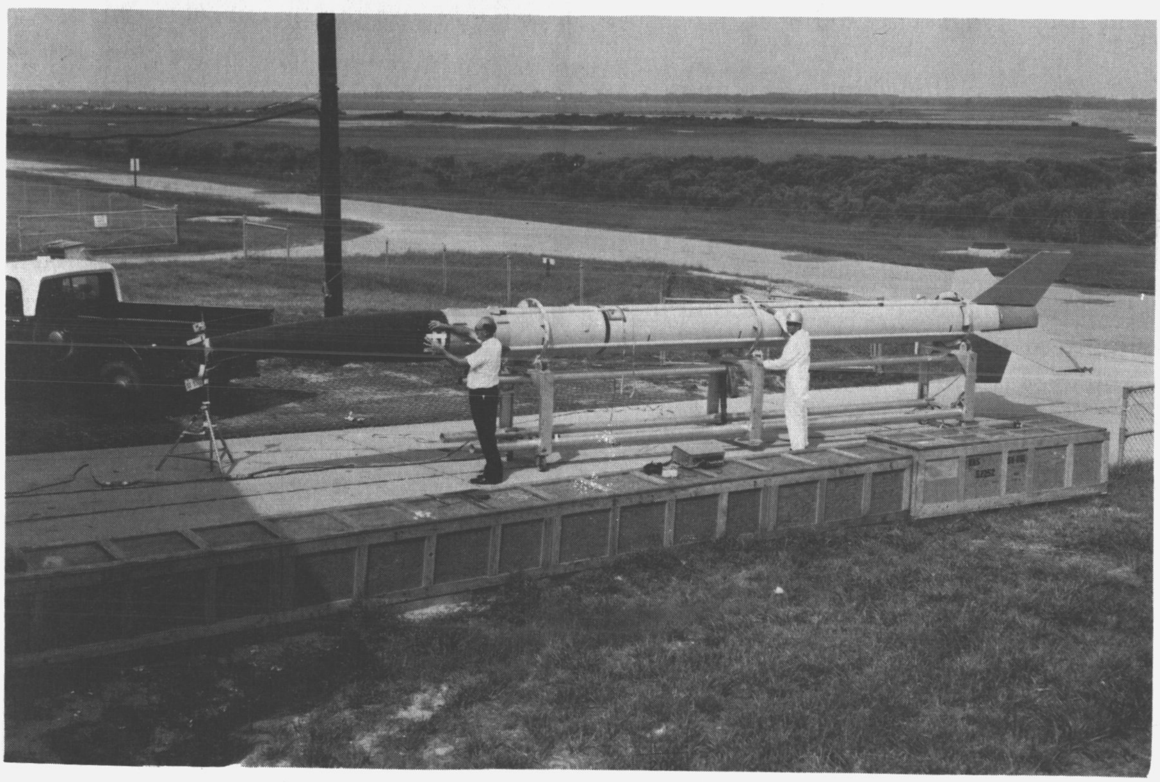


Figure 5. Rocket 17.01 Being Prepared for Flight

performance instrumentation, and signal conditioning components had to be mechanically designed into the tail section. Since an Aerobee 350 vehicle was at GSFC for a model survey (lateral and longitudinal frequency vibration tests), a design to house the tail section instrumentation was generated, using this vehicle. The design required an installation of four shelves, in the vicinity of the propulsion tank assembly aft ring, upon which were housed the signal conditioning components. A three axis vibration sensor was located at the center of the four thrust chambers on the longitudinal axis of the vehicle.

A need for three special oxidizer pressure gauges occurred when a problem was discovered during component installations on the sustainer at SGC. Originally, all pressure gauges had been calibrated by using an oil-filled dead weight tester. Small quantities of oil remained in the gauges following the calibration and subsequent testing. At the time of installation, oil was discovered in three of the gauges destined for inclusion in the oxidizer lines. Fuming nitric acid, which is the propellant oxidizer in the Aerobee 350, presents an explosive situation in contact with oil. Purging of the gauges was accomplished but the problem remained. Also, since the rocket could theoretically remain loaded with propellants for long periods of time before launching, a problem could occur with the silver "O" ring in these same three gauges. Three new gauges that contained gold "O" rings, which would resist the corrosive effect of the propellant for a longer period of time, were purchased from the manufacturer.

Three surface temperature gauges, mounted beneath the leading edge of the instrumented sustainer fin, were found to be inoperative during testing. The Project Office decided that substitute gauges would be installed on the outside surfaces of the fin, rather than to remove the units inside of the fin.

During the payload assembly phase, a problem occurred because of excessive payload noise. Noise problems were traced to ground loops, in the power supply and in signal grounds, and was resolved by providing a special common ground at a single point in the rocket.

A decision was made to decrease the data sampling rate of the two 5 revolutions per second data commutators to 2.5 revolutions per second, thereby increasing the data pulse width on both units, with no degradation to required data samples. This was considered an improvement to the overall system.

A temperature transducer was necessary to measure the normally-cool air of the engine compartment. Should the tail-can base-plate (across the tail can cylinder) start to deteriorate from the extremely hot exhaust, the transducer would, by telemetry, provide advanced warning of impending danger in this area. An air temperature sensor, having the temperature range requirements of 32 to 600 degrees F (zero to 316 degrees C) was mounted in the engine compartment.

Design of the command safety system ground control console was completed, and fabrication and wiring was started. Continuity and system checks were successfully conducted. Drawings of the command safety system ground control console were sent to personnel at Wallops Island Range Safety Branch. Match squibs, used during preflight system checks, were received and tested. Wallops Island indicated that they would furnish the command receiver batteries, as they do for all Aerobee 150A flights. The command safety system destruct (safe/arm) unit was successfully bench tested and work began on the assembly of the backup airborne unit. (See Figures 6 and 7.)

Wallops Island Range Safety personnel attended a complete checkout at GSFC, of both the command safety system and the payload instrumentation. Following the successful completion of the horizontal checkout, a critique was conducted by GSFC and Wallops Island personnel, and Wallops Island personnel accepted the system designs and operation.

A preliminary countdown was accomplished, and both the payload and the vehicle were prepared for shipment. Flight 17.01 was launched from Wallops Island on June 18, 1965 at 2311 Z. (See Figure 8.) The rocket performance was as predicted, and satisfactory flight performance data were obtained on all five telemetry systems.

FLIGHT 17.02

The third Aerobee 350 flight, designated 17.02, was the first flight of the Aerobee 350 for the purpose of gathering scientific information. The 17.02 was also a further evaluation of the Aerobee 350 vehicle performance for payload configuration (Figure 9).

The major differences between the flights of 17.01 and 17.02 were that of the scientific payload objective, and the telemetry and commutator allocations. Also, since the PPM telemetry system used on the 17.01 effectively proved itself to be satisfactory, it was used on Flight 17.02, not as a redundant system, but as an integral part of the 17.02 telemetry system.

Flight 17.02 was launched from Wallops Island, Virginia, on August 17, 1966 at 19.50 Z (3:50 EDT). (See Figure 10.) The rocket performance was as predicted. Flight performance data was satisfactory, and the results on the three scientific experiments were good. It was concluded that the Aerobee 350 was fully qualified for operational use with standard ogive-cylinder payload configurations.

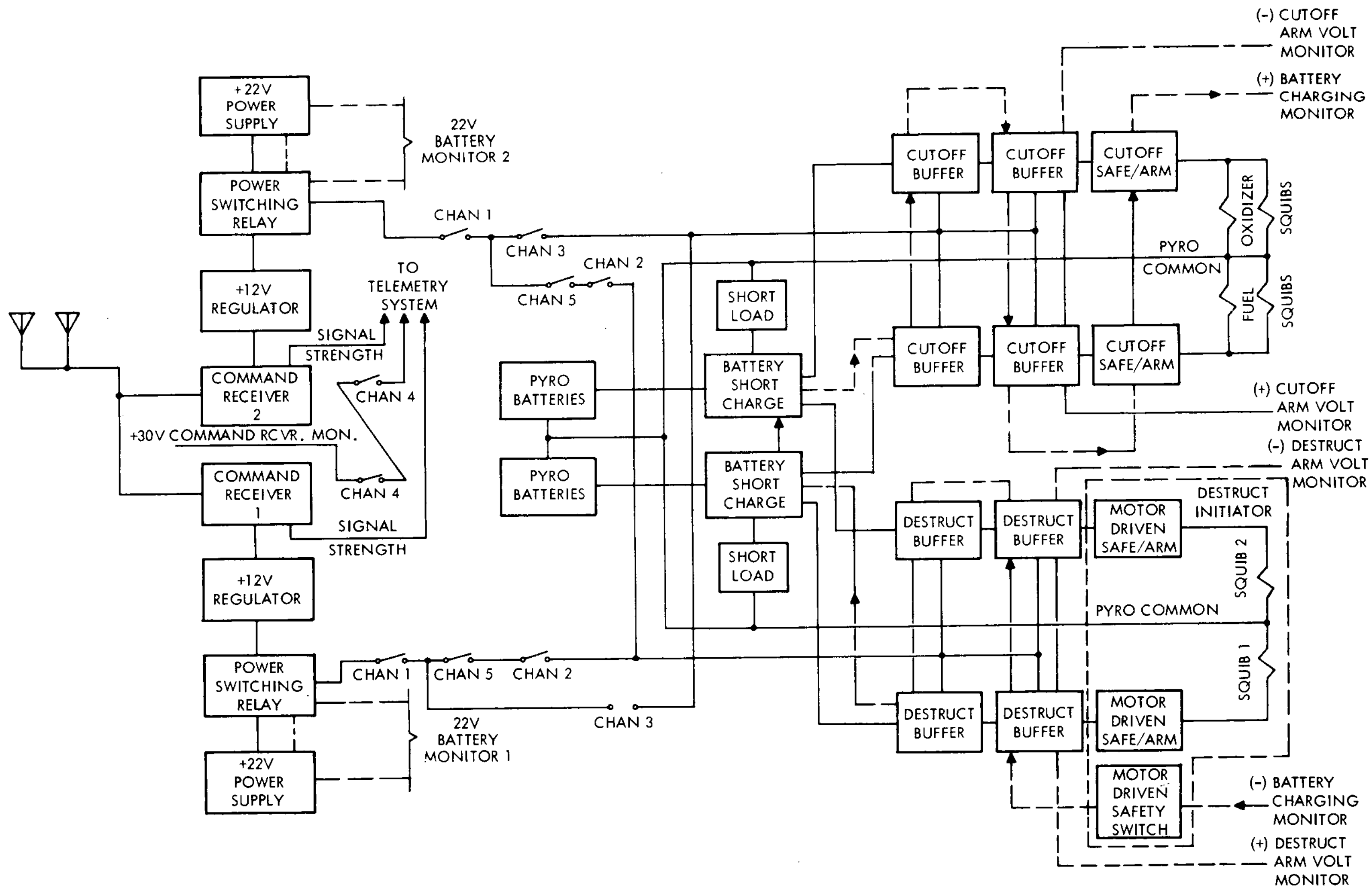
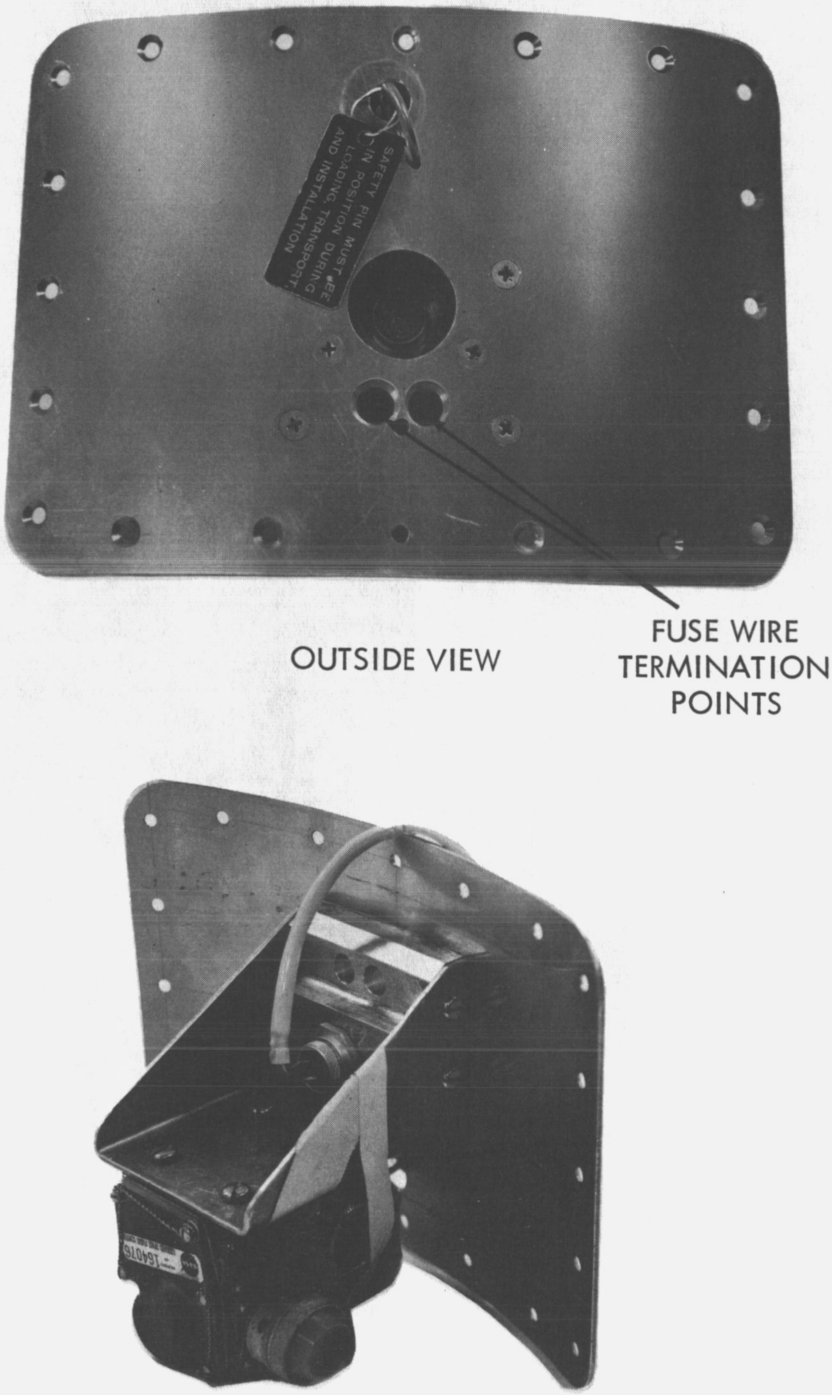


Figure 6. Finalized Command Safety System Block Diagram



INSIDE VIEW

Figure 7. Destruct Initiator Mounted on Regulator Door

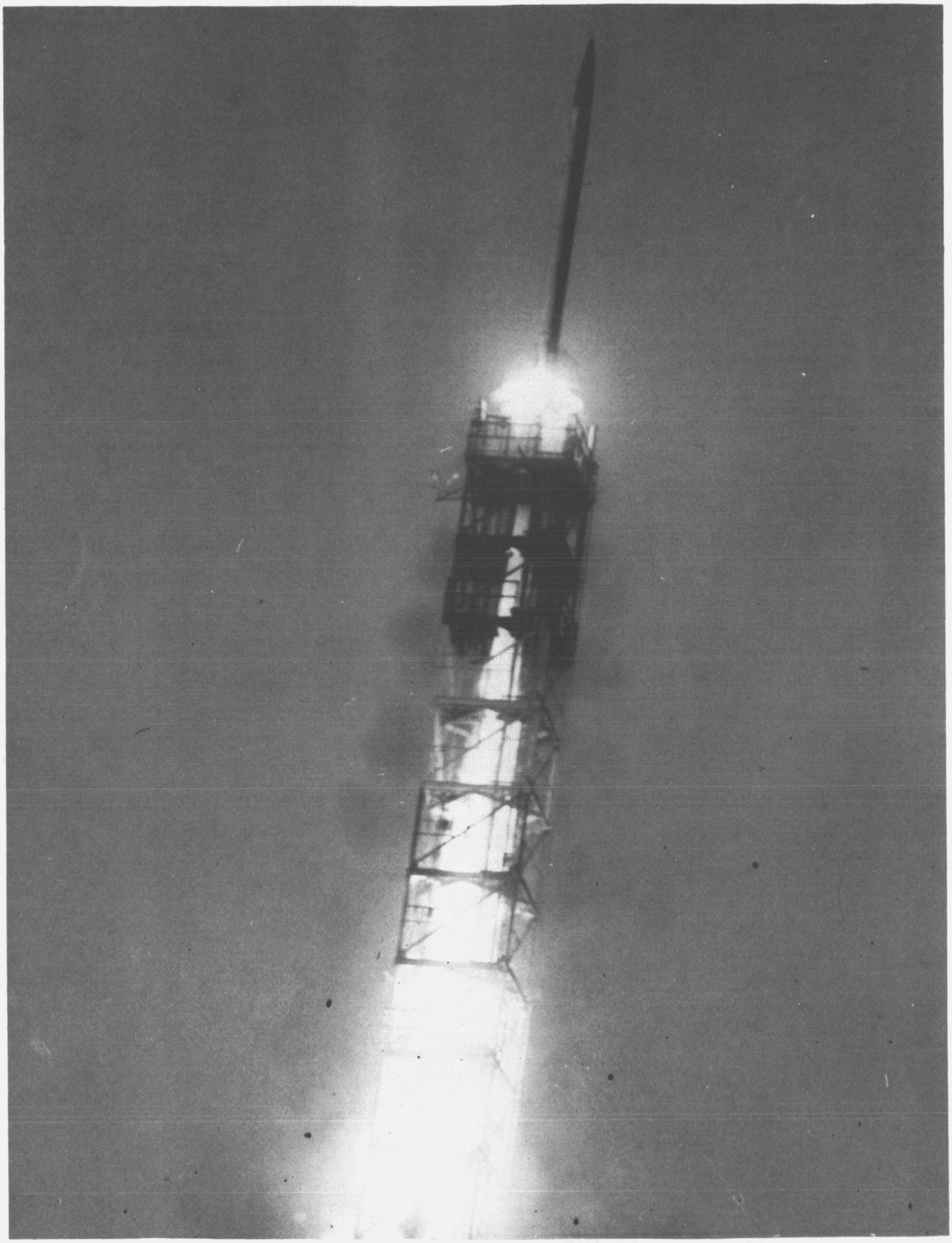


Figure 8. Launch of Flight 17.01

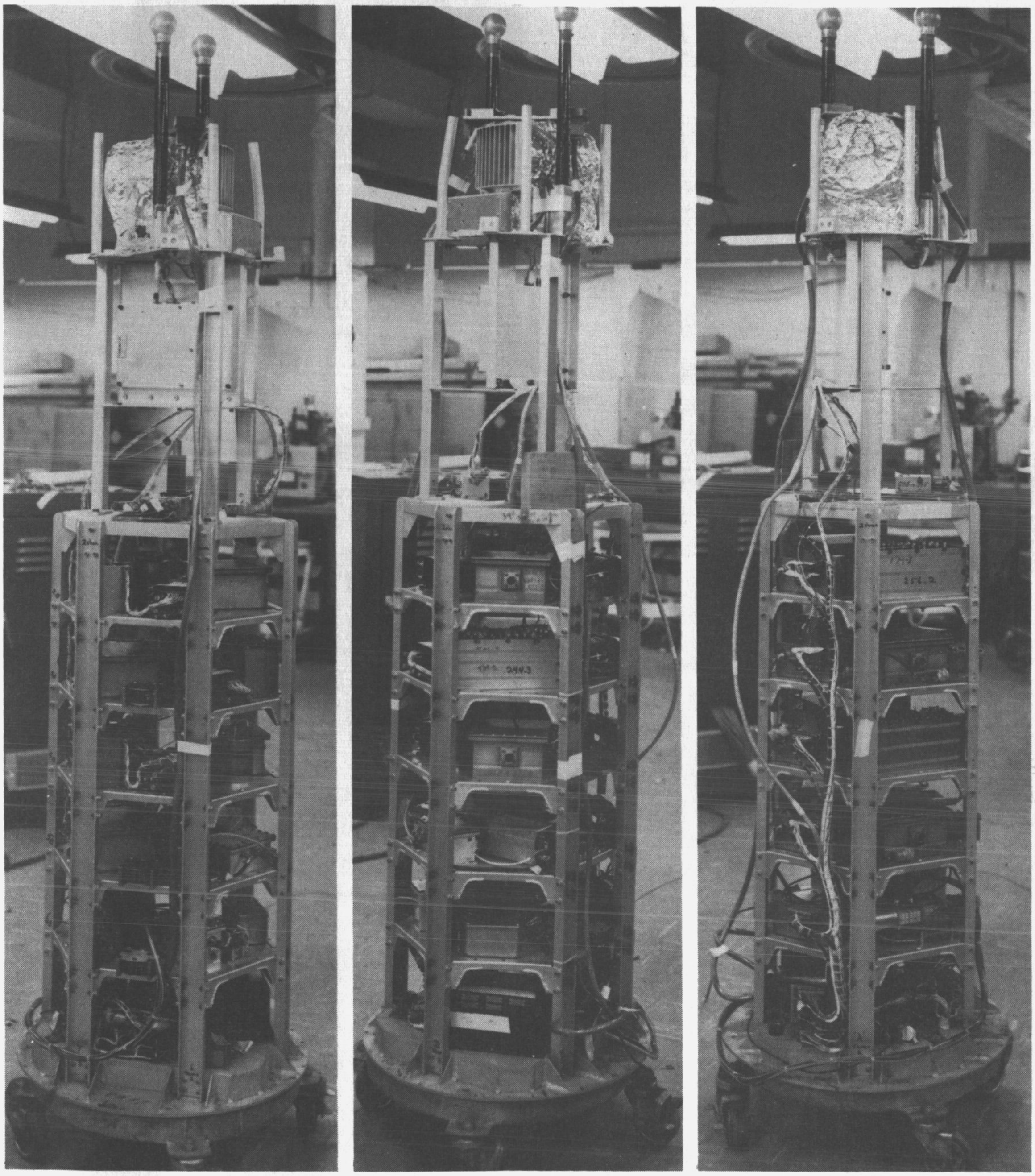


Figure 9. Three Views of Flight 17.02 Payload Instrumentation

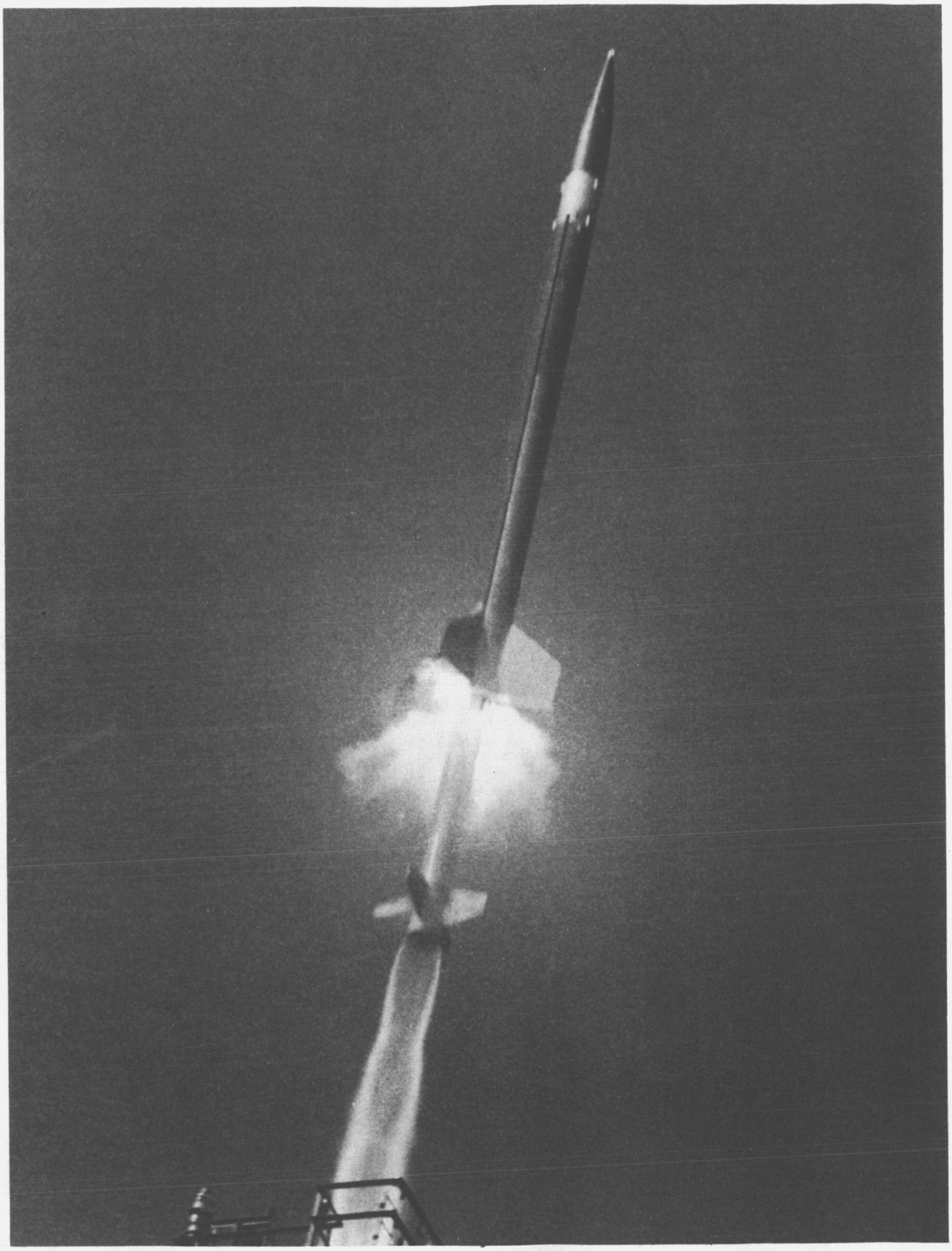


Figure 10. Launch of Flight 17.02

INSTRUMENTATION SYSTEMS

Instrumentation systems developed for the Aerobee 350 sounding rocket include systems to measure the vehicle attitude and motion dynamics, temperatures, rocket and payload pressures, vibration and strain.

ATTITUDE AND MOTION

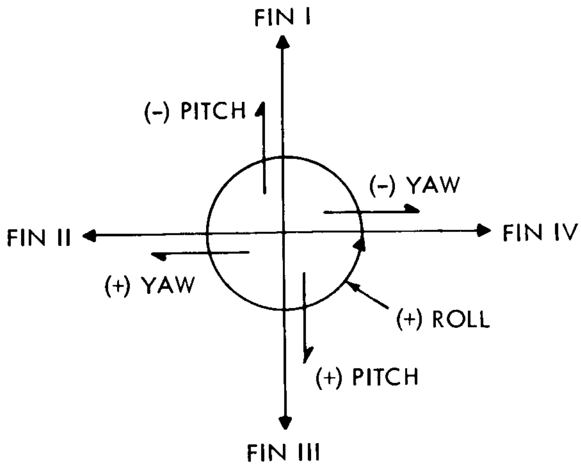
The systems designed to monitor the vehicle attitude and motion include the following:

- stable platform
- angle of attack transducer
- solar aspect
- magnetic aspect
- accelerometer

STABLE PLATFORM. A stable platform (gyro system) (Whittaker Model 518025 EC) utilizes the momentum of a spinning rotor to sense changes in angular motion of a rocket in flight. The spin axis of the gyro is common to the roll axis of the rocket, and it is this axis that is stabilized.

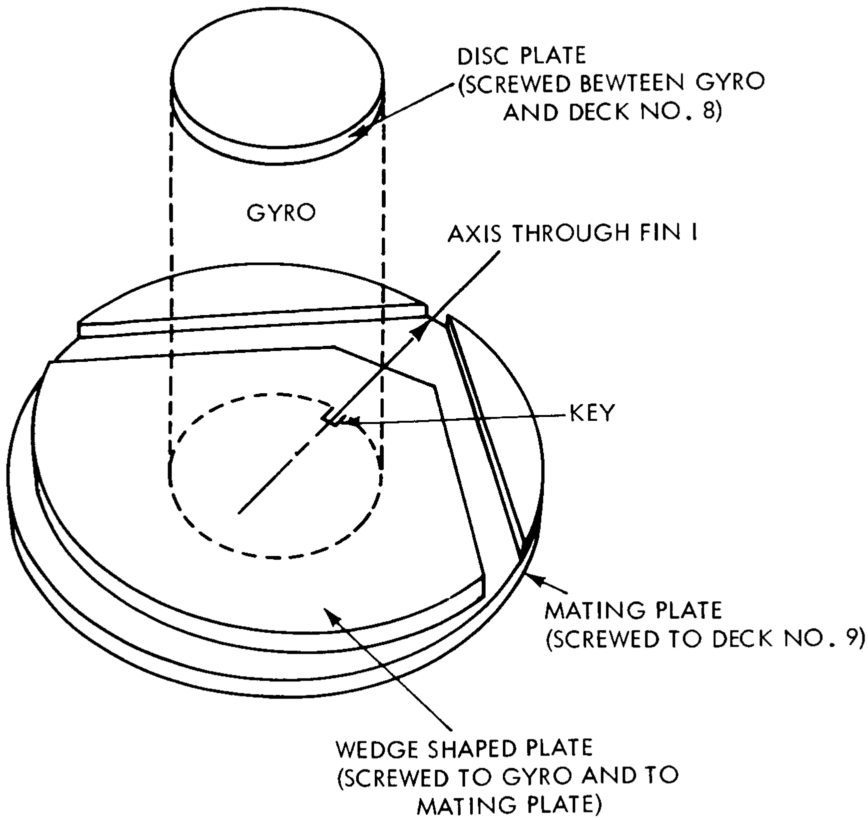
Any movement of the vehicle causes a corresponding change in the output of the potentiometers of the gyro. Voltage outputs are taken from a movable slide of each potentiometer and are within the voltage range of zero to 5 volts dc. Since this voltage is compatible with the telemetry systems employed, no signal conditioning unit is required, and outputs from each gyro potentiometer are directly applied to the telemetry system. For gyro mounting and orientation with respect to the vehicle, and potentiometer outputs, see Figure 11.

When the gyro is caged and operating, the pitch, yaw, and roll output signals of the system (Figure 12) are each at a nominal 2.5 volts-dc reference level, regardless of the position of the system. When the gyro is uncaged, the outputs remain at the 2.5 volts-dc reference level, but with a slight shift from the caged reference through mechanical tolerances, until the position of the gyro system is changed by movement of the rocket in flight. Deviations of the rocket result either in positive or negative (pitch, yaw, and roll) output, according to the angular deviation from the initial uncaged position. When the output reaches a



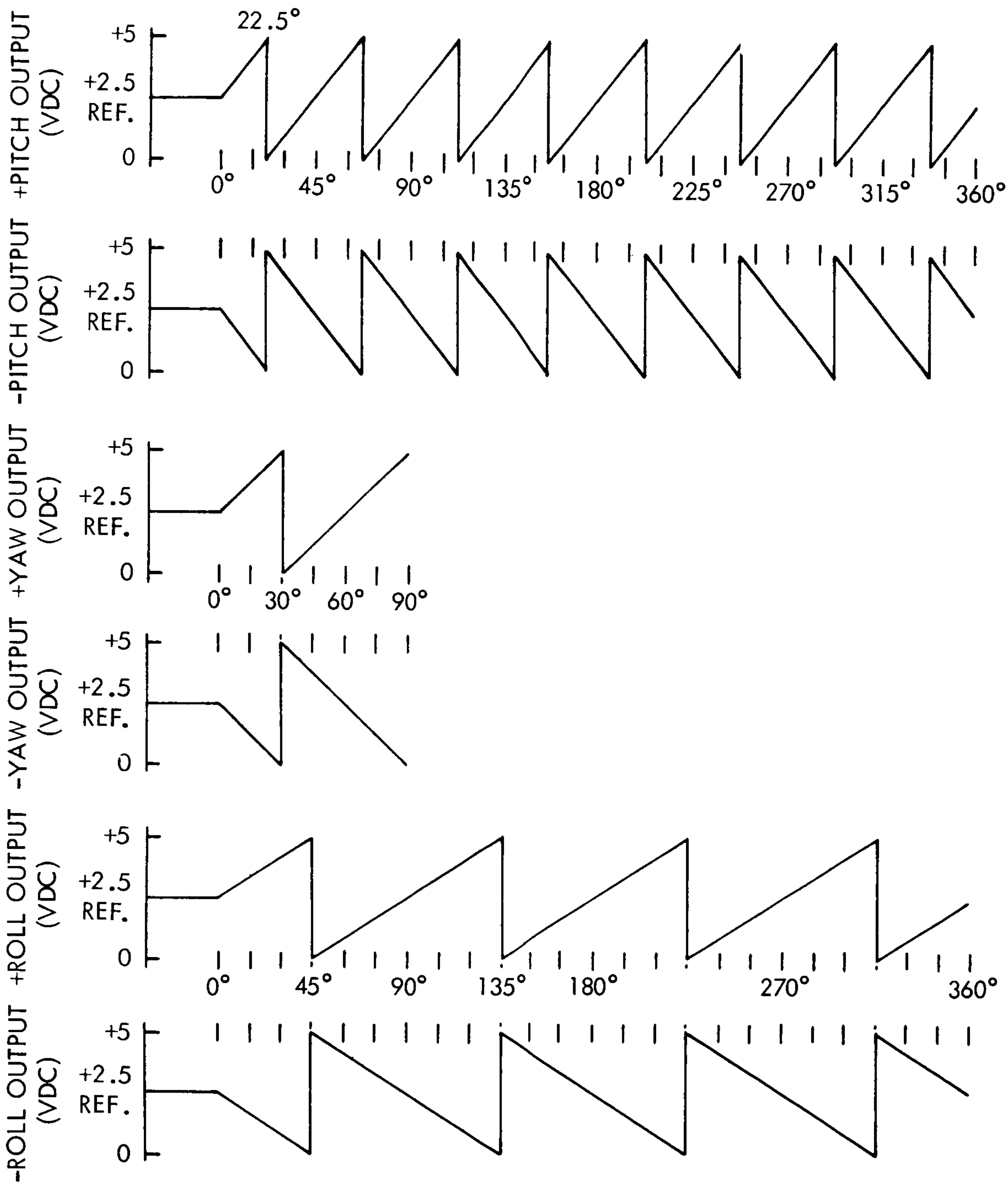
NOSE MOTION
LOOKING AFT

ORIENTATION



MOUNTING

Figure 11. Gyro System



ANGULAR VARIATION FROM INITIAL UNCAGED POSITION
(DEGREES)

Figure 12. Gyro Output Configuration

zero to 5 volt-dc level, it switches almost instantaneously to the opposite extreme level at a 5 or a zero volt-dc level, respectively. The three gimbals for pitch yaw and roll are driven by induction type motors. Associated with each gimbal is a segmented potentiometer-type pick-off, with each segment biased in parallel by a plus 5 volt-dc regulated source. The yaw potentiometer has three segments, each measuring 60 degrees; the pitch potentiometer has eight 45-degree segments; and the roll potentiometer has four 90-degree segments. Each segment has a resolution of 0.5 percent. The gyro system is operated from power supplied by the telemetry system batteries during flight.

Caging of the gyro system is accomplished by a caging motor, which mechanically locks the gimbals. Inputs of +30 volts dc are supplied from the ground console for performing this operation. Two outputs are provided for monitoring the caging system. When the gyro system is caged, the corresponding output signal is at ground potential, and the uncaged signal output is an open circuit. When the gyro system is uncaged by the solenoid, the reverse is true.

The gyro is mounted on deck 9, and secured to the bottom of deck 8. It is so oriented that the key on the bottom is directly in line with Fin I of the Aerobee 350 rocket. The gyro is attached to a modified wedge shaped plate, which is then screwed to a mating plate and fastened to deck 9. Precise measurements are made in locating the mounting holes to ensure that the longitudinal center axis of the gyro is exactly over the geometric center of the rocket. The top of the gyro is secured to the bottom of deck 8 by a disc shaped wedge to provide a snug fit between the two decks. This type of mounting offers a convenient means of removing the gyro, should a failure occur during testing, and of replacing the same or a new gyro in the exact location. Once the wedge shaped plate is unscrewed from its mating plate and the screws securing the top disc removed, the gyro, with the secured plates attached, can be slipped out of the extension rack. (See Figure 11.)

ANGLE OF ATTACK TRANSDUCER SYSTEM. An angle of attack transducer is a device which measures the angle of attack of the rocket as it travels through its trajectory. The device consists, essentially, of an ogival shaped shell with four protruding fins, each spaced 90 degrees apart. (See Figure 13.) The entire unit swivels, on a universal joint, about a stationary boom which is affixed to the tip of the payload nose-cone along the longitudinal axis of the vehicle. The ogive has two potentiometer pickoffs so mounted that one indicates the pitch angle, and the other the yaw angle. Mechanical stops are built into the unit to limit pitch and yaw angular displacement to ±8 degrees.

Theoretically, the rocket longitudinal axis falls directly along the fixed beam of the transducer, thus providing a point at the graphic intersection of both axes.

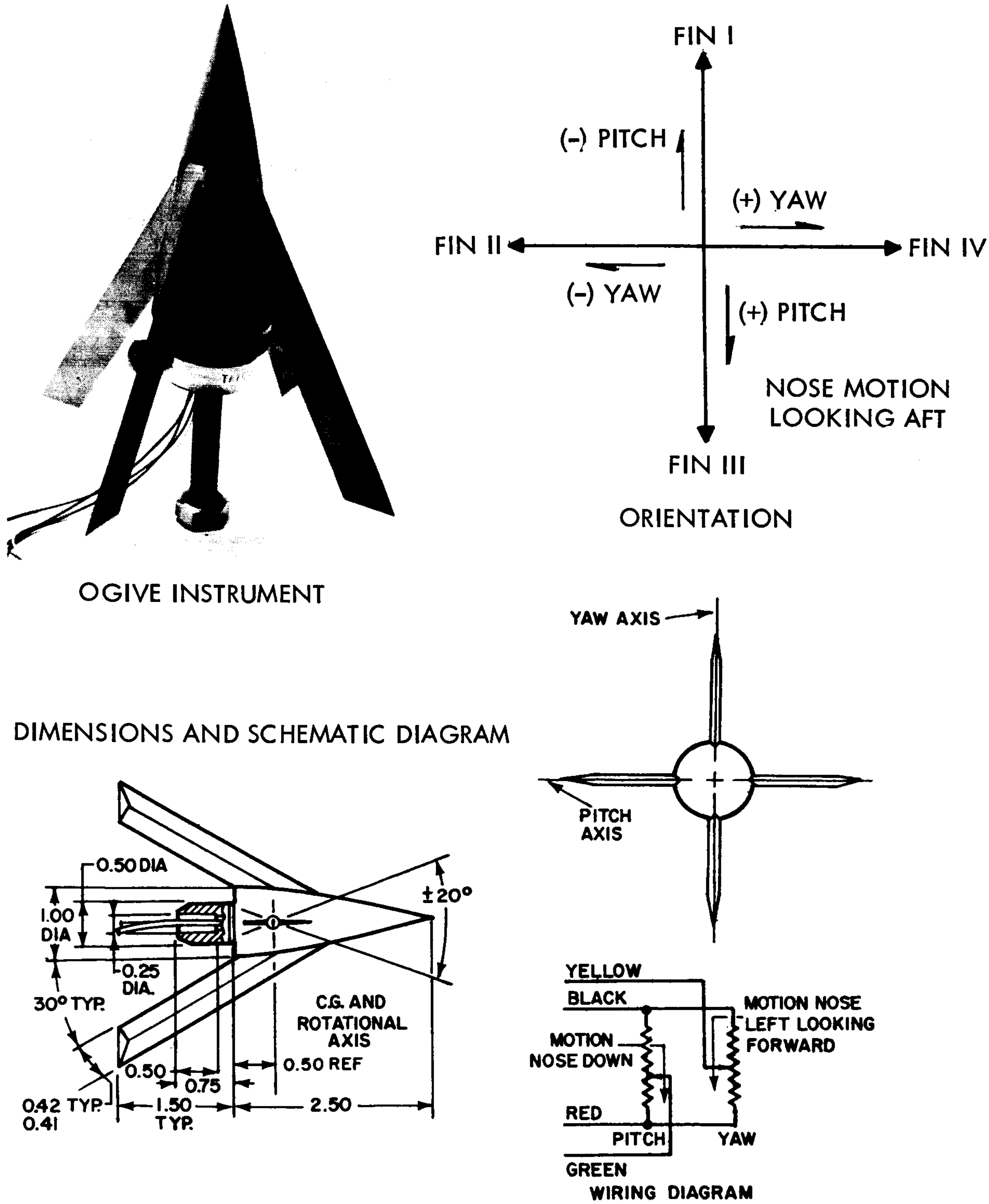


Figure 13. Angle of Attack Transducer System

Practically, however, because of tolerances in the balance of the rocket, and of the alignment of the vehicle and thrust chamber, the longitudinal axis of the rocket can not be in perfect alignment with the direction of flight. This results in an undesirable component of motion along the lateral axis of the rocket. See Figure 13 for ogive orientation.

Algebraically summed combinations of pitch and yaw angles describe the vehicle angle of attack. This angle represents the deviations from flight path, in both pitch and yaw directions, from the vehicle longitudinal axis. Figure 13 schematically illustrates the ogive system. A plus 5 volt-dc regulator is used in conjunction with the transducer. When an excitation voltage of +5 volts dc to ground is applied across the parallel-connected potentiometers by the +5 volts-dc regulator, outputs from the transducer are in a voltage range compatible with the telemetry system. When the pitch and yaw angles are both zero, both potentiometers are positioned midway between the potentiometer extremes, producing pitch and yaw output reference levels of +2.5, ±0.07 volts dc. In flight, the transducer shell aligns itself with the windstream and, from the zero degree reference position, produces -2.5 to zero volts-dc pitch and yaw outputs, for zero to -8 degree deviations, respectively. For the zero to -8 degree deviations, outputs of -2.5 volts dc to +5 volts dc are produced. (See Compilation of Calibration Curves for the Aerobee 350, Flight 17.02-GT-GI-GS-UI, Goddard Space Flight Center Document X-721-66-468.) The transducer resolution is such that a ±0.2 degree variation produces a change in output voltage.

The linearity of the transducer is 2.0 percent of full range. The full operable range of ±8 degrees of the transducer is compatible with the full scale range of the regulator, which is zero to 5 volts dc. Compatibility with the telemetry system is also governed by a one-megohm input impedance of each of the telemetry system voltage-controlled subcarrier oscillators (VCO). The potentiometer for each axis of the ogive transducer is rated at 2000 ohms full scale. The output of each potentiometer, taken at approximately mid-scale (1000 ohms), is placed in parallel with the one-megohm impedance of the VCO, thus providing a minimal input error to the telemetry system of 0.1 percent. Compatibility between the transducer system and the telemetry system is thus satisfied. One other modification to the ogive system remains. A 1000-ohm resistor is added to the monitor output of the system as a protection device for the regulator. Should a short-circuit occur between deck cabling and the regulator, the presence of the resistor prevents damage to, and loss of the regulator. The voltage drop across this 1000-ohm resistor is negligible when added to the signal voltage across the one-megohm resistance of the VCO. Thus, an accurate voltage monitor of the ogive system is obtained.

SOLAR ASPECT SYSTEM. Another method used to obtain attitude and dynamic motion data on the Aerobee 350 sounding rocket system is through the use of a solar aspect sensor system. As the name implies, attitude data is gathered to obtain the vehicle position with respect to the sun. Reduced data from this system provides a digital measurement of the angle between a line pointing at the sun and the coordinate system of the rocket. To obtain this information, two types of solar aspect sensors were used on Flights 17.01 and 17.02.

On Aerobee Flight 17.01, the solar aspect sensor used was an Adcole Model 135. The Model 135 sensor (Figure 14) consists of a command reticle, a data reticle, and an earth reticle. The command reticle activates a photocell mounted directly under its eye. When the photocell is activated, a command is generated for the sun sensing eye to read out its data in a binary Gray-coded format. See Table I.

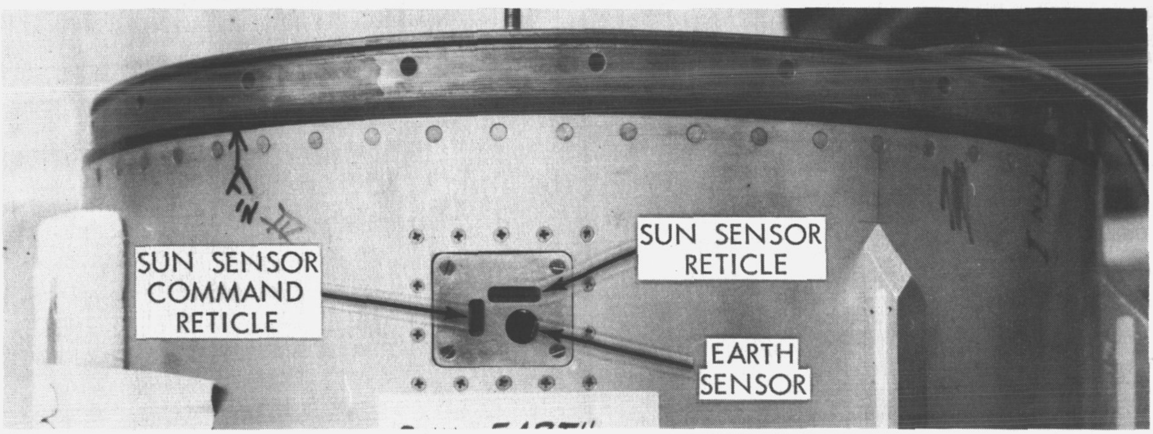


Figure 14. Solar Aspect Sensor, Model 135, Used on Flight 17.01

Located under the sun sensor eye are seven photocells. Sunlight, passing through the slit of the Gray-code reticle, is screened either to illuminate or not to illuminate each of the photocells. The seven photocells provide a pattern of 128 unique combinations of zeros and ones, representing a coverage of 128 degrees in one-degree increments (plus and minus 64 degrees). Outputs from each photocell are amplified and stored in a shift register.

The earth sensor consists of a telescope lens and a photocell. Tilted at a 10-degree angle with respect to the lateral axis of the rocket, the earth telescope illuminates a photocell when the earth airglow is in its field of view. The amount of light on the photocell determines the reference level of the Gray-coded word at the output of the shift register.

On the Aerobee Flight 17.02 the solar aspect sensor used was an Adcole Model 133. (See Figure 15.) Unlike the Model 135 used on Flight 17.01, the

Table I
SOLAR ASPECT SENSOR (MODEL 135) CODE TABLE

ANGLE	GRAY CODE	ANGLE	GRAY CODE
-63.5	0000000	-30.5	0110001
-62.5	0000001	-29.5	0110011
-61.5	0000011	-28.5	0110010
-60.5	0000010	-27.5	0110110
-59.5	0000110	-26.5	0110111
-58.5	0000111	-25.5	0110101
-57.5	0000101	-24.5	0110100
-56.5	0000100	-23.5	0111100
-55.5	0001100	-22.5	0111101
-54.5	0001101	-21.5	0111111
-53.5	0001111	-20.5	0111110
-52.5	0001110	-19.5	0111010
-51.5	0001010	-18.5	0111011
-50.5	0001011	-17.5	0111001
-49.5	0001001	-16.5	0111000
-48.5	0001000	-15.5	0101000
-47.5	0011000	-14.5	0101001
-46.5	0011001	-13.5	0101011
-45.5	0011011	-12.5	0101010
-44.5	0011010	-11.5	0101110
-43.5	0011110	-10.5	0101111
-42.5	0011111	- 9.5	0101101
-41.5	0011101	- 8.5	0101100
-40.5	0011100	- 7.5	0100100
-39.5	0010100	- 6.5	0100101
-38.5	0010101	- 5.5	0100111
-37.5	0010111	- 4.5	0100110
-36.5	0010110	- 3.5	0100010
-35.5	0010010	- 2.5	0100011
-34.5	0010011	- 1.5	0100001
-33.5	0010001	- 0.5	0100000
-32.5	0010000	+ 0.5	1100000
-31.5	0110000	+ 1.5	1100001
		etc.	

Positive angles same as negative except most significant bit is a 1.

Model 133 sensor can consist of up to six sun sensor eyes, with no command reticle or earth telescope. A combination of two sensor eyes scans 128 degrees, in two planes. Three of these units, each placed 120 degrees from the other, were mounted on the nose cone. This combination provides a field of view of 360 degrees throughout the flight. In addition, when the nose cone tip was ejected, two other Model 133 sun sensors were used to gather data in support of a solar experiment. In this attitude, two sensors are required to prevent loss of data in the event that the view of one of the sensors is blocked. Orientation of the three nose cone sensors is illustrated in Figure 16.

The "B" reticle reads out seven binary bits of position data, which positions a point for that reticle in a plus or minus position for that axis. The "A" reticle reads out an additional seven bits of position data, which positions the point for the second reticle in a plus or minus position for the other axis, similar to the plotting of a point on a graph. The resultant data are fed into the shift register which reads out the 14 bits of position data, followed by three identification bits and an end of word bit. (See Figure 17.)

When the shift register used with the Model 135 is triggered by the command reticle, an eight bit word consisting of zeros and ones and having a duty cycle of 50 percent, is produced. The most significant bit indicates whether the sun is at a plus or minus angle with respect to the rocket lateral axis. The next six bits determine the sun's angle of incidence in a 64 degree range, and the least significant bit indicates which model of aspect sensor is being used. The bit rate of the output of Model 135 can be varied up to a rate of 1000 bits per second by changing the value of an external capacitor on the shift register.

When the shift register used with the Model 133 is triggered by the presence of information from the sensors, it allows for time sharing of up to six sensors by reading out the one eye that is most directly facing the sun. The bit rate of the Model 133 system is not variable.

Power is supplied to either solar aspect system from a +30 volt-dc power supply associated with the shift register.

A block diagram of a Solar Aspect System for a spinning rocket with one sensor is shown in Figure 18.

MAGNETOMETER SYSTEM. Another method used to obtain aspect and dynamic motion information of the vehicle is the measurement of the magnetic field strength through which the vehicle travels. Magnetometers (Schonstedt RAM-5c) were used to obtain the magnetic field measurements with respect to both the transverse and longitudinal axes of the rocket. With the aid of a trajectory

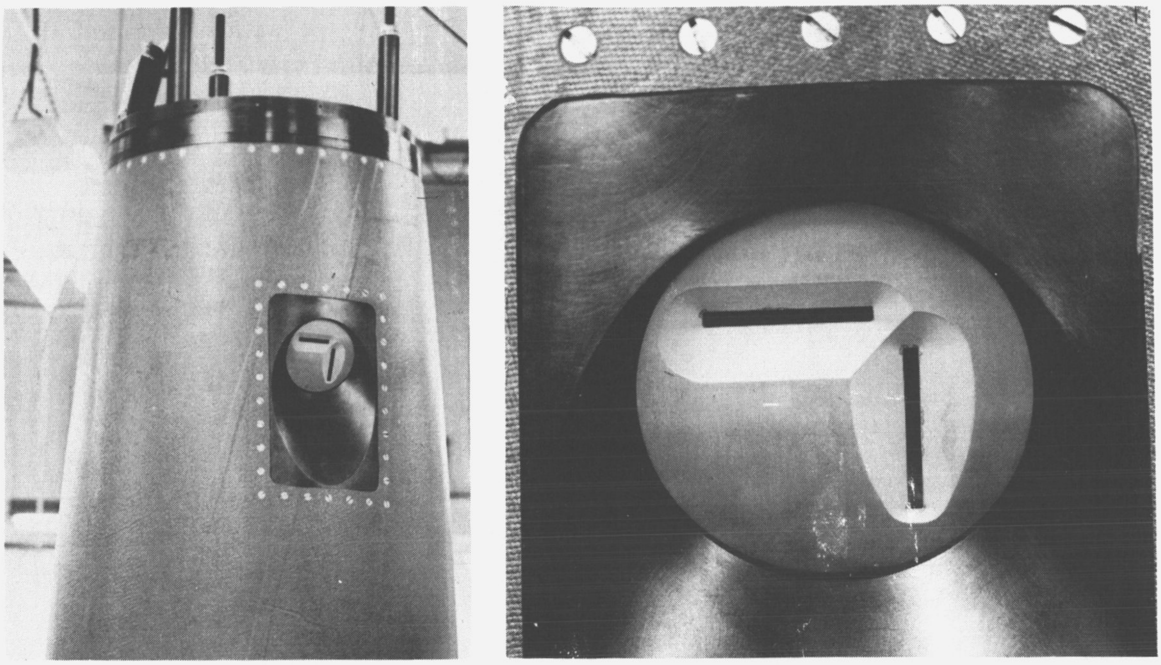


Figure 15. Solar Aspect Sensor, Model 133, Used on Flight 17.02

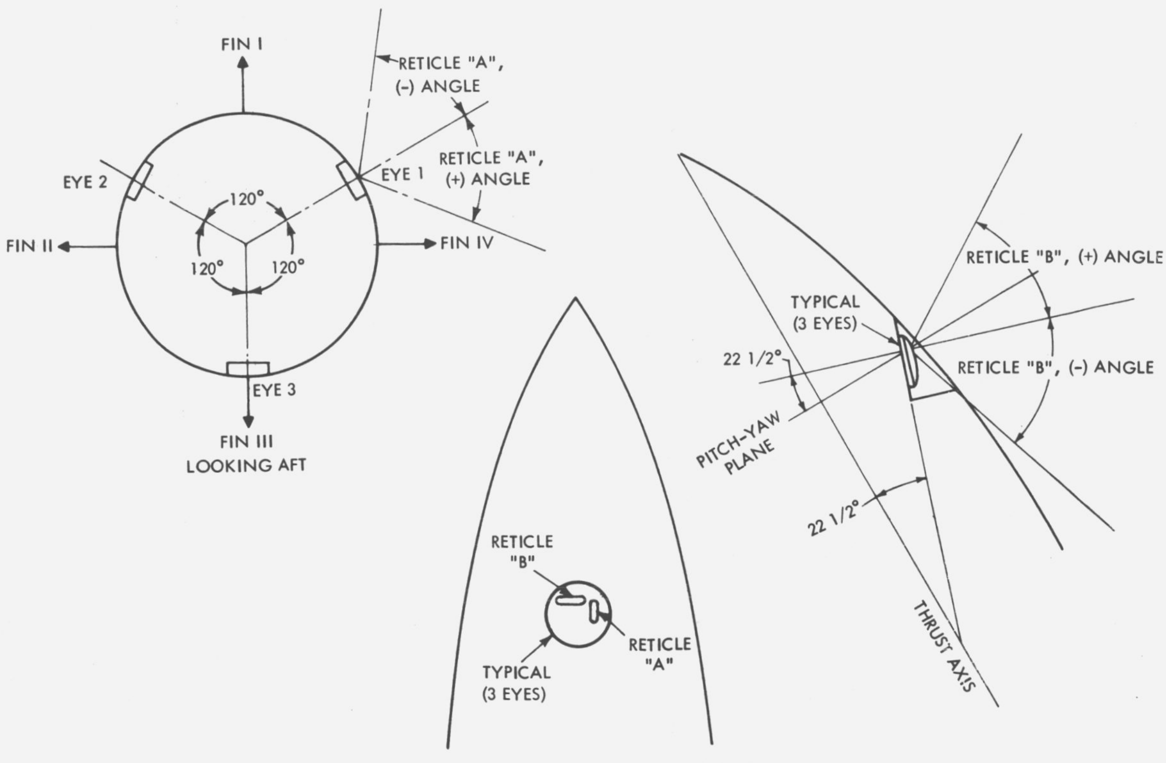


Figure 16. Solar Aspect Sensor Orientation

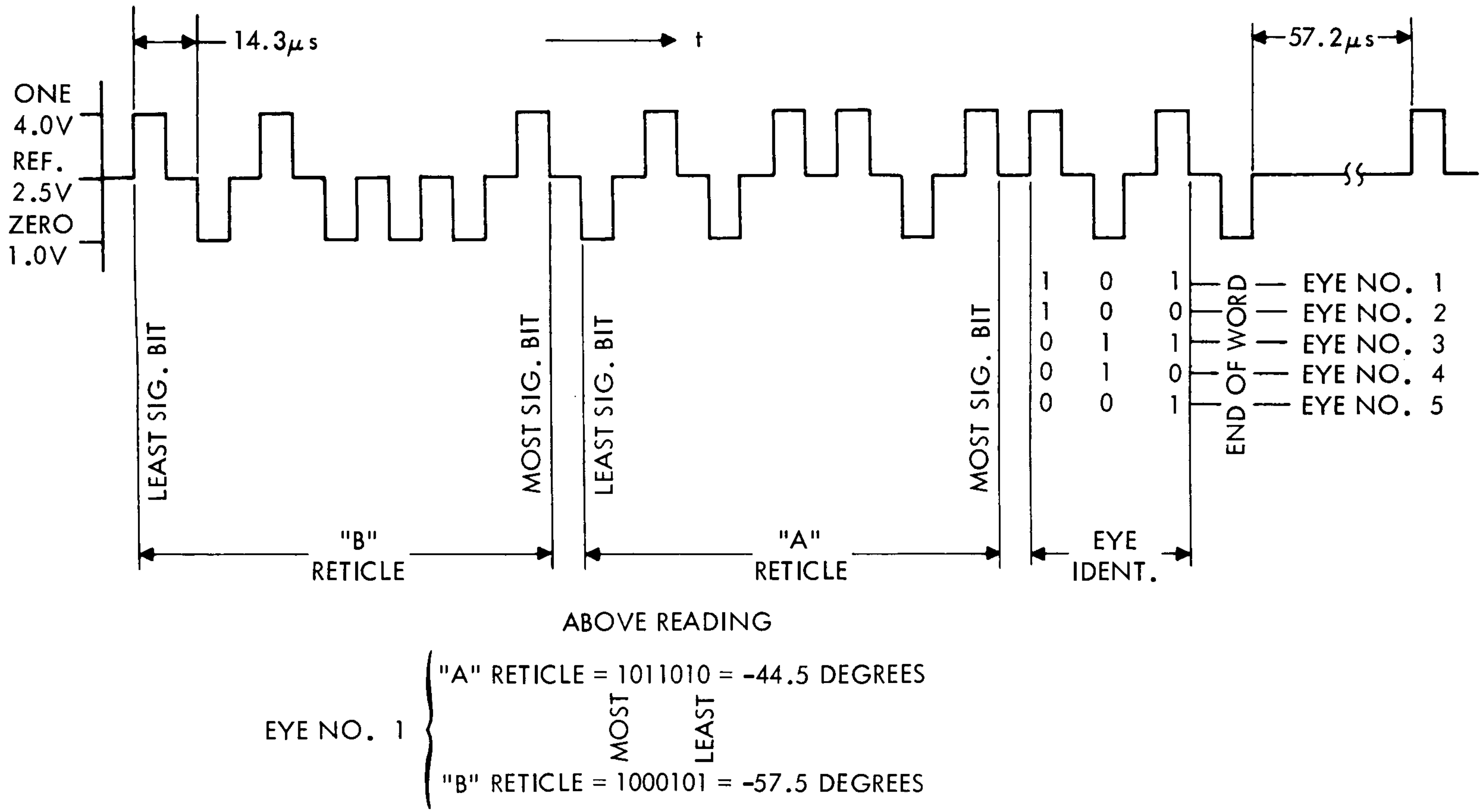


Figure 17. Solar Aspect Data Train

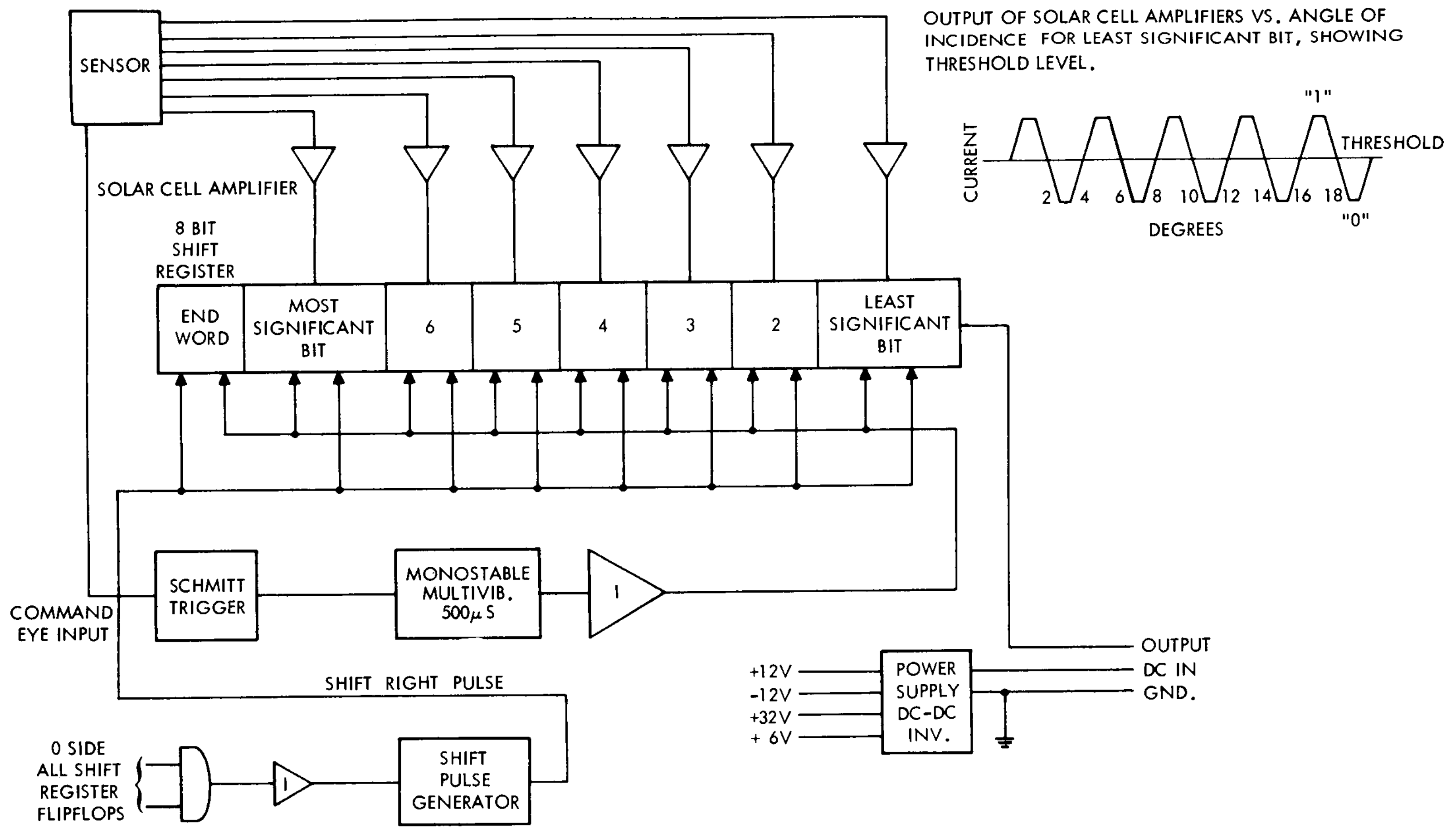


Figure 18. Solar Aspect System Block Diagram

(altitude-versus-time) radar plot, and a magnetic sounding plot (field-strength-versus-altitude) for the launch site at Wallops Island, the attitude and dynamic motion of the vehicle are determined.

Flight 17.01 utilized a magnetometer system which was comprised of two independent magnetic aspect sensors, one each for the rocket's transverse and longitudinal axes, and each having a ±600 millioersted field of range, and producing corresponding outputs between zero and +5 volts dc.

Flight 17.02 utilized a magnetometer system comprised of three independent sensors, two for the rocket transverse axis (pitch and yaw), and the third for the rocket longitudinal axis (roll). These three sensors had ±600 millioersted field ranges and the same corresponding output voltages of zero to +5 volts dc. With the use of proper load resistors, the output from each magnetic aspect sensor is applied to a commutator of the telemetry system. For magnetometer orientation and block diagram on Flight 17.02, see Figure 19.

The single component magnetometer consists of a field sensor and an electronic unit. The field sensor is comprised of a permalloy core surrounded by two windings, and the electronic unit is composed of a regulated oscillator, a nominal +2.5 volt-dc bias source, and a phase sensitive rectifier. The regulated oscillator generates an excitation current, at a frequency of 9000 hertz, which is passed through one of the windings of the field sensor. Associated with this current is a magnetic field of sufficient magnitude to cyclically drive the core into saturation. If there is a component of an external field threading the core, parallel with its longitudinal axis, a voltage which is the second harmonic of the excitation frequency is induced into the other winding. The second harmonic signals are converted into a d-c voltage signal by the phase sensitive rectifier, whose output is biased by the nominal +2.5 volt-dc output of the bias source. Thus, the output signal of the magnetic aspect sensor, properly loaded, swings about the bias level in the range of zero to +5 volts dc.

ACCELERATION AND VIBRATION SYSTEMS

Acceleration and vibration instrumentation for the Aerobee 350 consisted of two types of accelerometers. Acceleration measurements were made by a strain gauge type transducer (Figure 20), and vibration measurements were made by a piezoelectric type transducer.

Low range sensitivity strain gauge accelerometers were used along the yaw (X) and pitch (Y) axes of the rocket, and both low and high range sensitivity strain gauge accelerometers were used along the thrust (Z) axis. The thrust axis accelerometers were used to measure the high range thrust forces from time of

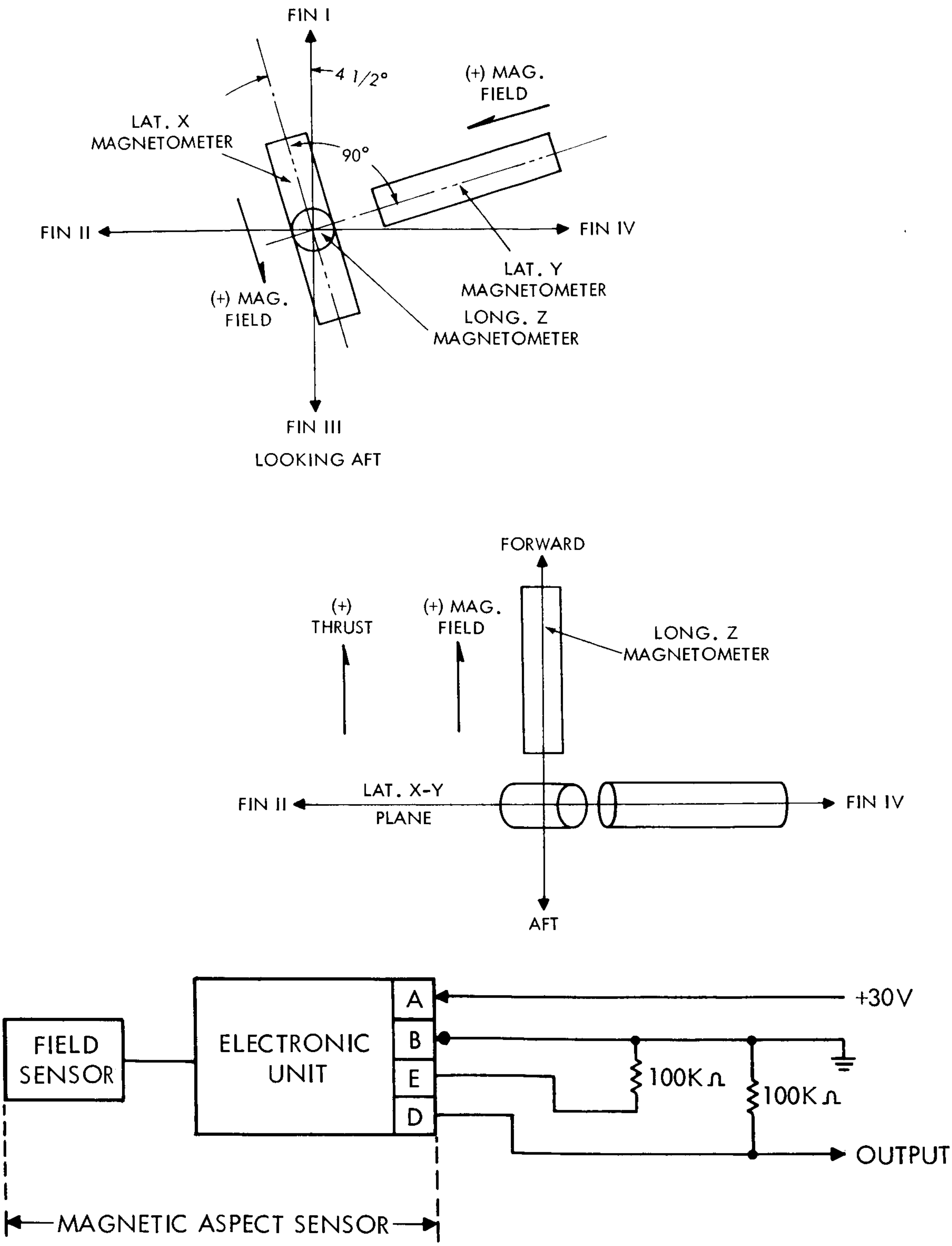


Figure 19. Magnetometer Orientation and Block Diagram

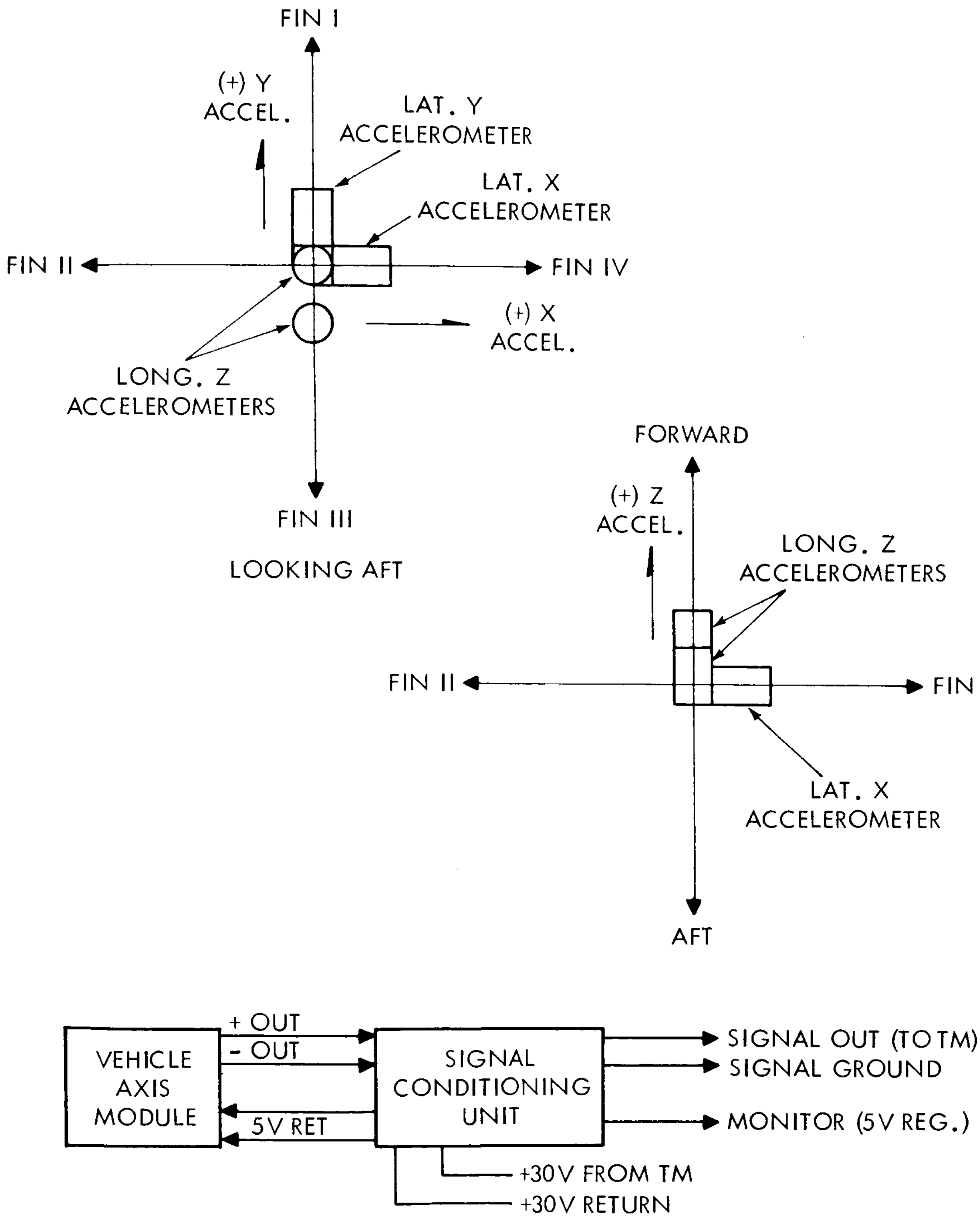


Figure 20. Accelerometer Orientation and Block Diagram

engine ignition at launch to sustainer burnout, and the low range forces during the remainder of flight.

The strain gauge accelerometers used on Aerobee 350 were manufactured by Consolidated Electronics Corporation (CEC), and consisted of types 4-204-0001 and 4-202-0001. The two types differ in that the former is a triaxis unit and the latter a single axis unit. Both units are unbonded, bidirectional accelerometers.

Within the strain gauge accelerometer modules, strain gauge elements are arranged in a Wheatstone bridge configuration which permits a significant amount of output voltage when an excitation voltage is applied to the bridge. The elements are arranged in such a way that when a force is applied to the element by the seismic mass, the bridge becomes unbalanced. The amount of unbalance provides a voltage output in the order of 50 millivolts. This signal output is connected to a signal conditioning unit [Baldwin Lima Hamilton, Electronics Division, Model 950 (BLH)], which converts the signal to the proper level of zero to +5 volts dc. This is a compatible signal level for telemetry use.

Vibration instrumentation measurements, as stated, were made by piezoelectric type accelerometers. These accelerometers also were used in the yaw (X), pitch (Y), and thrust (Z) axes of the rocket.

The piezoelectric type accelerometers used on Aerobee 350 were manufactured by the Endevco Corporation, and consisted of types 2232 and 2221-D.

In the piezoelectric accelerometer, a crystal displacement is proportional to the applied acceleration at frequencies up to one fifth the accelerometer resonance frequency. Therefore the upper frequency limit is determined by the design of the accelerometer. Piezoelectric accelerometers are designed to have their resonant frequency in the range of 1 to 100 kilohertz. As an example, the sensitivity constant (output voltage divided by applied acceleration) of an accelerometer with a 12 kilohertz resonant frequency is 300 millivolts per g.

Piezoelectric materials have a very low internal damping, therefore the response for an ideal accelerometer is constant throughout its frequency range, with a rise in sensitivity of nearly five percent at one fifth its resonance.

Another important characteristic of piezoelectric accelerometers is that the effect of temperature is minimal on their performance. Some designs maintain satisfactory characteristics within temperature ranges of minus 450 degrees to plus 750 degrees Fahrenheit. The resistivity of piezoelectric materials decreases as the temperature increases.

TEMPERATURE MEASURING SYSTEMS

Aerobee 350 payloads utilize two different temperature measuring systems. One of the temperature measuring systems consists of a resistance thermometer, a commutator, and a temperature amplifier. This system is used to measure environmental temperatures on the payload and vehicle. A second system, composed of thermocouples, zone boxes, and low level d-c signal conditioner, is mounted in the sustainer to measure propellant area temperatures.

The Flight 17.01 temperature transducer system, as an example, is depicted in block diagram form in Figure 21. This temperature transducer system consisted of a platinum sensing element, electrically insulated from the stainless steel cover plate, with the output being the resistance across the sensing element. Output resistance varies linearly with the temperature in the operating range of zero to +1000 degrees Fahrenheit (-17.8 to +538 degrees C). The 45-ohm nominal resistance of the transducer at zero degrees Fahrenheit increases to a nominal 145 ohms at +1000 degrees F (538 degrees C). The transducer has a thermal time constant nominally less than one second, and an accuracy within ±1 percent of the span of its operating range.

With one terminal of the Aerobee 350 transducer grounded, the other terminal is connected to an input of a mechanical commutator. The resistive inputs from the temperature transducers, as well as inputs from grounded calibration resistors, are applied by a commutator in a recurrent sequence to the inputs of a temperature amplifier. This amplifier converts the applied resistance values to d-c voltage levels in the zero to 5 volt-dc input range of the telemetry systems which transmit the temperature data.

The gas-temperature sensing transducer of the Aerobee 350 is mounted in the thrust chamber engine compartment to measure the temperature of the gases flowing about that compartment. The unit consists basically of a platinum wire sensing-element, wound on four posts, all contained within a hollow cylindrical probe, which is open at one end to provide a gas inlet, and vented to produce a free flow of the gas over the sensing element. The transducer output, being the resistance of the sensing element, varies directly with the gas temperature. The transducer that has been used in these flights provided a 50-ohm resistance to a nominal 110-ohm output for temperatures of +32 to +600 degrees Fahrenheit (0 to 316 degrees C) respectively.

For the Aerobee 350, it is possible to provide signal conditioning and calibration for all temperature transducers by commutating the resistive outputs of the temperature transducers and the calibration resistors.

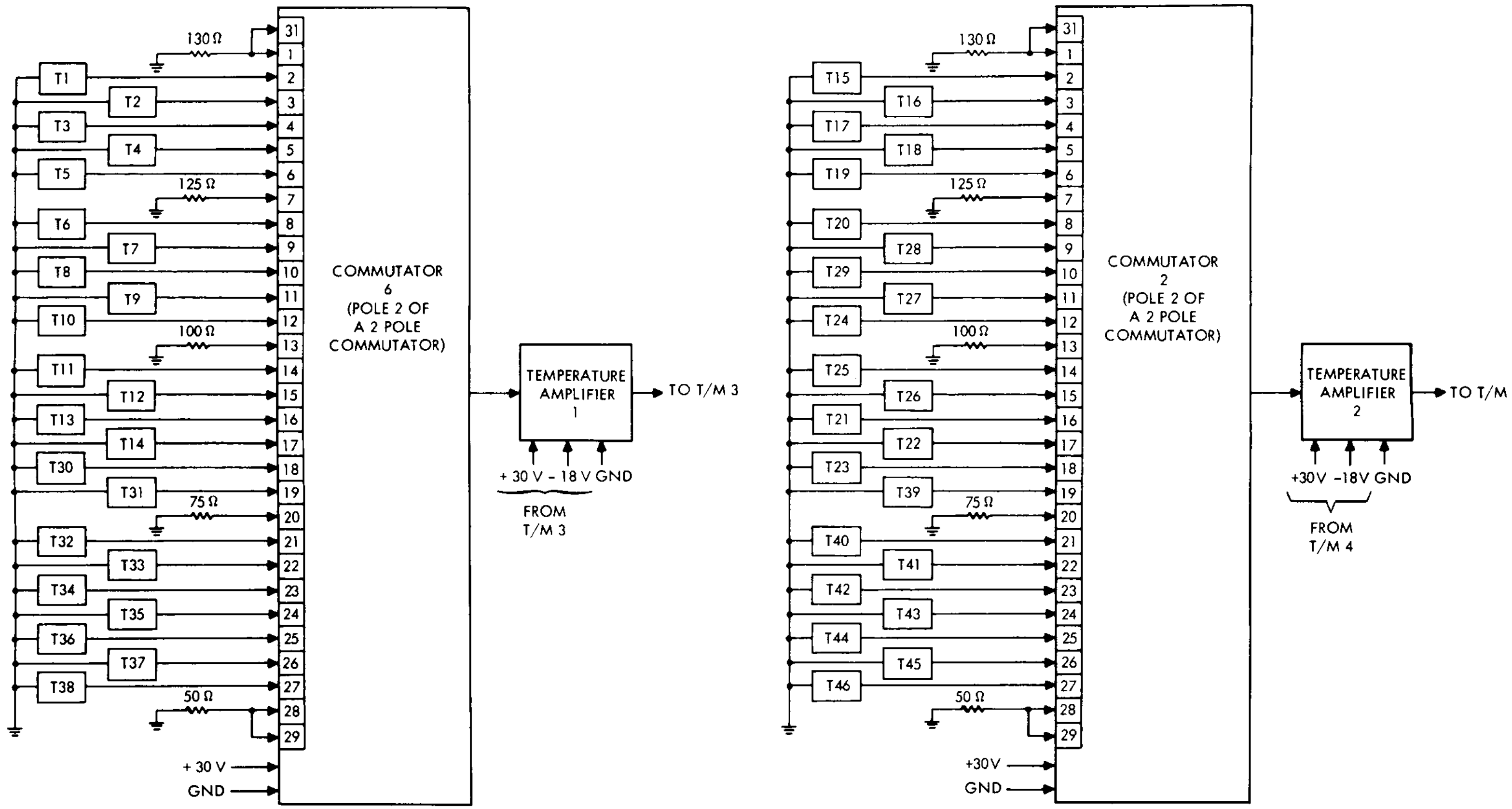


Figure 21. Temperature Transducer System, Block Diagram (as used for Flight 17.01)

Calibration resistors applied to the commutators are metal film type resistors, with selected values of 50, 75, 100, 125, and 130 ohms. The calibration resistors (50-ohm) are used for the frame sync pulse at the input of the commutators. This corresponds to a +5 volt-dc level at the output of the temperature amplifiers. The 130-ohm resistors are connected to the pedestals of each commutator, and result in a zero volt-dc output level from the amplifiers. (See Figures 22 and 23.)

Thermocouple systems are used to measure the temperatures of the coolant and helium tank regulator inlets and outlets. The rocket fuel, which is used as the coolant, circulates through the jackets around the thrust chambers before being injected into the chambers, thus cooling the thrust chambers, and at the same time heating the fuel to increase its combustibility. The coolant inlet temperature therefore, refers to the initial fuel temperature, and the coolant outlet temperature to the final fuel temperature before combustion.

Each of the thermocouple systems are comprised of a thermocouple probe, a zone box and a low level d-c signal conditioner. See Figure 24 for a block diagram of the thermocouple systems. Each thermocouple probe produces a potential difference across its output terminals which corresponds to the temperature level of the helium or coolant being measured. The zone box, in conjunction with a regulated dc power supply, and additional circuitry contained in the amplifier unit, provides a temperature-compensated reference voltage for the thermocouple system, by supplying the appropriate bias level to the millivolt output generated by the thermocouple probe. The gain and the balance of each amplifier is individually so adjusted that the required temperature measuring range of its associated thermocouple probe produces a zero to +5 volt-dc output, compatible with the full scale input range of the telemetry systems.

An imersion-type thermocouple is used in the Aerobee 350. The imersion-type thermocouple probe consists basically of two thermoelement wires, one of them being copper and the other a constantan, insulated in an enclosed stainless steel cylindrical sheath. A grounded junction element configuration is utilized, with both thermoelement wires terminated to the base end of the grounded sheath to form the copper versus constantan thermoelement junction. The sheath, the part of the probe that is immersed in the substance whose temperature is being measured, extends from a stainless steel tubular housing, in which the thermoelement wires are attached to flexible insulated output lead wires, surrounded by additional insulated tubing.

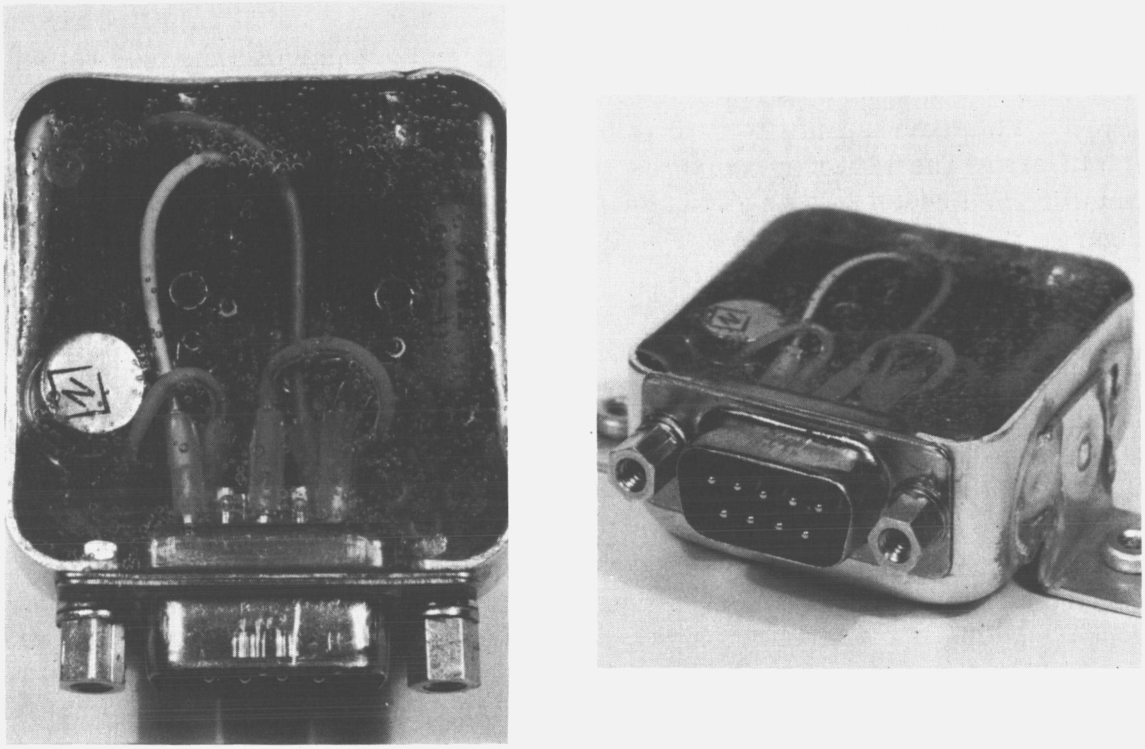


Figure 22. Temperature Amplifier

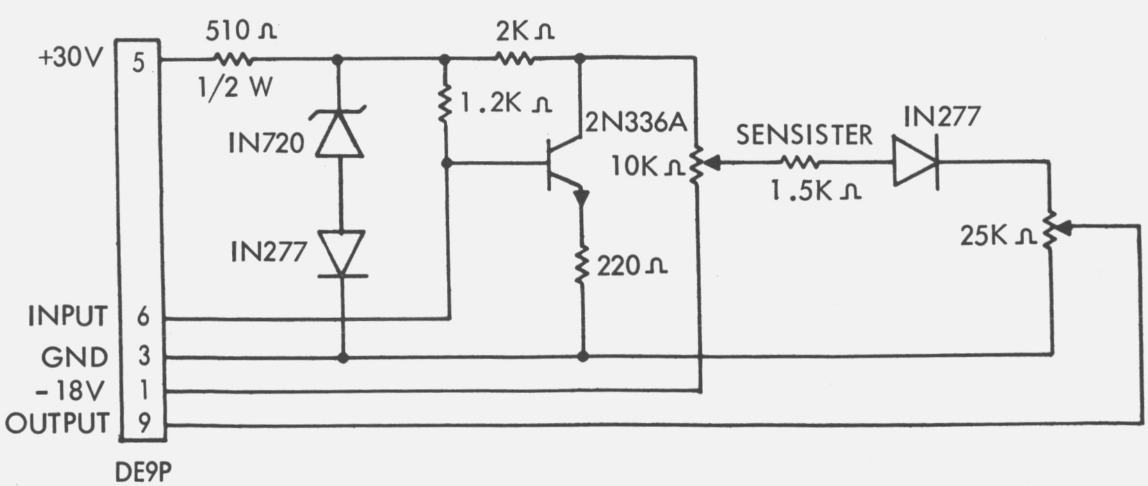
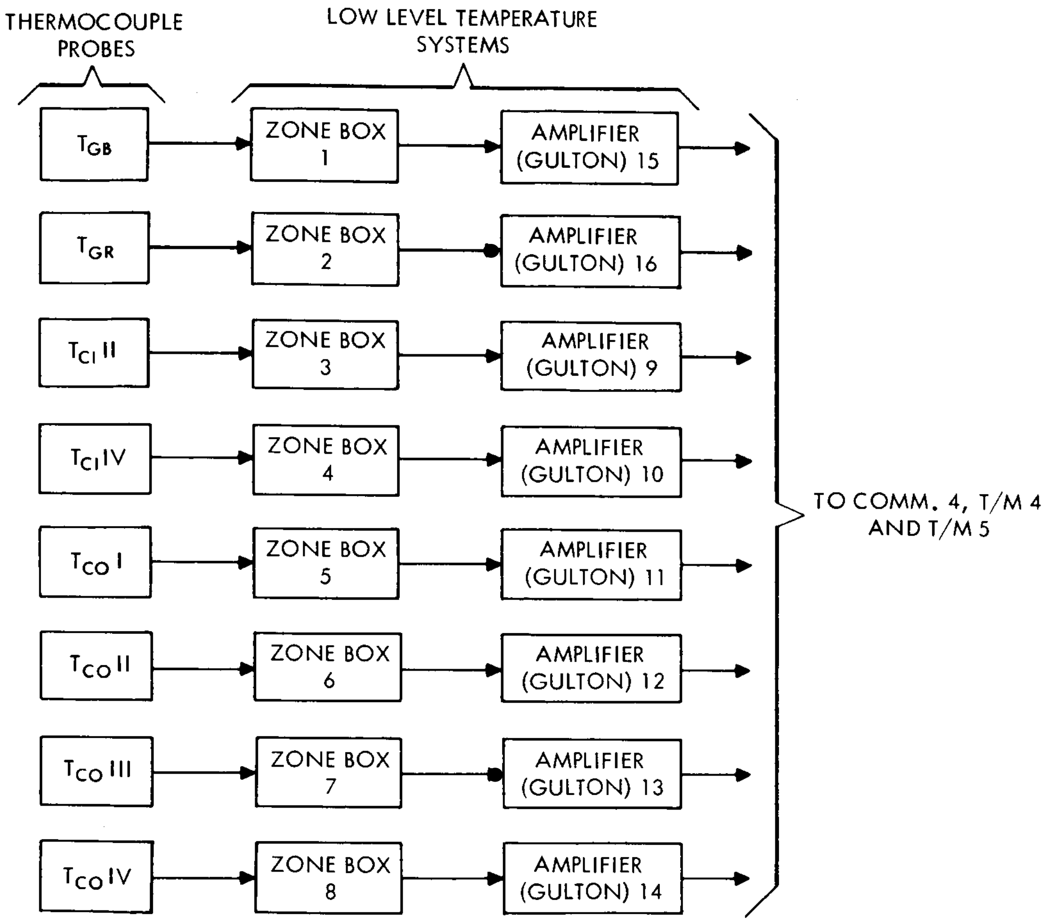


Figure 23. Temperature Amplifier Schematic Diagram



TYPICAL CONNECTIONS FOR THERMOCOUPLE SYSTEMS

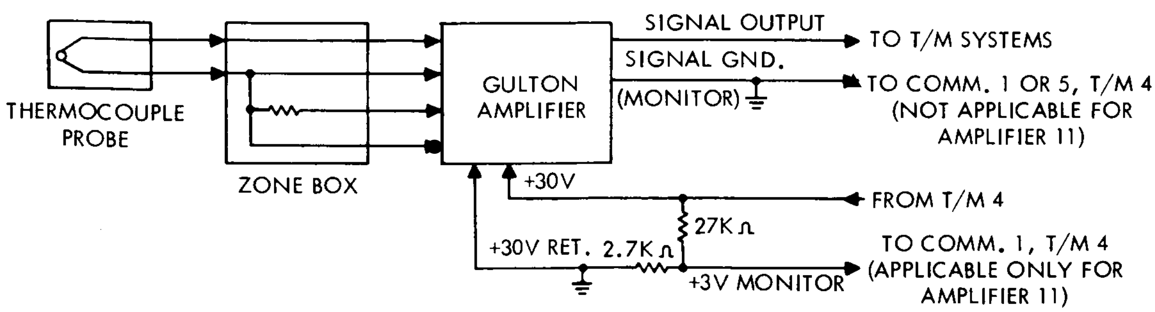


Figure 24. Thermocouple Systems, Block Diagram (as used for Flight 17.01)

VSWR MONITOR SYSTEM

The Voltage Standing Wave Ratio (VSWR) Monitor System was developed by the Sounding Rocket Instrumentation Section because of the need for a system to monitor the proper operation and the load matching characteristics of antenna systems with their associated telemetry transmitters, during prelaunch testing and inflight operation. The method decided upon was to sense the average incident and reflected power, by means of a bidirectional antenna coupler, amplify and compute this data by means of a signal processing unit, and transmit the output through one of the other telemetry systems. The SWR or VSWR between the transmitter and antenna system can then be computed from the transmitted data by using the following equation:

$$\mathrm{SWR} = \frac{1+\dfrac{(P_r)^{1/2}}{(P_i)}}{1-\dfrac{(P_r)^{1/2}}{(P_i)}} = \mathrm{VSWR}$$

where P_i and P_r are the incident and reflective powers, respectively.

A bidirectional antenna coupler (Figure 25) detects the average incident and reflected powers subjected to its input, and produces two outputs which are linearly proportional to their respective inputs. The incident power input is directly connected to the reflected power input by way of a feedthrough line. These inputs have a 50-ohm impedance which gives the antenna coupler the appearance of an additional short piece of 50-ohm cable, inserted on the connection between the 50-ohm transmitter output and the 50-ohm antenna system input. Thus, no determinable loss of transmitted power occurs due to the use of the antenna couplers. For VSWR calibration curve, see Figure 26.

The Signal Processing Unit provides the required gain to produce nominal zero to +5 volt-dc outputs for the respective inputs of zero to 40 millivolts.

PRESSURE TRANSDUCER SYSTEMS

Pressure transducers are used to measure the absolute pressures of fuel and oxidizer liquids, helium gas, and other gases including air, on the various sections of the rockets. Associated with each pressure transducer is a signal conditioning unit which provides a +5 volt-dc regulated output for the excitation

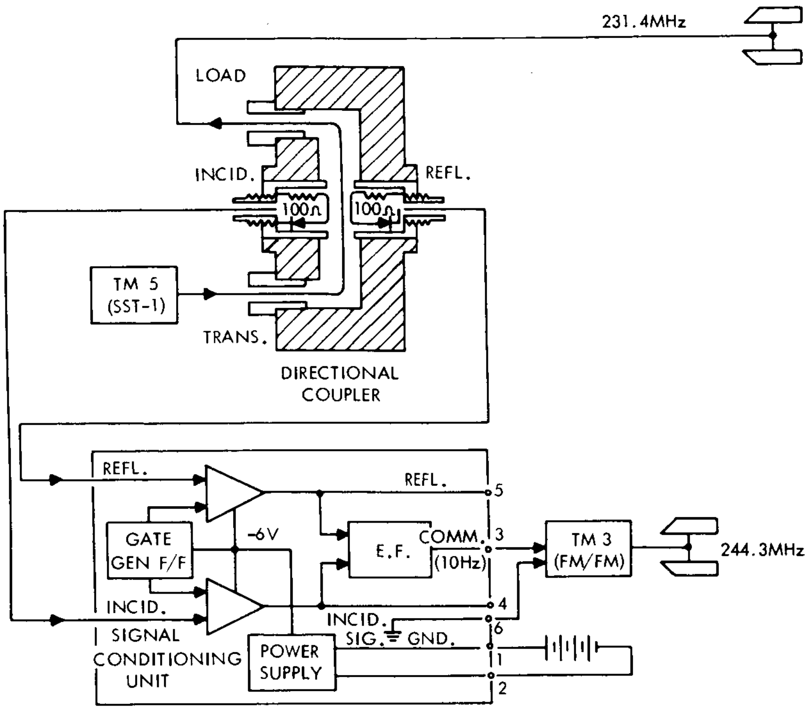


Figure 25. Antenna Coupler

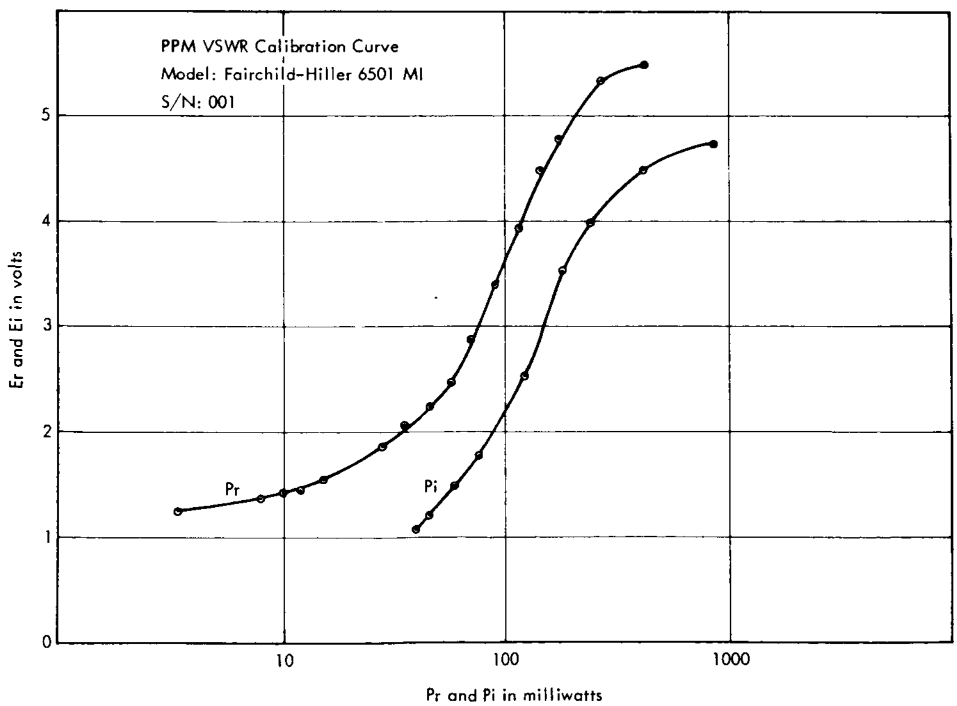


Figure 26. PPM VSWR Calibration Curve

voltage required by the pressure transducer. The unit also amplifies the output of the pressure transducer to a d-c level compatible with the input requirements of the telemetry system which transmits the pressure data. (See Figure 27 for a block diagram of a pressure instrumentation system.)

For the Aerobee 350, pressure transducers were selected from Consolidated Electrodynamics Corporation (CEC), from the several available ranges of the type 4-326-0001 standard model pressure transducer, and a low pressure transducer was selected from the type 4-326-0003 model.

The factory selected ranges of the pressure transducers correspond with the required pressure ranges, with the exception of the 15 and 1500 psia. In order to operate the selected transducers in the required pressure range, the gains of the associated signal conditioner units are so adjusted that the maximum required pressure of each transducer produces an output from its SCU of +5 volts dc, (the maximum calibrated input level to the telemetry system). A few of the pressure transducers require that a precision metal film resistor be placed across their outputs in order to reduce their sensitivity, where the minimum gain of the associated SCU is too high to produce a +5 volt-dc output for the maximum required transducer pressure. The signal conditioning unit is the same as was used and described for the accelerometer.

In the case of the 15 psia pressure transducers, the gains of their associated SCU's are so adjusted that the maximum required pressure produces a +4 volt-dc output, instead of a +5 volt-dc output, as in the case of the other transducers. It is desired that abnormal pressures, greater than 15 psia in the areas measured by these transducers, can be clearly indicated by the data transmitted.

The functional operation for the three types of strain gauge pressure transducers utilized is the same. Pressure applied to the transducer causes displacement of a force-summing diaphram. Connected to the center of the diaphram is a force rod which transmits the force to a sensing element, composed of a spring structure which is supported from posts about which the strain gauge wire is wound. Movement of the center of the spring causes movement of the posts, thus increasing the strain and resistance of two of the windings at one end of the posts while decreasing the strain and resistance of the two windings at the other end. With a d-c excitation voltage applied across two opposite junctions of the Wheatstone bridge design of the four active arms, a d-c voltage output across the remaining two opposite junctions is produced, which is precisely proportional to the applied pressure.

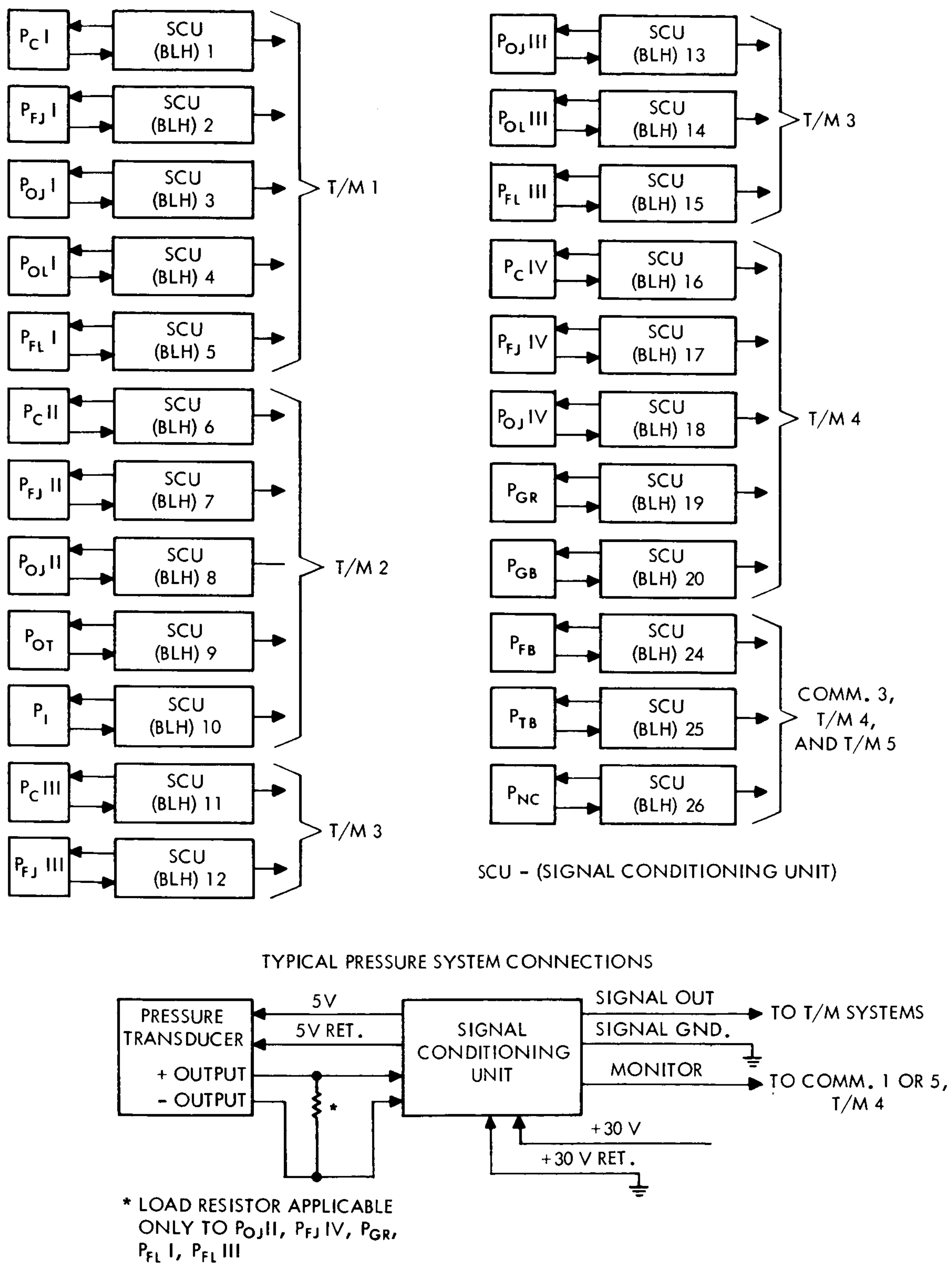


Figure 27. Pressure System Instrumentation, Block Diagram

TELEMETRY SYSTEMS

Telemetry systems used for transmission of data from the Aerobee 350 rocket launches were of two types. The basic system was a Frequency Modulation (FM/FM) type of data transmission, which has been used on all three rockets launched to date, and the other was a Pulse Position Modulation (PPM) data transmission system, which was used on the second rocket launch, the 17.01, as a redundant system in order to prove its capability. For the third rocket launched, the 17.02, the PPM system was then used as a part of the basic system, along with FM/FM.

The FM/FM telemetry systems for the three rockets were similar in the type of data being transmitted. The major differences were in the number of systems, and the telemetry channels and their commutator assignments, required for each.

The first launched vehicle, the 12.02, which was used to prove the compatibility of the rocket and rocket launcher, required a total of two FM/FM telemetry systems and their commutator systems. (For all channel and commutator allocations, see Appendix A.)

The second launch, the 17.01, required a more sophisticated system of telemetry, since it was necessary to transmit data back from a full flight performance payload. This made it necessary to utilize a total of four FM/FM and one PPM telemetry systems, with six commutator systems. Since, as stated above, this was the first time a PPM telemetry system had been used in conjunction with the rocket launches, it was used primarily as a redundant system, to transmit back selected data which was also transmitted by the FM/FM systems in order to make a comparison check of the quality of the data, and to thus prove the feasibility of using a PPM type of telemetry system.

Telemetry of Flight 17.02 was basically the same as that used for the previous rocket, 17.01. The main difference was that, in this latest launch, there was a total of three FM/FM systems, and one PPM type system used to transmit the rocket performance and scientific data. The PPM system on this flight was used to transmit data without the previous redundancy, since its capability had been satisfactorily proven by the previous flight.

To explain the detailed operation of the Aerobee 350 telemetry instrumentation, the following discussion covers the systems used for the second rocket (17.01). The major differences between this and the other two rockets was in the particular selection and allocations for the data transmissions desired, as required by the type of mission flown.

Aerobee 350 Flight 17.01 carried four FM/FM telemetry systems and one PPM telemetry system for data transmission. FM/FM telemetry systems Numbers 1 and 2 each contained twelve subcarrier frequency bands; FM/FM telemetry systems Numbers 3 and 4 each consisted of fifteen subcarrier frequency bands; and the PPM telemetry system (Number 5) contained sixteen channels.

In Table II is a list of the five telemetry systems, and information on their individual mode of transmission, transmission frequency, transmitter frequency deviation, power output, and payload deck location.

Table II
FLIGHT 17.01 TELEMETRY SYSTEM INFORMATION

Telemeter Number	Mode of Transmission	Transmission Frequency (MHz)	Transmitter Frequency Deviation (kHz)	Power Output (rated watts)	Payload Location (Deck Number)
1	FM/FM	259.7	±125	2.4	2
2	FM/FM	256.2	±125	2.0	3
3	FM/FM	244.3	±125	3.0	4
4	FM/FM	240.2	±125	3.8	5
5	PPM	231.4	N/A	27.0 (peak)	10

POWER MONITORING AND CONTROL

The power-control relay switches the power-circuitry of the telemetry and instrumentation systems from the external power supply, used on the ground, to the internal battery-supply used when the rocket is in flight. The internal battery supply consists of 20 re-chargeable Yardney Silver-Cel batteries, rated at 3 ampere-hours, which may be re-charged from an external source before flight. The power relay is magnetically latched in either position by an external power source. The Inertia Switch ("G" switch) which actuates from the inertial force created by the rocket lift-off, locks the power circuitry on internal-power as a safety precaution, should the power relay malfunction as a result of the launch environment. Circuitry for monitoring of the inertia switch and the battery supply is provided. Figure 28 shows a typical power-monitoring and control system, as used in Flight 17.01.

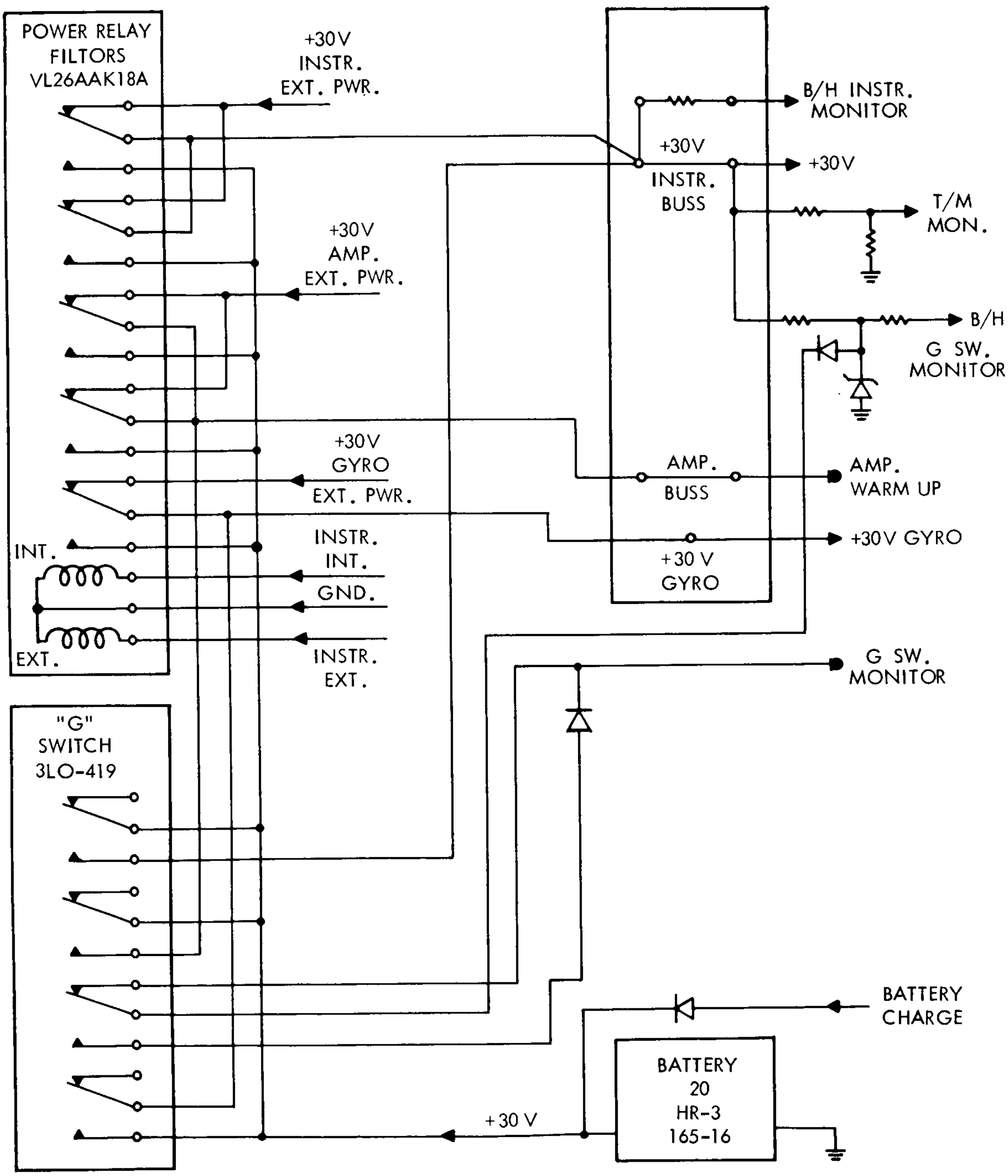


Figure 28. Telemetry Power Control and Monitoring Diagram

TELEMETRY SYSTEM 1

For the block diagram of Telemetry System 1, refer to Figure 29. The twelve data inputs allocated to this FM/FM telemetry system fed the sequential in-flight voltage calibrator, which consisted of a master calibrator (Tempo 90972) and two slave units (Tempo 90972-1). The master calibrator contains a precision-voltage source, timer-channel stepping relay, and four data relays for the data inputs of channels E, C, A and 13. Slave units A and B each contain four data relays for the remaining eight data input channels. The master calibrator activates each of the twelve data relays separately, for a full eight step calibration cycle. When a data channel is activated, the associated input to the voltage clipper or VCO (as applicable) is removed from the data input line and connected to the calibration source. A series of calibration steps consisting of zero, 1, 2, 3, 4, 5 volts dc and an open circuit were programmed into the appropriate input of the voltage clipper, or VCO (as applicable), in a staircase fashion. These voltage steps were accurate to ±0.1 percent under any environmental conditions. The total calibration time per channel is adjusted to the maximum of 1.6 seconds, to provide sufficient duration of the transmitted calibration voltage data to produce accurate final calibration. The period of continuous data between calibration cycles is adjusted to 102 seconds to provide calibration measurements just prior to rocket lift-off. This provides continuous data transmission until second stage burnout is completed (approximately T+52 seconds), and also provides calibration measurements shortly after second stage burnout.

The voltage clipper limits the levels of the ten applied data-channel inputs to a nominal range of -0.8 to +5.3 volts dc. In case of a malfunction in a transducer, any extreme voltages that might be produced would be suppressed to these limits. This prevents overloading of the voltage-controlled oscillators (VCO's).

The voltage-controlled oscillators (Vector TS56A), responsive to the data inputs, generates subcarriers for each of the data channels. The VCO's produce a frequency-modulated sine wave, modulated linearly in direct proportion to the zero to +5 volt-dc input voltage.

The voltage regulator (Vector TV53-6) supplies a regulated +6 volt-dc potential from the +30 volt-dc battery supply to the VCO's. The change in output voltage due to load current, or line voltage changes, is less than +0.1 percent.

The mixer amplifier (Vector TA58A) is used to amplify the mixed VCO signals, and as a low impedance driver for the transmitter. The high-gain audio amplifier utilizes negative feedback to minimize distortion and gain variations.

The telemetry transmitter (Vector TR PT-2V-1) used is a crystal-stabilized, phase-modulated, transistorized unit, with a 259.7 megahertz center frequency.

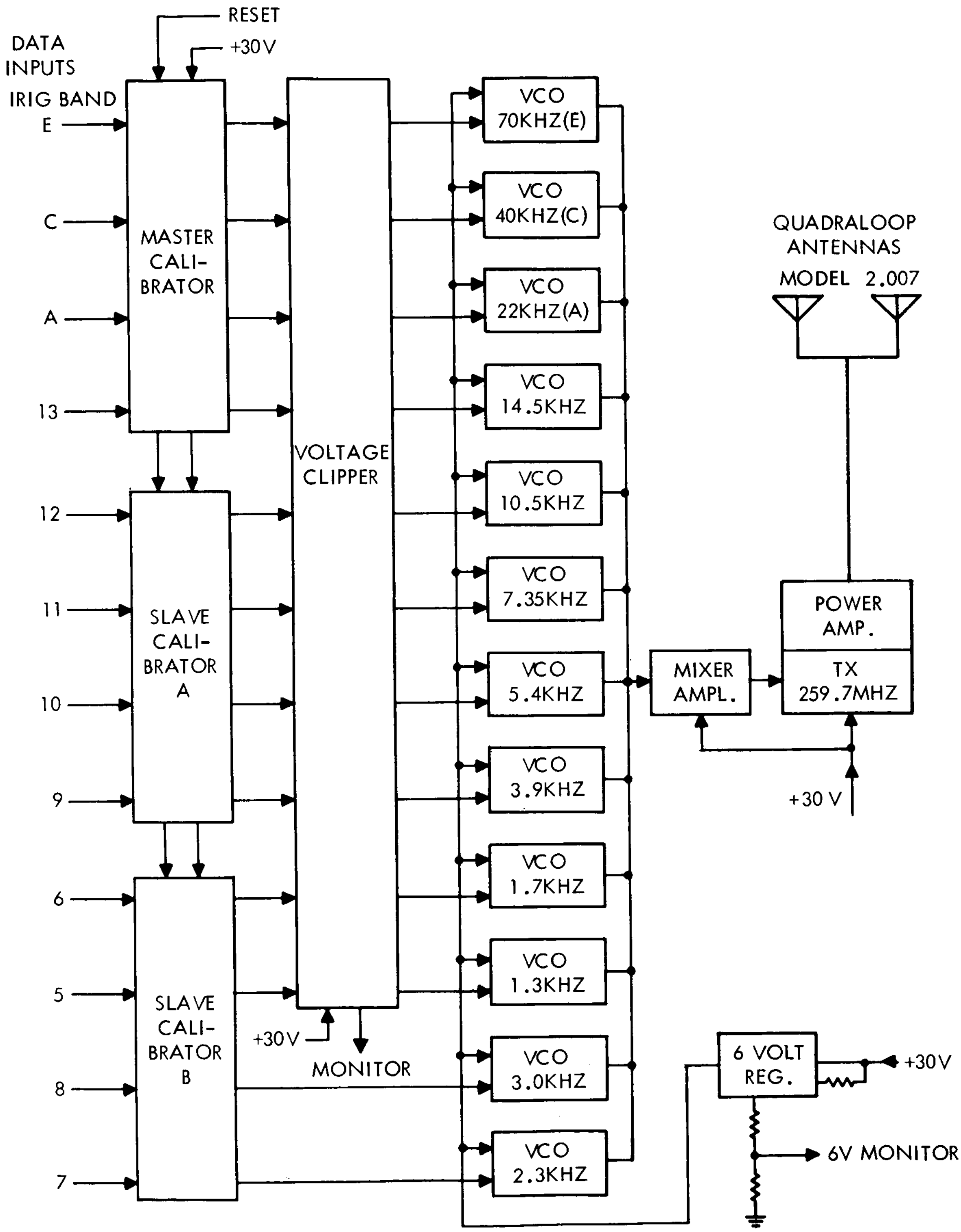


Figure 29. Telemetry Systems 1 and 2, Block Diagram

An integral part of the transmitter unit is a transistorized power amplifier which boosts the transmitter output to a minimum of 2 watts. The transmitter drives a pair of quadraloop antennas. (See Antenna section.)

TELEMETRY SYSTEM 2

The operation of Telemetry System 2 is similar to that of Telemetry System 1. The block diagram of Telemetry System 1 (Figure 29) applies to Telemetry System 2, noting that the transmission frequency of Telemetry System 2 is 256.2 megahertz.

TELEMETRY SYSTEM 3

The operation of Telemetry System 3 is also similar to that of Telemetry System 1. The block diagram of Telemetry System 3 is shown in Figure 30. Note that the transmission frequency is 244.3 megahertz.

In conjunction with Temperature Amplifier 1, and the ±18-volt regulator (Vector TV56A), Commutator 6 is utilized to transmit 23 data, and 5 calibration inputs, on one of the 15 channels. The 23 data inputs to the commutator are resistances of temperature transducers and calibration resistors.

The ±18-volt regulator provides a regulated negative-bias supply to the temperature amplifier. The recurrent sequence of dc voltages, in the zero to +5 volt-dc range from the temperature amplifier, is applied through the voltage clipper to the VCO.

The commutator used (Commutator 6) was a Datametrics (type 952-3), 2.5 x 30 IRIG. The switch format is for automatic decommutation. It has a 50 percent duty cycle, and is a two pole mechanical unit. Inputs to commutator 6 are made to 29 stationary contacts. A wiper arm, rotated at 2.5 revolutions per second, samples these contacts, and the pedestals between each, in a recurrent sequence, with a duty cycle of 50 percent. A frame synchronization pulse, for synchronizing with the ground station, is provided by having two adjacent stationary contacts, and the pedestal between them, internally connected, and an input consisting of a 50-ohm resistor-to-ground applied to them. This 50-ohm resistance is transformed by the temperature amplifier into a +5 volt-dc pulse. All of the pedestals, except the one used in the frame synchronization pulse, are connected to a 130-ohm resistor-to-ground, corresponding to a zero volt-dc level at the output of the temperature amplifier.

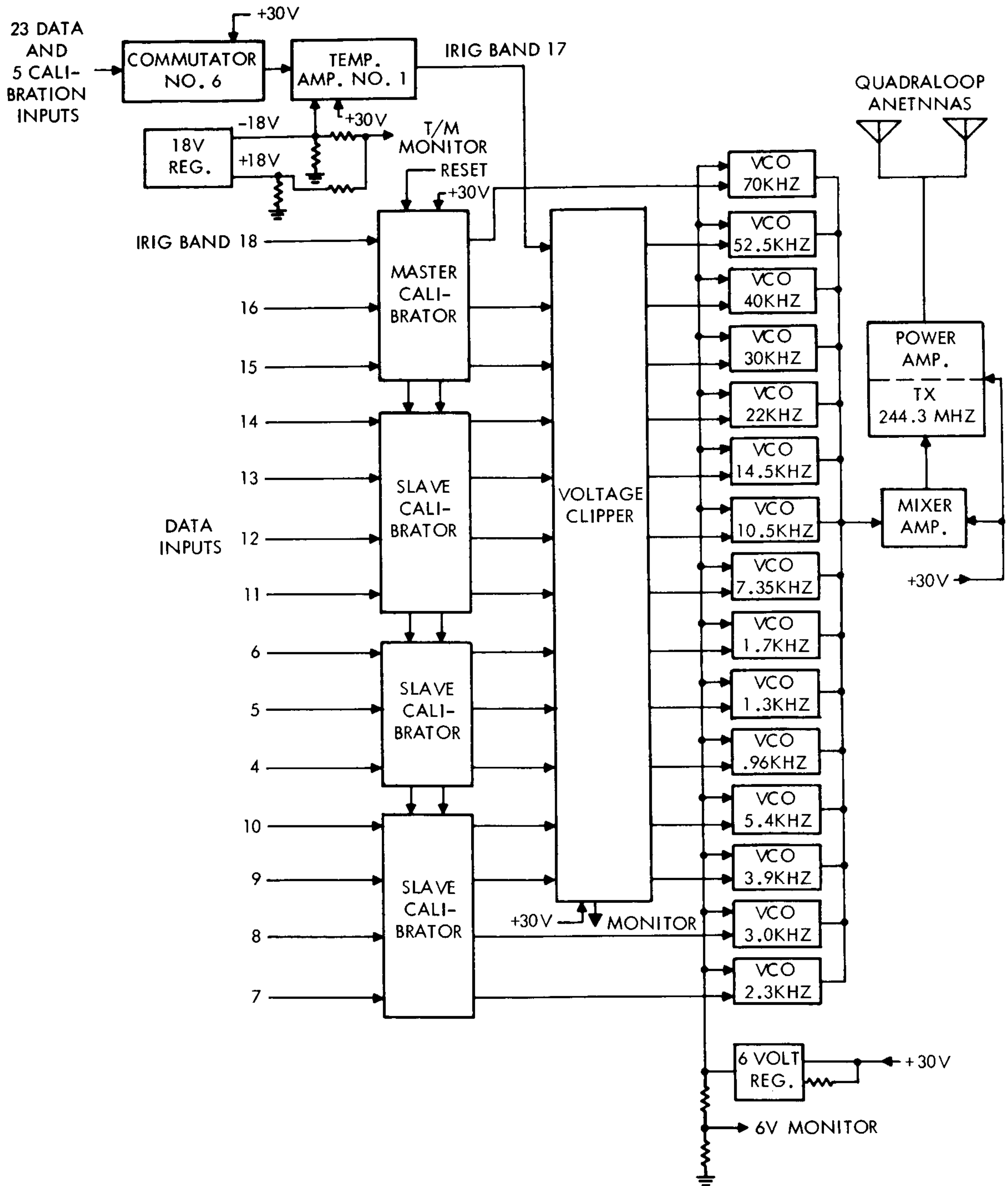


Figure 30. Telemetry System 3, Block Diagram

TELEMETRY SYSTEM 4

The operation of Telemetry System 4 is similar to that of Telemetry Systems 1 and 2. The block diagram for Telemetry System 4 is shown in Figure 31. However, note that the transmission frequency is 240.2 megahertz.

TELEMETRY SYSTEM 5

Telemetry System 5 consisted basically of the Model SST-1 Telemetry System, developed by the Sounding Rocket Instrumentation Section, as a small, lightweight, low power consumption, replacement of the An/DKT-7 telemetry transmitter. The Model SST-1 is a 16-channel, pulse-position/amplitude modulated (PPM/AM) system. Refer to Figure 32 for a block diagram of Telemetry System 5, with its associated power control and monitoring system.

The in-flight calibrator (See Figures 33 and 34) is similar to that described in Telemetry System 1.

The premodulator accepts and converts the data outputs from the in-flight calibrator to pulse position modulated (PPM) format. Overloads to +10 volts dc or to -3 volts dc can be sustained without damage to the system, or adverse effect on other data channels. The time sharing multiplex format for each channel consists of a 25-microsecond guard band, a 150-microsecond data transmission width, and another 25-microsecond guard band. Within the first guard band of the first channel, a triple pulse is generated to provide synchronization of the transmitted data with the ground station. The premodulator provides a sampling, with provisions for doubling or quadrupling the rate at the expense of the number of channels of data. The position, in the PPM format, of the data pulse generated for each channel, is linearly proportional to the input voltage of the data input.

The PPM format output of the premodulator is applied to a four-stage (cathode modulated) pulse transmitter, designed to operate at 231.4 megahertz. The output of the transmitter was a series of constant-amplitude radio-frequency pulses, duplicating the pulse width and spacing of the premodulator output. The unit was designed to produce approximately 30 watts-peak-power into an antenna of 50-ohms impedance, with a maximum VSWR of 2:1.

The nominal power requirement for the system is 28 volts dc, at 0.9 ampere. Since the output of battery supply may range from 26 to 35 volts dc, the power supply is designed to provide regulated voltages to the premodulator and the transmitter for any input within that range. A remotely controlled latching relay within the power supply permits operation, either on internal battery supply or on the external supply, as well as on-off control up to the moment of flight.

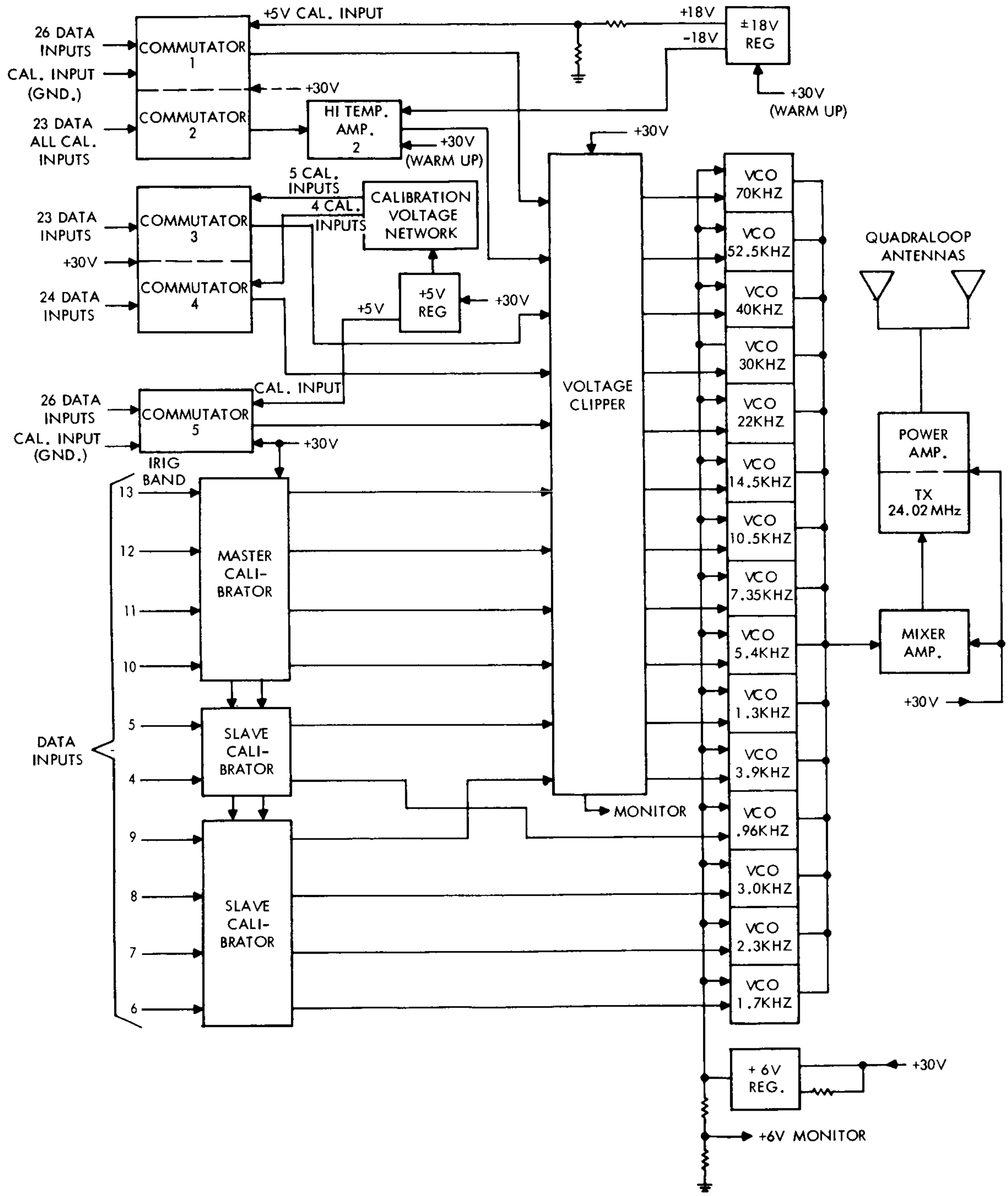


Figure 31. Telemetry System 4, Block Diagram

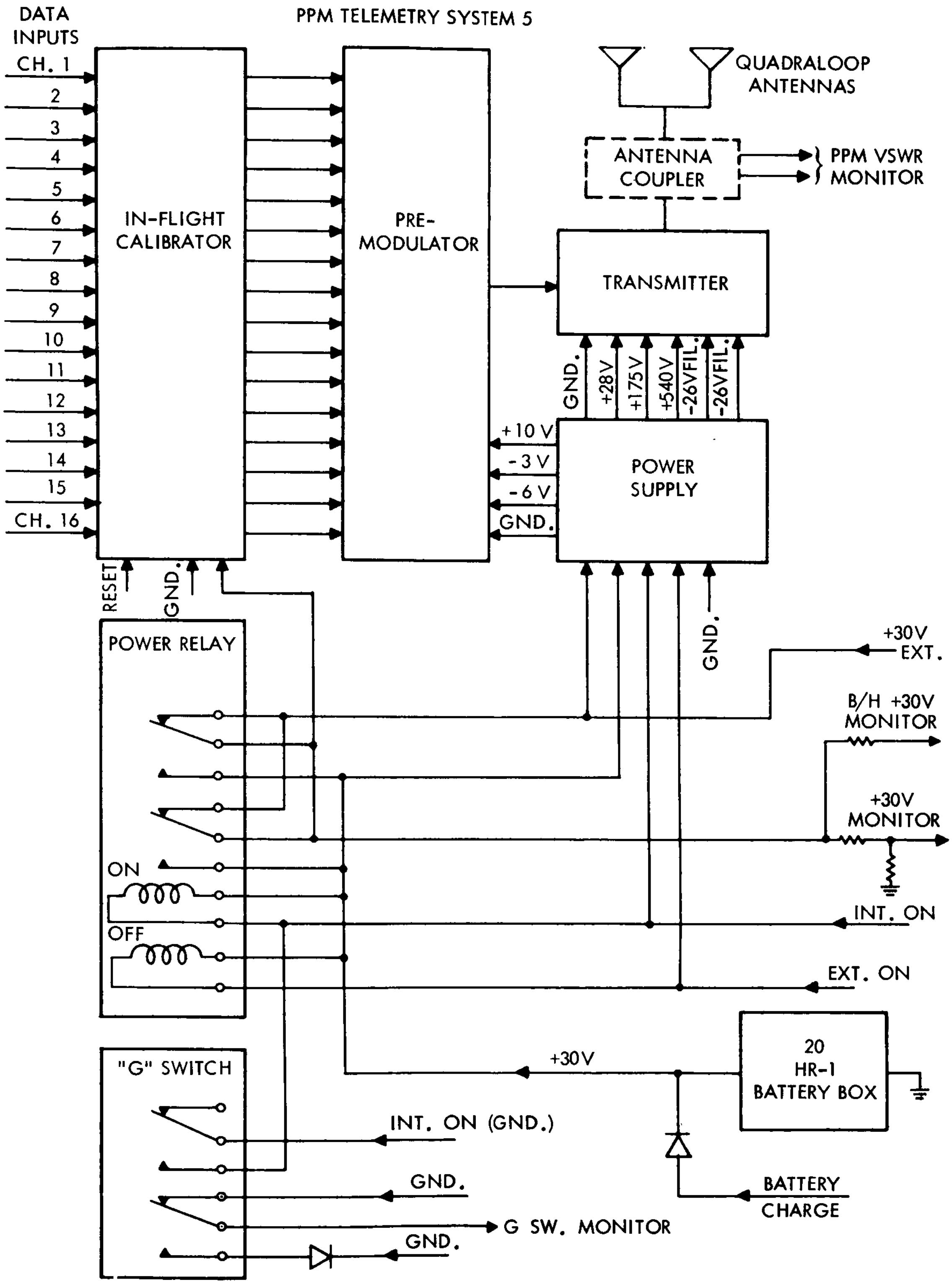


Figure 32. Telemetry System 5, Block Diagram

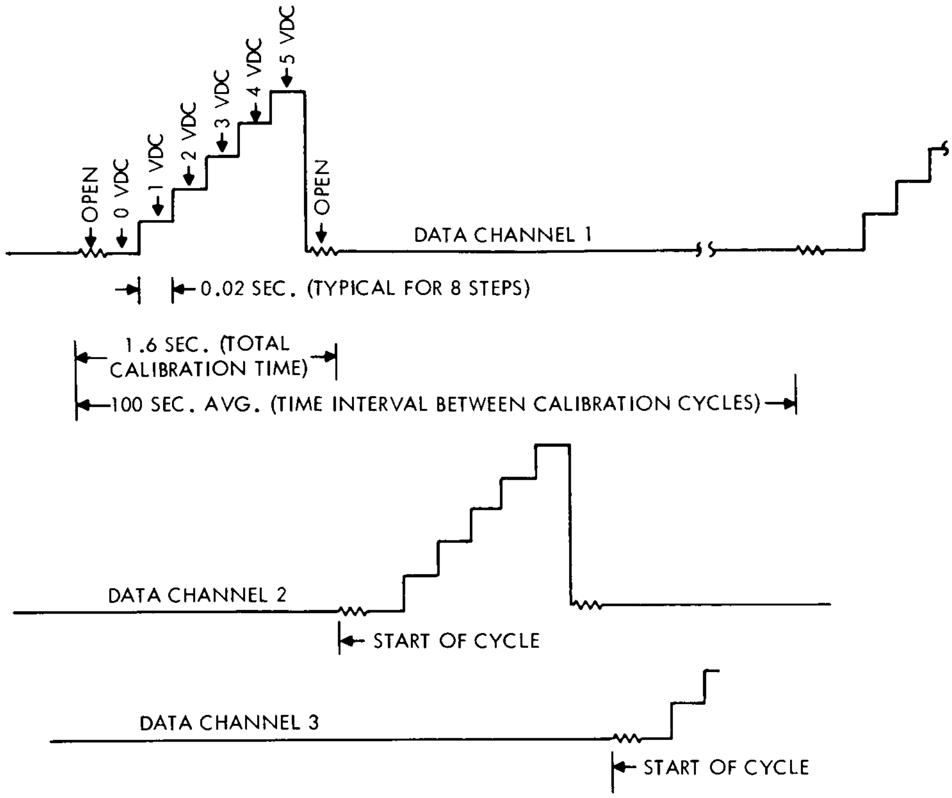


Figure 33. Inflight Calibrator Diagram

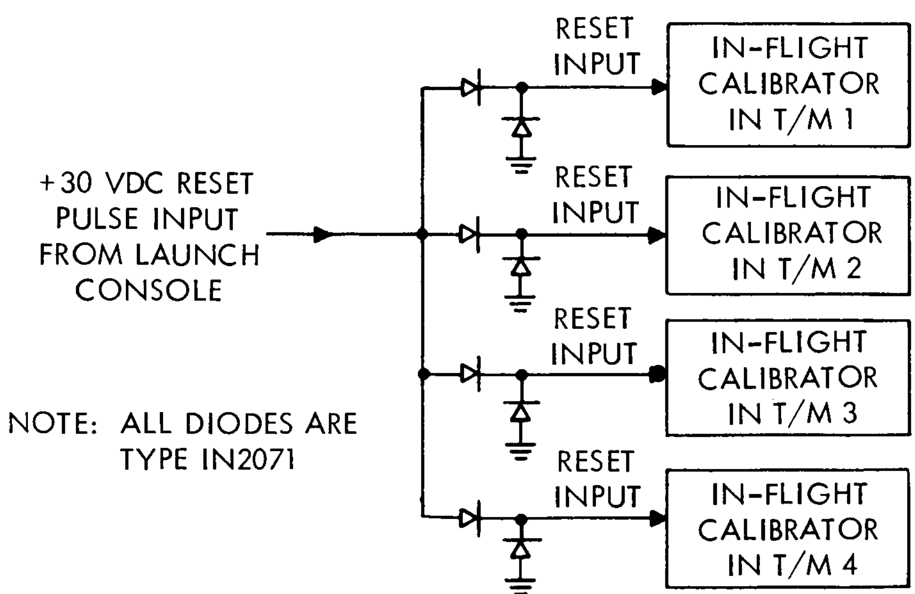


Figure 34. Reset Circuit for In-flight Calibrator for FM/FM Telemetry Systems

The "G" switch and the power control relay will switch the power circuitry of the telemetry system from the external power sources, used on the ground, to the battery used in the rocket. The battery consists of 20 rechargeable Yardney Silver Cels rated at 1 ampere-hour, which may be recharged before flight. The rest of the power control circuitry is similar to that described for the FM/FM Telemetry systems.

The following reference gives a detailed description of the Model SST-1 Telemetry System:

NASA Technical Note (NASA TN D-2151)

Airborne Transistorized Telemeter System Model SST-1

ANTENNA SYSTEMS

There was one pair of antennas for each of the telemetry systems, and one pair of antennas for the Command Safety System. Each antenna system consisted of a harness and two identical quadraloop antennas, mounted on diametrically opposite sides of the rocket. All of the antennas were of the non-flush type, and were mounted on the outer skin of the vehicle. (Refer to Table III and to Figure 35.)

Table III
QUADRALOOP ANTENNAS

Function	Antenna Model No.	Frequency (MHz)	Location
Telemetry System 1	2.007	259.7	Payload Extension
Telemetry System 2	2.007	256.2	Payload Extension
Telemetry System 3	2.007	244.3	Payload Extension
Telemetry System 4	2.007	240.2	Payload Extension
Telemetry System 5	2.055	231.4	Forward Skirt
Command Safety System	4.005	412.0	Forward Skirt

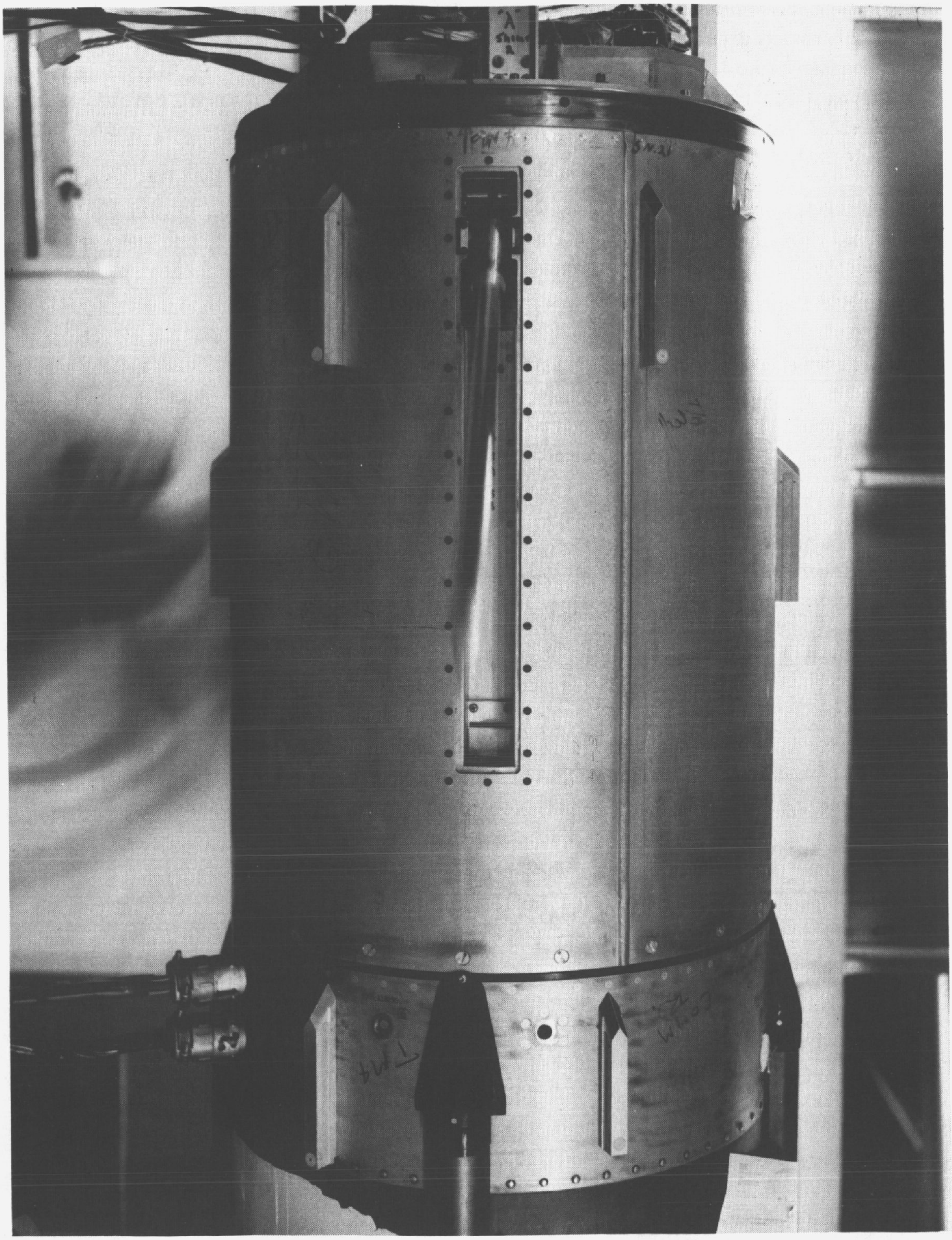


Figure 35. Aerobee 350 Antenna System

The type of quadraloop antennas used had been flight qualified on previous types of NASA sounding rockets. The designer and supplier, New Mexico State University Physical Science Laboratory, made the necessary modifications to the basic design to be compatible with the Aerobee 350.

For telemetry and command safety applications of antennas to sounding rockets, it is desirable to obtain an omni-directional radiation pattern. There should be sufficient gain that both the ground station and the rocket can receive and transmit intelligible signals continually, throughout the flight. The antennas must be impedance matched to the transmitters and receivers, have appropriate bandwidths, and be tuneable to the transmitted and received frequency. They should be mounted along various positions of the rocket without appreciable effect on vehicle performance.

As opposed to the basic loop element, the quadraloop antenna has electrical parameters and a geometric configuration similar to a "quarter-of-a-loop," from which the word "quadraloop" was originated. For impedance analysis purposes, the quadraloop element may be considered as a parallel-wire transmission line with distributed inductance and capacitance. Such a line, when less than quarter-wave in length and short circuited at one end, is inductive. The quadraloop antenna is deliberately made slightly inductive so that a coaxial tuning condenser may be employed at the open circuit end of the line, to resonate the antenna at the design frequency and to provide a finite tuning range for the antenna. The driving point impedance, that is, the position along the transmission line at which the matching generator is attached, may be selected by controlling the antenna design to yield the desired impedance with zero reactance. This feature is advantageous in array design, since two antennas may be connected in parallel by utilizing an appropriate cable harness.

The antenna driving-point impedance was selected at 100 ohms so that a pair of antennas could be connected in parallel by using standard 50-ohm connectors and coaxial cable.

For the cable harness used with the Command Safety System, the same basic theory holds. Since two command receivers are used in parallel with one pair of antennas, a quarter wavelength section of 75-ohm cable had to be inserted in series with a quarter wavelength section of 50-ohm cable to match the resulting 50 ohms at the "tee" between the antennas and the resulting 25 ohms at the "tee" between the two 50-ohm command receiver input impedances. See Figure 36 for antenna impedance configuration. See Figure 37 for Command Safety System Antenna Contour plot.

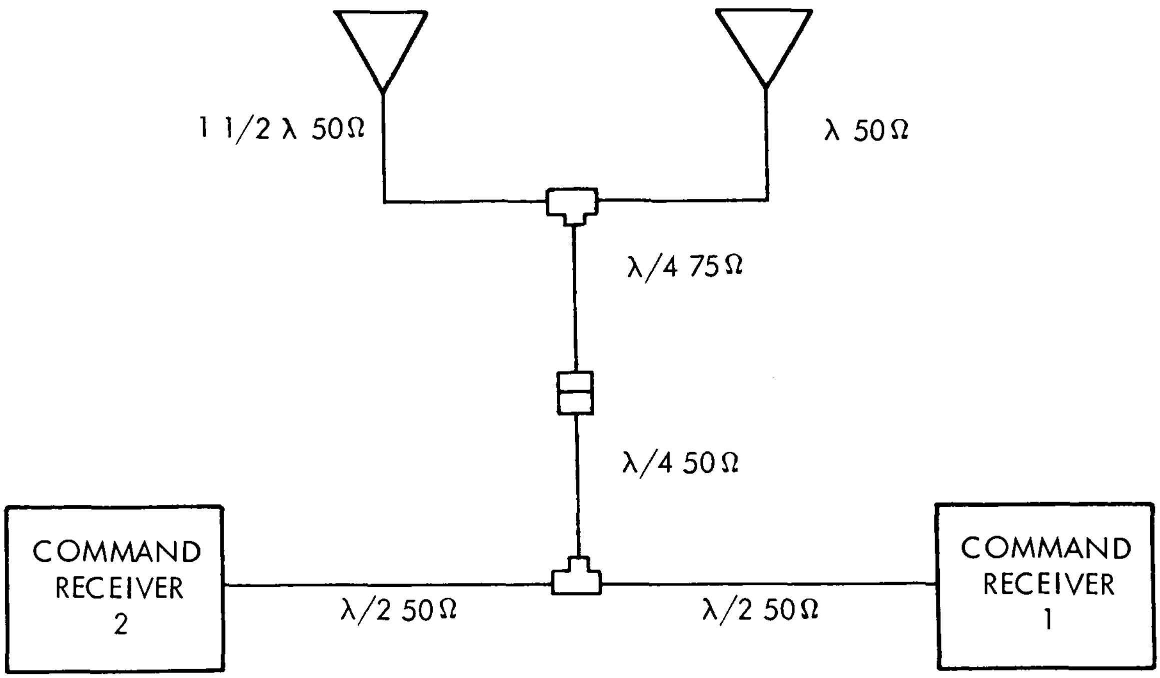


Figure 36. Antenna Impedance Configuration for Flight 17.01

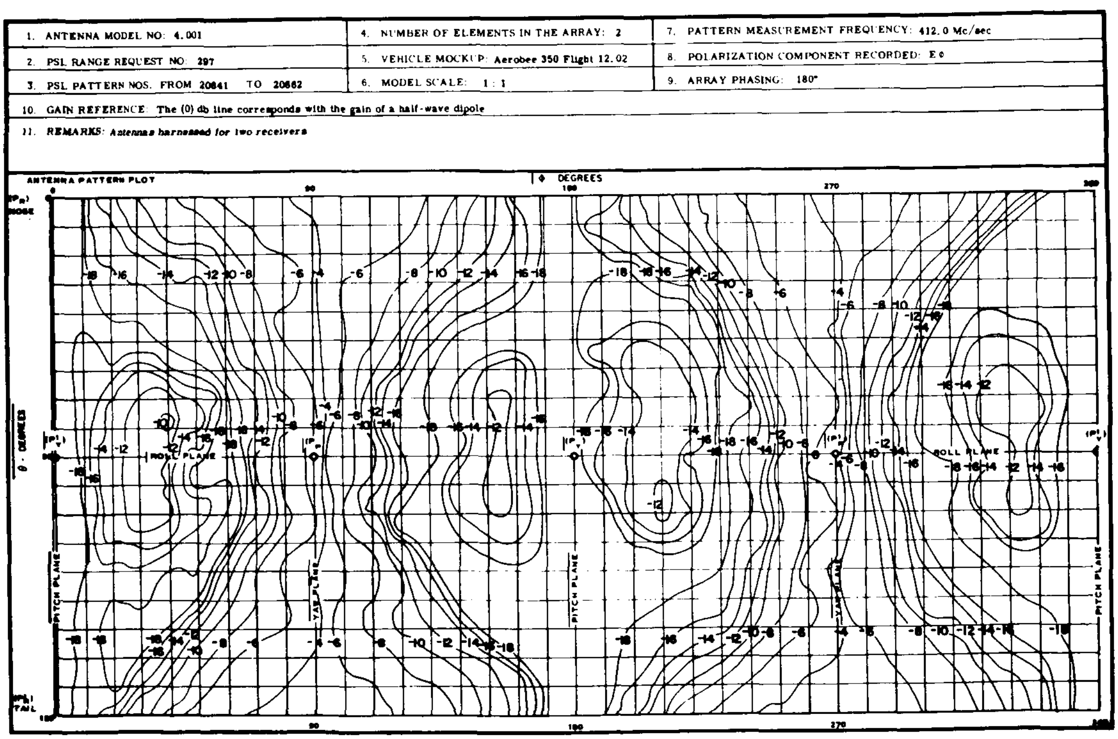


Figure 37. Command Safety System, Antenna Contour Plot

GROUND SUPPORT SYSTEMS

PAYLOAD GROUND CONTROL CONSOLE

The payload ground control console was designed to support large sounding rocket payloads employing up to four telemetry systems. It was necessary to design versatility into the console, allowing for rapid change-over from one payload to another, and to check out two large payloads, such as an Aerobee 350 and an Astrobee 1500, with a minimal time expenditure for the changeover.

A unit design incorporating removeable patch panel boards was generated. Following the decision to utilize five telemeters on Flight 17.01, the payload ground control console was modified to accommodate a fifth system. (See Figure 38 for a front panel view of the console.)

A special circuit was designed into the payload ground control console to prevent the loss of caging power during the caging cycle of the gyro. When the GYRO CAGING pushbutton is pressed, a 45-second timer energizes to lock the power buss in the ON position until the gyro is fully caged. Time required for this operation is approximately 30 seconds. The remaining 15 seconds are provided as a safety margin. Should the gyro not receive power for 30 seconds, the erection motors cease operating, partially caging the gyro. During the caging operation, power required to uncage the gyro is locked out of the system by an electrical system interlock. This necessitates the removal of the payload or gyro from the rocket for correction.

COMMAND SAFETY SYSTEM

A Command Safety System (Figure 39) was utilized to provide propellant shutoff or a command-destruct capability. The purpose of the system was to provide a complete electrical checkout, remote arming, and charging of the pyrotechnic batteries from the blockhouse. This system was designed by the Sounding Rocket Instrumentation Section of Goddard Space Flight Center. A Ledex position function block diagram for the command system console is shown in Figure 40.

GROUND STATION SUPPORT

Six (fixed and mobile) FM and PPM telemetry ground stations support these preflight checkouts and the launch tracking of Aerobee 350 Flights. Five of the

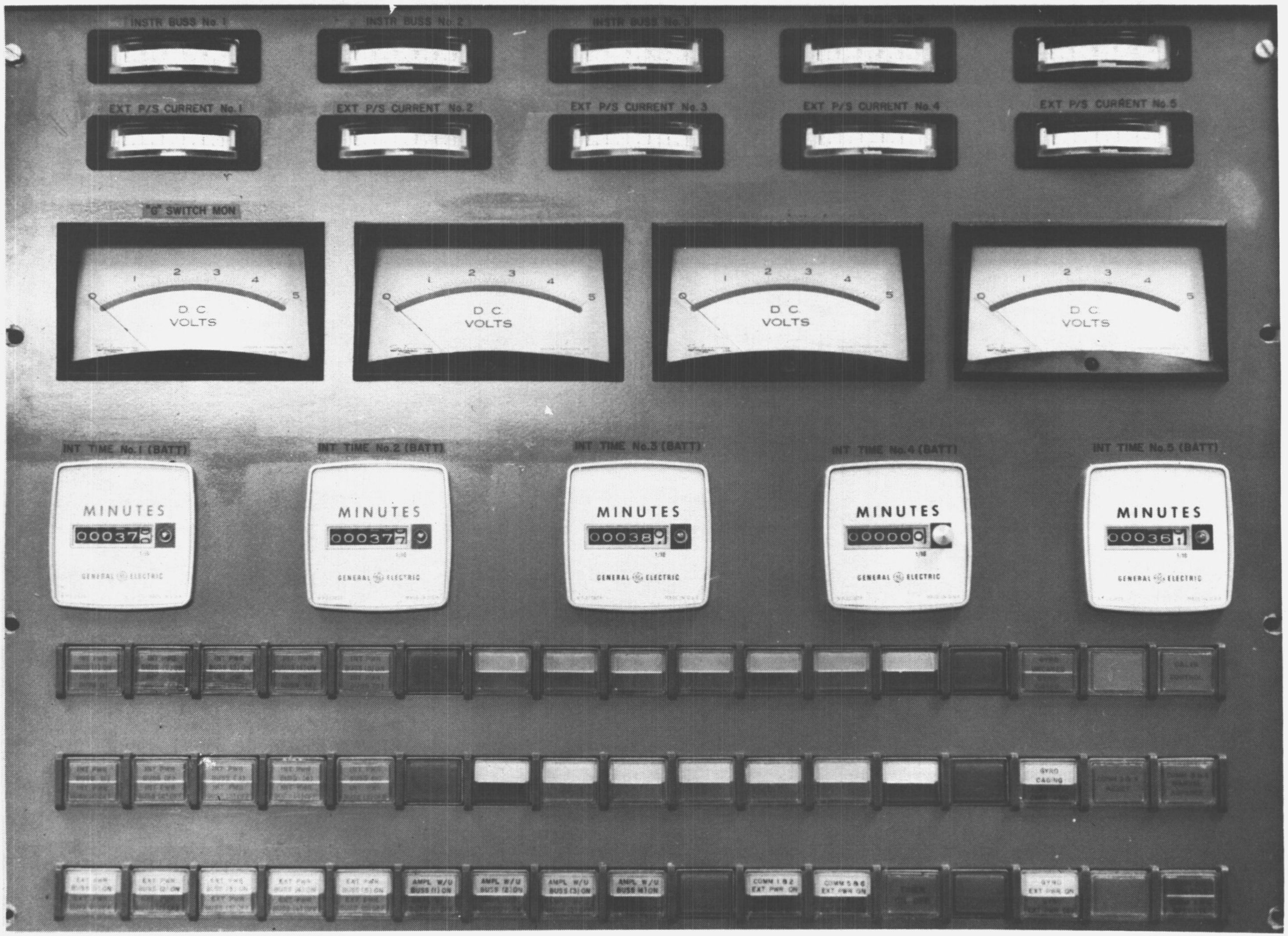


Figure 38. Payload Ground Control Console

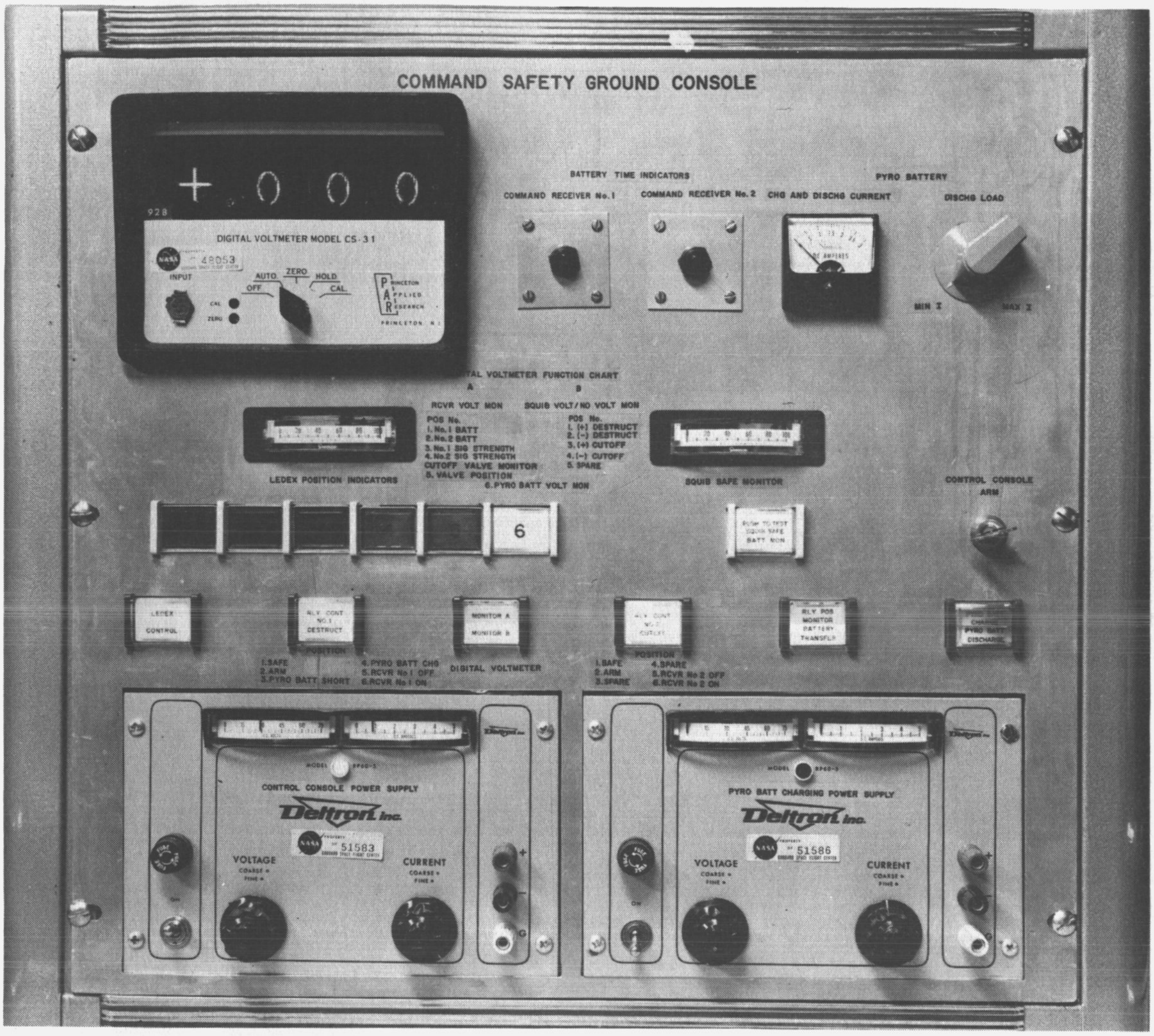


Figure 39. Command Safety System, Ground Control Console

six stations were under the cognizance of GSFC's Sounding Rocket Instrumentation Section, while the remaining station, located at Wallops Island Main Base (WIMB), was operated by Wallops personnel. Original tapes from each recording ground station were sent to GSFC for retention and data reduction.

WALLOPS MAIN BASE STATION. This telemetry ground station was used to perform the horizontal and vertical preflight tests, and to track the rocket during its flight. As Wallops was a primary telemetry receiving facility, the station was equipped with one General Bronze (high-gain), and one Agave medium-gain antenna. Outfitted with a multicoupler, the high-gain antenna received the telemetry signals. In addition, WIMB provided r-f tapes for GSFC's Station C and for the trailer pad.

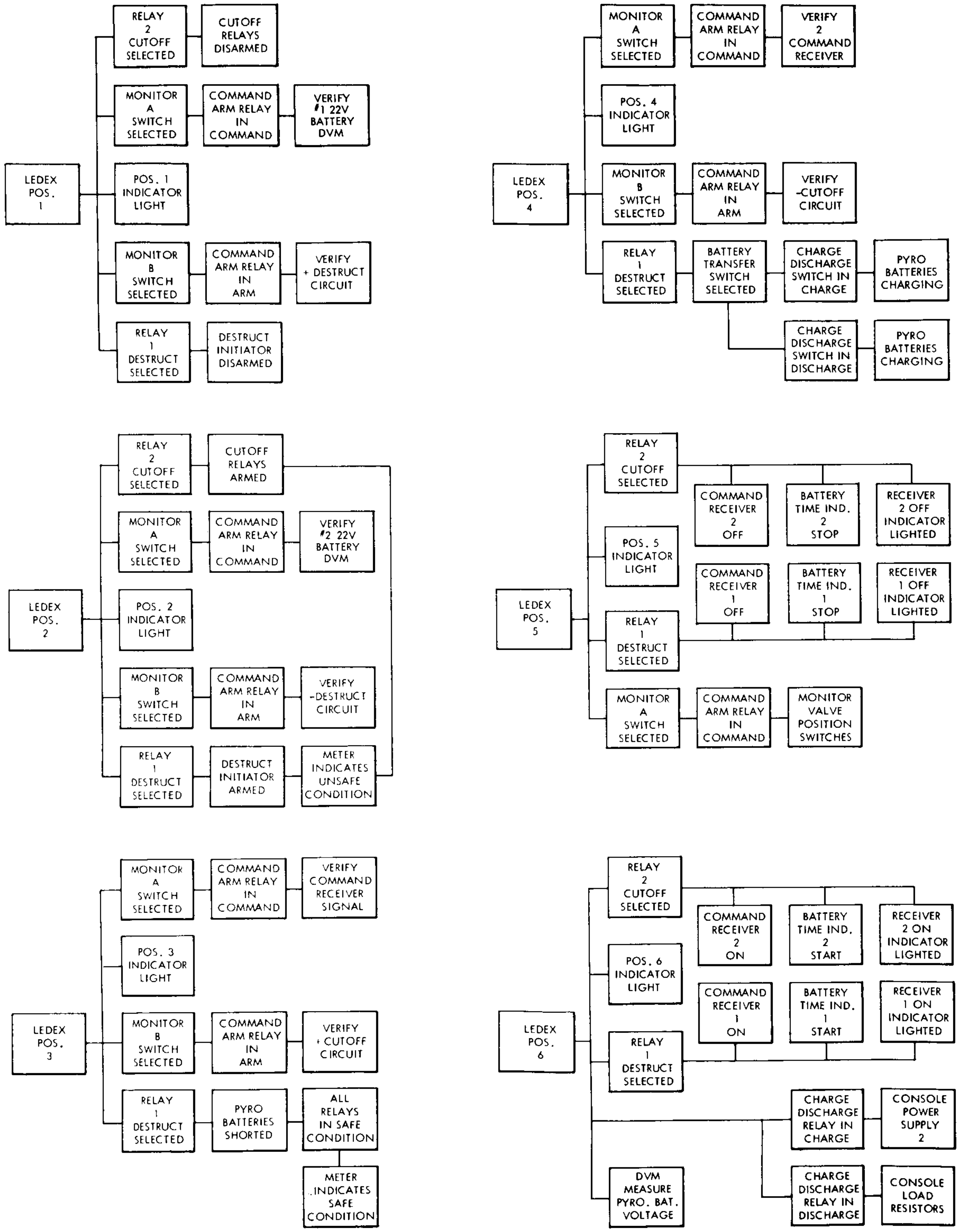


Figure 40. Command Safety System, Ledex Position, Functional Block Diagram

WIMB was responsible for preparing a seven-track, IRIG-standard, magnetic tape at 60 inches per second. The tape was one-half inch in width, wound on a 14-inch reel, and included 100-kilohertz tape-speed compensation, and 17-kilohertz servo control.

GSFC TELEMETRY RECEIVING STATIONS

Other telemetry receiving stations assigned to support the Aerobee 350 during preflight and actual flight were Stations A, C, G, H, and J.

TELEMETRY STATION A. Station A was a fixed telemetry receiving station (see Figure 41), located on Wallops Island in Blockhouse No. 1. Capabilities of this station included the ability to receive and copy both FM/FM and PPM/AM transmissions.

Two of the station's antennas were of the eight-turn tri-helix configuration. Both of these antennas were assigned to track payload telemetry transmissions; one tri-helix for the PPM, and the other antenna, outfitted with a multicoupler, assigned to the FM frequencies. Magnetic tape requirements for Station A were similar to those for Wallops Main Base.

FM capabilities at Station A included: nineteen subcarrier discriminators; two magnetic oscillographs for real-time recordings; two seven-channel IRIG tape recorders; five receivers; and a diversity combiner. Recording facilities for PPM at Station A consisted of three 6-channel (NRL) type racks.

TELEMETRY STATION C. Station C was a fixed telemetry receiving and recording facility located in the Wallops Main Base telemetry building. (See Figure 42.) Capabilities of this station were limited to PPM recording only. Input r-f was received from Wallops by way of a multicoupler connection. Contained within this PPM station were three six-channel NRL type racks to record, in real time, data transmitted from the airborne PPM telemetry system. Magnetic flight recordings were recorded by the seven-channel high-frequency tape unit.

TELEMETRY STATION G. This telemetry station was a fixed station located at GSFC's facility in Beltsville, Maryland. (See Figure 43.) Antenna capabilities at Station G included one tri-helix and two single-helix antennas, all of the eight-turn configuration, and all manually operated. The station was capable of receiving and recording both FM/FM and PPM transmissions.

FM/FM capabilities, at Station G, included: 16 subcarrier discriminators; two magnetic oscillographs, for both real-time and playback records; two seven-channel magnetic recorders; four receivers; and a BCD time code generator.

Figure 41. Telemetry Station A

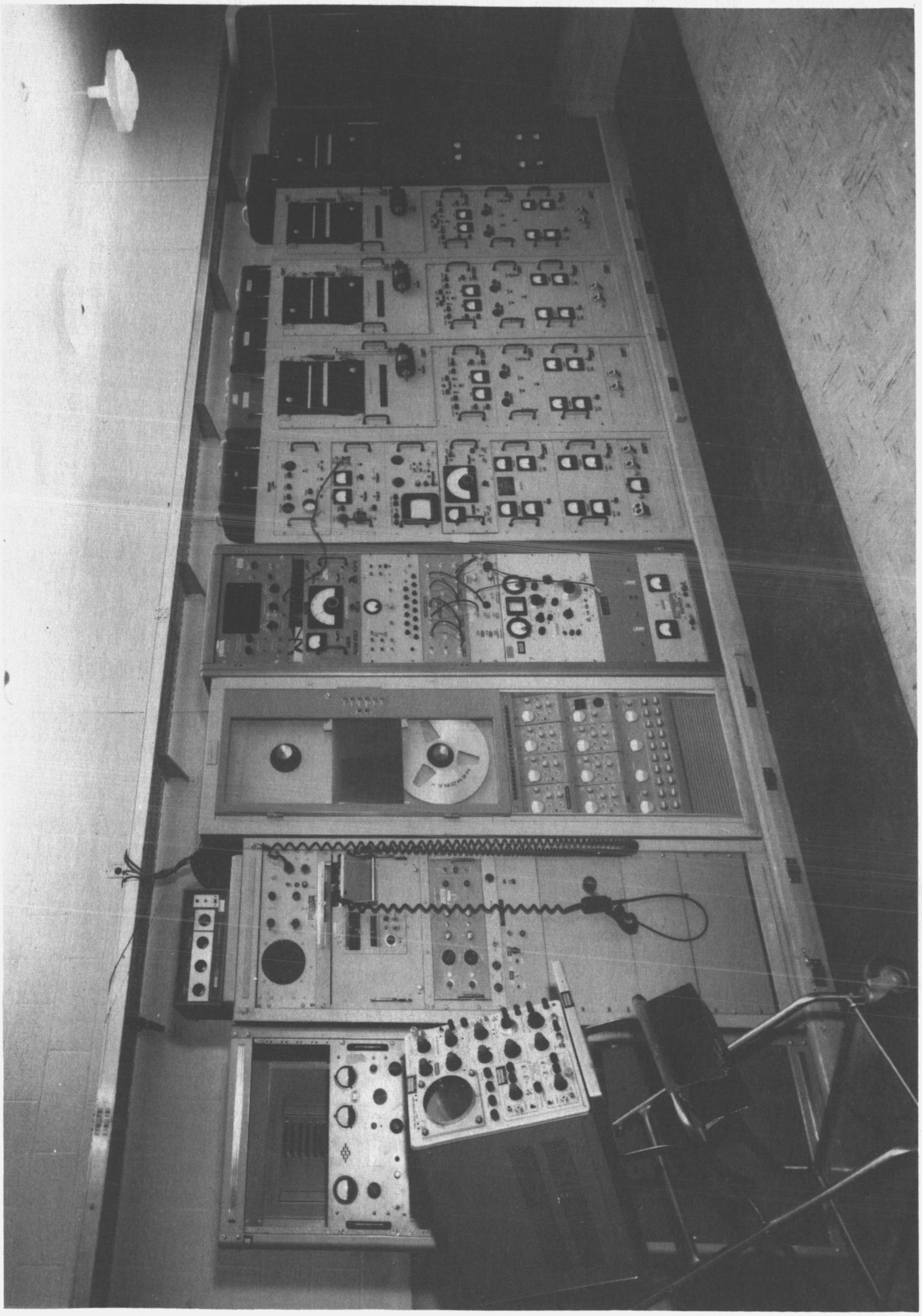


Figure 42. Telemetry Station C

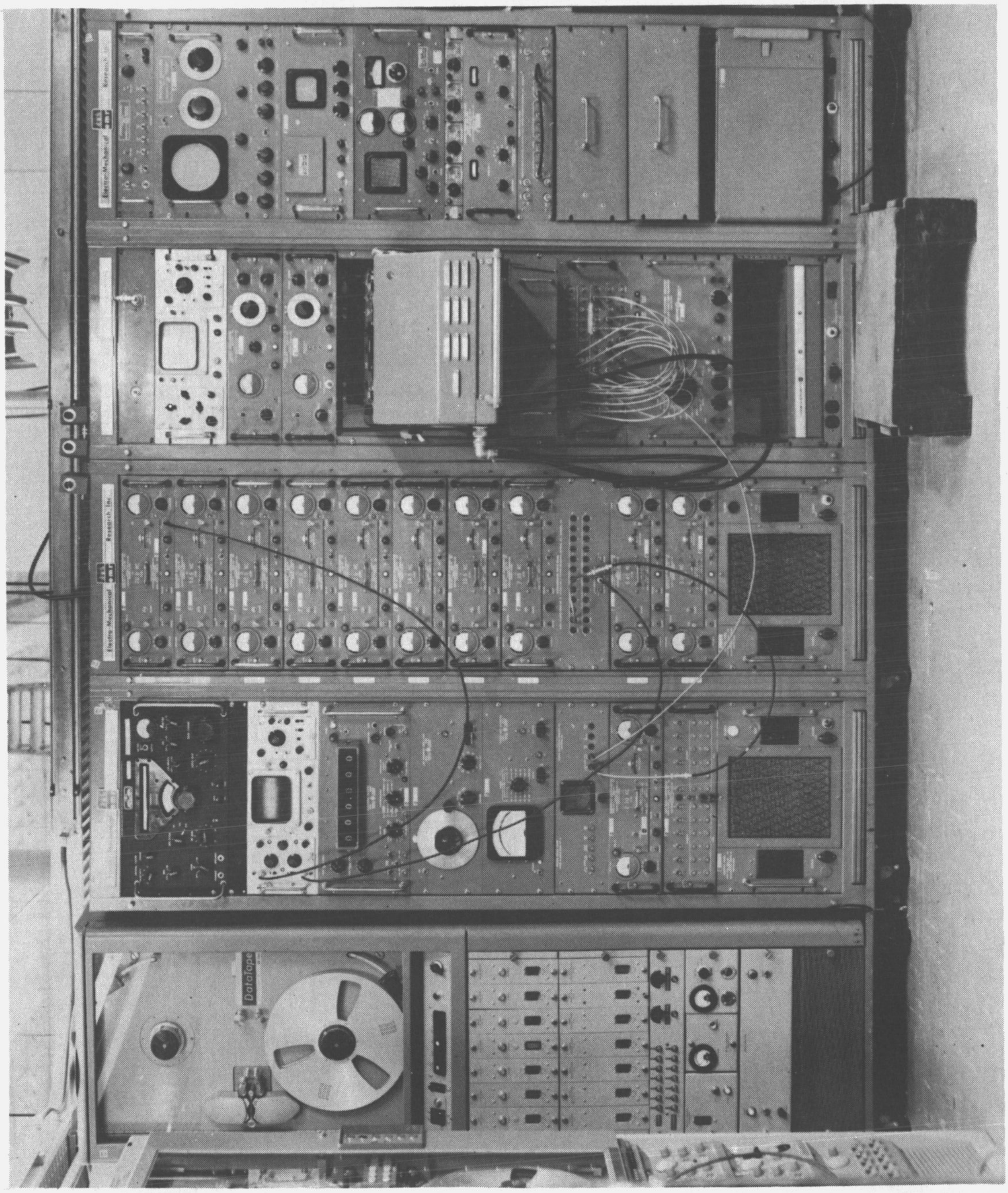


Figure 43. Telemetry Station G

In support of Aerobee 350 Flights, Station G was responsible for recording the integration checks conducted at GSFC's Beltsville building, and for monitoring the flight of the rockets. Sole responsibilities assigned to Station G during the flights of the Aerobee 350 were to record, on magnetic tape, the countdown and the flight telemetry data.

TELEMETRY STATION H. Station H, a 28-foot semi-trailer, was a mobile FM/FM telemetry receiving and recording facility which was transported to Wallops Island specifically to support Flight 17.01 during horizontal and vertical preflight checks, and during the launch period. (See Figure 44.)

Antenna capabilities of this mobile station consisted of two manually operated, eight-turn, single-helix antennas, one of which was equipped with a multicoupler. Normally, this ground station is equipped with only a single magnetic oscillograph; however, to support the 17.01 payload, another oscillograph was temporarily installed in the trailer.

Magnetic tape requirements for Station H were similar to those of Wallops Main Base.

TELEMETRY STATION J. The final telemetry receiving facility, supporting the preflight checks and launch of Flight 17.02, was Station J, a GSFC mobile unit. (See Figure 45.) Station J was housed in a semi-trailer located adjacent to blockhouse number 1 at the Wallops Island launch facility. Antennas for the reception of telemetry data for this payload were provided by Station A. Four magnetic oscillograph recorders were employed in this receiving station to provide real-time PPM data in an allocation identical to that of GSFC's Station A, and in the same format.

WALLOPS ISLAND UMBILICAL CABLE INSTALLATION

Permanently connected umbilical cables were installed from the blockhouse to the third and seventh levels of the tower. The tower was modified to provide five distribution boxes for wiring from the tower to the blockhouse. The wiring for the Aerobee 150A was terminated on the third level, and was paralleled by identical terminal boxes on the seventh level, for the Aerobee 350. Terminations in the blockhouse were identical to those in the tower.

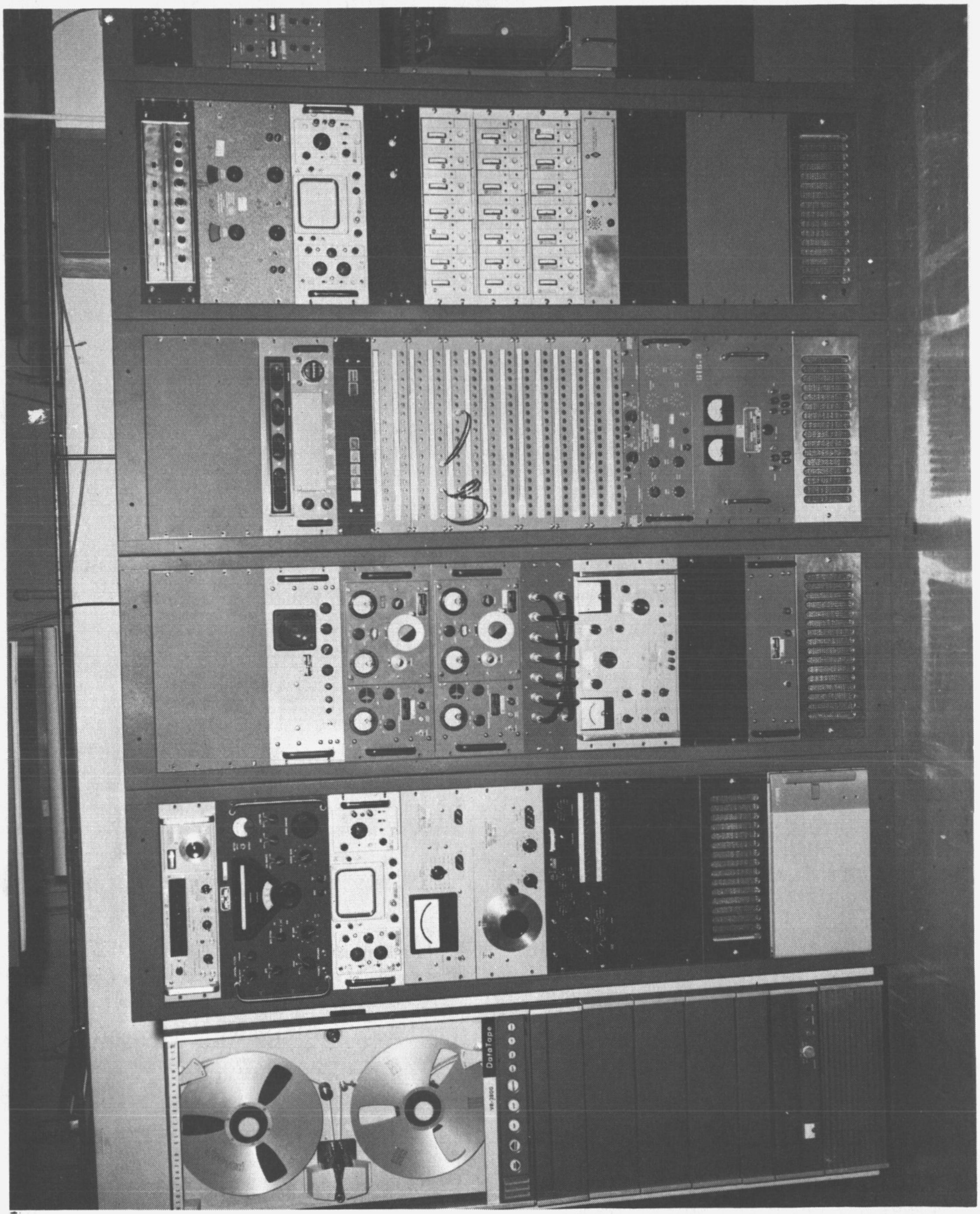


Figure 44. Telemetry Station H

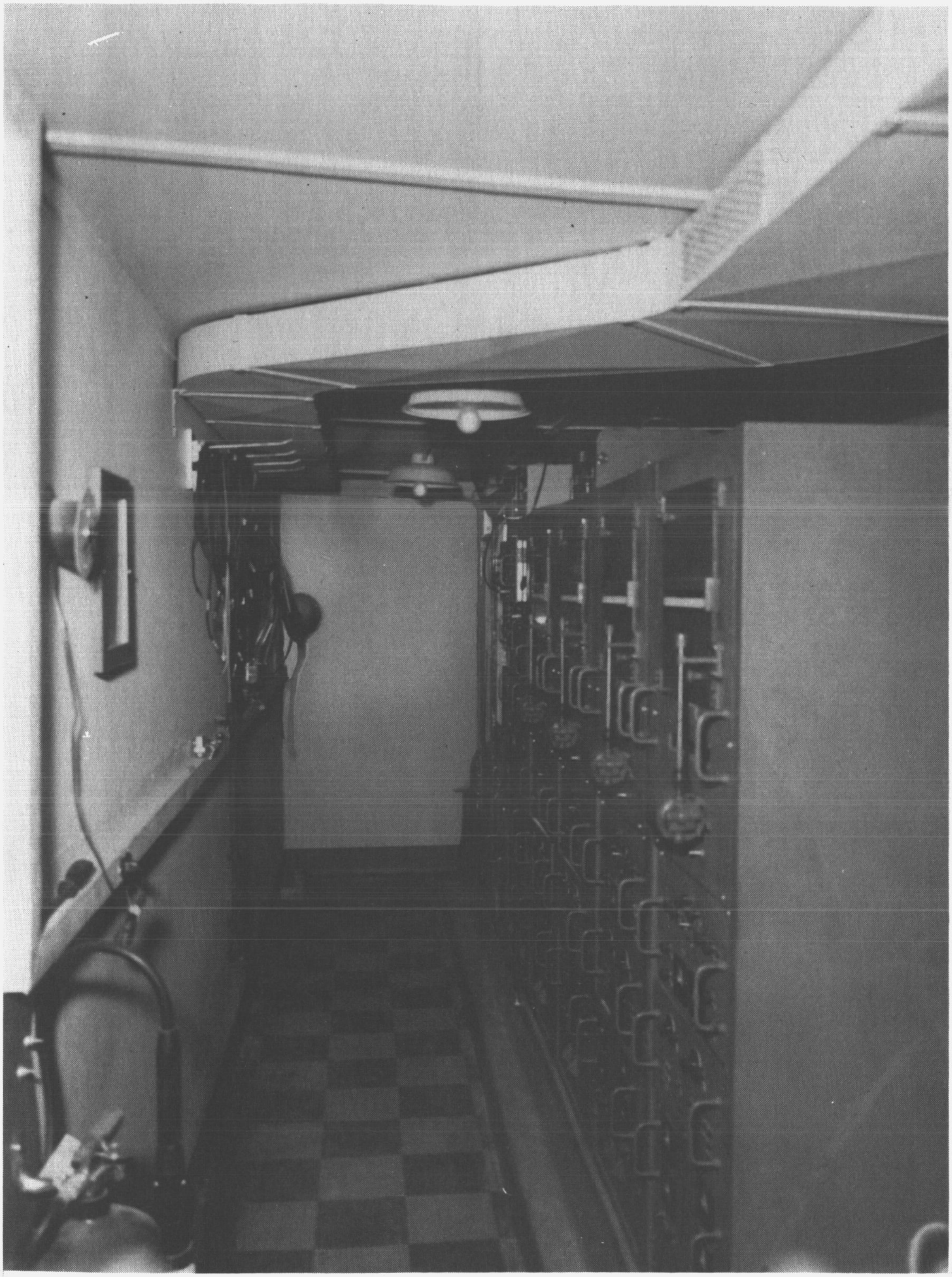


Figure 45. Telemetry Station J

APPENDIX A

12.01 TELEMETRY ALLOCATIONS

Telemetry 1 - 240.2 Megahertz (FM/FM)

IRIG Channel	Data Function
18	Riding Shoe Load-Cell
17	Riding Shoe Load-Cell
16	Riding Shoe Load-Cell
15	Commutator "B"
14	Commutator "A"
13	Oxidizer Line Pressure
12	Fuel Line Pressure (manifolded)
11	Helium Gas Regulator Pressure
10	Helium Gas Bottle Pressure
9	Nike Booster Chamber Pressure

12.01 TELEMETRY ALLOCATIONS (continued)

Telemetry 2 - 231.4 Megahertz (FM/FM)

IRIG Channel	Data Function
E	Payload Vibration, "Z" axis
C	Payload Vibration, "Y" axis
A	Payload Vibration, "X" axis
13	Strain Gage, Booster Thrust Structure Strut
12	Strain Gage, Booster Thrust Structure Strut
11	Strain Gage, Booster Thrust Structure Strut
10	Strain Gage, Booster Thrust Structure Strut
9	Vehicle Acceleration, "X" axis
8	Vehicle Acceleration, "Y" axis
7	Vehicle Acceleration, "Z" axis
6	Stable Platform, Pitch
5	Stable Platform, Yaw
4	Stable Platform, Roll

12.01 TELEMETRY ALLOCATIONS (continued)

COMMUTATOR CHANNEL A

Segment	Function
1	Aerodynamic Yaw
2	Aerodynamic Pitch
3	Lateral Magnetometer
4	Longitudinal Magnetometer
5	Signal Ground
6	Signal Ground
7	Aerodynamic Yaw
8	Aerodynamic Pitch
9	Lateral Magnetometer
10	Longitudinal Magnetometer
11	+4 Volts dc Calibration
12	+4 Volts dc Calibration
13	Aerodynamic Yaw
14	Aerodynamic Pitch
15	Lateral Magnetometer
16	Longitudinal Magnetometer
17	+3 Volts dc Calibration
18	+3 Volts dc Calibration

12.01 TELEMETRY ALLOCATIONS (continued)

COMMUTATOR CHANNEL A (continued)

Segment	Function
19	Aerodynamic Yaw
20	Aerodynamic Pitch
21	Lateral Magnetometer
22	Longitudinal Magnetometer
23	Power Ground
24	Power Ground
25	Aerodynamic Yaw
26	Aerodynamic Pitch
27	Lateral Magnetometer
28	Longitudinal Magnetometer

12.01 TELEMETRY ALLOCATIONS (continued)

COMMUTATOR CHANNEL B

Segment	Function
1	Payload Interval Pressure
2	Helium Regulator Inlet Temperature
3	Helium Regulator Outlet Temperature
4	Monitor for Current to H_2O Dump Squib
5	Monitor for Current to Nose Cone Separation Squib
6	Monitor for Nose Cone Separation
7	Command Receiver Signal Strength
8	Monitor for Current to Main Chute Deployment Squib
9	+4 Volts dc Calibration
10	Power Ground
11	Payload Interval Pressure
12	Helium Regulator Inlet Temperature
13	Helium Regulator Outlet Temperature
14	Monitor for Current to H_2O Dump Squib
15	Monitor for Current to Nose Cone Separation
16	Monitor for Nose Cone Separation
17	Command Receiver Signal Strength
18	Monitor for Current to Main Chute Deployment Squib

12.01 TELEMETRY ALLOCATIONS (continued)

COMMUTATOR CHANNEL B (continued)

Segment	Function
19	+ 3 Volts dc Calibration
20	Signal Ground
21	Payload Interval Pressure
22	Helium Regulator Inlet Temperature
23	Helium Regulator Outlet Temperature
24	Monitor for Current to H_2O Dump Squib
25	Monitor for Current to Nose Cone Separation Squib
26	Monitor for Nose Cone Separation
27	Command Receiver Signal Strength
28	Monitor for Current to Main Chute Deployment Squib

17.01 TELEMETRY ALLOCATIONS

Telemetry 1 - 259.7 Megahertz (FM/FM)

IRIG Channel	Data Function
E	Payload Thrust Vibration (X) (on block with Z accel—mounted by rail C)
C	Payload Lateral Vibration (Y) (between rails B and D—mounted by rail D)
A	Payload Lateral Vibration (Z) (between rails C and A—mounted by rail C)
13	(PC I) Chamber Pressure
12	(P_{fj} I) Pressure Fuel Injector (upstream of Nike Valve)
11	(P_{oj} I) Pressure Oxidizer Injector (manifold)
10	(P_{ol} I) Pressure Oxidizer Line (upstream of Nike Valve)
9	(P_{fl} I) Pressure Fuel Line (Fuel Jacket inlet)
8	(S/G 1) Strain Gage (bending moment—regulator section)
7	(S/G 2) Strain Gage (bending moment—regulator section)
6	Stable Platform—Gyro—(Roll)
5	Nike Valve Monitor I

Telemetry 2 - 256.2 Megahertz (FM/FM)

IRIG Channel	Data Function
E	Engine Cluster Vibration Thrust (X) (180° out of phase with P/L thrust vibration)
C	Engine Cluster Vibration Lateral (Y) (between rails D and B)
A	Engine Cluster Vibration Lateral (Z) (between rails C and A) (180° out of phase with P/L vibration pickup)
13	(PC II) Chamber Pressure
12	(P_{fj} II) Pressure Fuel Injector (upstream of Nike Valve)
11	(P_{oj} II) Pressure Oxidizer Injector (manifold)
10	(P_{ot}) Pressure Oxidizer Tank (outlet fitting)
9	(P_{nk}) Pressure Nike Booster
8	(S/G 3) Strain Gage (bending moment—regulator section)
7	(S/G 4) Strain Gage (bending moment—regulator section)
6	Stable Platform—Gyro—(Pitch) (between Fins I and III)
5	Nike Valve Monitor II

17.01 TELEMETRY ALLOCATIONS (continued)

Telemetry 3 - 244.3 Megahertz FM/FM

IRIG Channel	Data Function
18	Solar Aspect
17	Comm. 62.5rs Forward Temperature
16	Payload Accelerometer Thrust (X)
15	Payload Accelerometer Lateral (Y) Pitch (between Fins I and III)
14	Payload Accelerometer Lateral (Z) Yaw (between Fins II and IV)
13	(PC III) Chamber Pressure
12	(P_{fj} III) Pressure Fuel Injector (upstream of Nike Valve)
11	(P_{oj} III) Pressure Oxidizer Injector (manifold)
10	(P_{ol} III) Pressure Oxidizer Line (upstream of Nike Valve)
9	(P_{fl} III) Pressure Fuel Line (Fuel Jacket inlet)
8	(S/G 5) Strain Gage (bending moment—regulator section)
7	(S/G 6) Strain Gage (bending moment—regulator section)
6	Stable Platform—Gyro—Yaw (between Fins II and IV)
5	Nike Valve monitor III
4	PPM VSWR Monitor

17.01 TELEMETRY ALLOCATIONS (continued)

Telemetry 4 - 240.2 Megahertz FM/FM

IRIG Channel	Data Function
18	Comm. 1, 5.0 r/s Aft Voltage monitor
17	Comm. 2, 5.0 r/s Aft Temperatures (T 21 ----32°-600° F)
16	Comm. 3, 215 r/s Transducers
15	Comm. 4, 2.5 r/s Thermocouples
14	Comm. 5, 2.5 r/s Voltage Monitors
13	(PC IV) Chamber Pressure
12	(P_{fj} IV) Pressure Fuel Injector (upstream of Nike Valve)
11	(P_{oj} IV) Pressure Oxidizer Injector (manifold)
10	(P_{gr}) Pressure Gas Regulator Outlet, Helium manifold
9	(P_{gb}) Pressure Gas Bottle (Helium Regulator inlet in line with Fin IV)
8	(S/G 7) Strain Gage (bending moment—regulator section)
7	(S/G 8) Strain Gage (bending moment—regulator section)
6	Ogive (pitch) (between Fins I and III) Fin III + Pitch
5	Nike Valve monitor IV
4	Ogive (Yaw) (between Fins II and IV) Fin IV + Yaw

17.01 TELEMETRY ALLOCATIONS (continued)

Telemetry 5 - 231.4 Megahertz (PPM)

Data Channel	Data Function
1	Tgb
2	Tgr
3	T_{CI} (II)
4	T_{CI} (IV)
5	T_{CO} (I)
6	T_{CO} (II)
7	T_{CO} (III)
8	T_{CO} (IV)
9	Longitudinal Magnetometer
10	Latitudinal Magnetometer
11	Fin Base Pressure Pfb BLH 24
12	Tail Can Base Pressure Ptb BLH 25
13	Nose Cone Pressure Pnc BLH 26
14	Ogive (Pitch)
15	Cutoff Valve Position Monitor
16	Booster Tail Switch Monitor (Tail Switch eliminated)

Commutator 1 (T/M 4 - 70 Kilohertz) - Aft Voltage Monitors

Segment	Function
1	Ground
2	5 volts
3	BLH Amplifier 1
4	BLH Amplifier 2
5	BLH Amplifier 3
6	BLH Amplifier 4
7	BLH Amplifier 5
8	BLH Amplifier 6
9	BLH Amplifier 7
10	BLH Amplifier 8
11	BLH Amplifier 9
12	BLH Amplifier 10
13	BLH Amplifier 11
14	BLH Amplifier 12
15	BLH Amplifier 13
16	BLH Amplifier 14
17	BLH Amplifier 15
18	BLH Amplifier 16

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 1 (T/M 4 - 70 Kilohertz) - Aft Voltage Monitors (continued)

Segment	Function
19	BLH Amplifier 17
20	BLH Amplifier 18
21	±18 volts regulator monitor (aft)
22	BLH Amplifier 25
23	Gulton Amplifier 9
24	Gulton Amplifier 10
25	Gulton Amplifier 11
26	Gulton Amplifier 12
27	Gulton Amplifier 13
28	Gulton Amplifier 14

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 2 (T/M 4 - 52.5 Kilohertz) - Aft Temperature

Segment	Function
1	Calibration Step 1 - 130 ohms
2	Body shroud skin temperature (aft, inside) T 15
3	Ox tank skin temperature (forward, outside) T 16
4	Ox tank skin temperature (aft, outside) T 17
5	Tank assembly aft skirt skin temperature (inside) T 18
6	Tail cylinder skin temperature (forward, inside) T 19
7	Calibration step 2 - 125 ohms
8	Tail cylinder skin temperature (aft, inside) T 20
9	Tail cylinder aft ring (inside, 45° to Fins II and III)
10	Tail cylinder aft ring (inside, in line with Fin II) T 29
11	Tail cylinder base plate temperature (inside, 8" from vehicle centerline toward Fin II) T 27
12	Tail cylinder base plate temperature (inside, 8" from vehicle centerline toward Fin IV) T 24
13	Calibration step 3 - 100 ohms
14	Tail cylinder base plate temperature (inside, 2" from vehicle centerline toward Fin IV) T 25
15	Tail cylinder base plate temperature (inside, 2" from vehicle centerline toward Fin II) T 26

Commutator 2 (T/M 4 - 52.5 Kilohertz) - Aft Temperature (continued)

Segment	Function
16	Engine Compartment, gas temperature (mounted on engine cluster A-frame member) T 21
17	Thrust chamber head temperature (number II chamber) T 22
18	Thrust cluster support—diagonal brace T 23
19	Booster thrust structure strut temperature (number I strut, middle, outside surface) T 39
20	Calibration Step 4 - 75 ohms
21	Booster thrust structure strut temperature (number I strut, aft, outside surface) T 40
22	Nike motor wall temperature (forward, outside) T 41
23	Nike motor wall temperature (middle, outside) T 42
24	Nike fin skin temperature (root-chord, forward, inside) T 43
25	Nike fin skin temperature (mid-chord, forward, inside) T 44
26	Nike fin skin temperature (tip-chord, forward, inside) T 45
27	Nike fin skin temperature (mid-chord, middle, inside) T 46
28	Calibration step 5 - 50 ohms

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 3 (T/M 4 - 40 Kilohertz) - Transducers

Segment	Function
1	Tail Can Base Pressure Ptb
2	Nose Cone Pressure Pnc
3	Calibration step 1 - Ground
4	Fin Base Pressure Pfb
5	Tail Can Base Pressure Ptb
6	Magnetometer Longitude
7	Magnetometer Latitude
8	Nose Cone Pressure Pnc
9	Calibration step 2 - 1 volt
10	Fin Base Pressure Pfb
11	Tail Can Base Pressure Ptb
12	Magnetometer Longitude
13	Magnetometer Latitude
14	Nose Cone Pressure Pnc
15	Calibration step 3 - 2 volts
16	Fin Base Pressure Pfb
17	Tail Can Base Pressure Ptb
18	Magnetometer Longitude

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 3 (T/M 4 - 40 Kilohertz) - Transducers (continued)

Segment	Function
19	Magnetometer Latitude
20	Nose Cone Pressure Pnc
21	Calibration step 4 - 3 volts
22	Fin Base Pressure Pfb
23	Tail Can Base Pressure Ptb
24	Magnetometer Longitude
25	Magnetometer Latitude
26	Nose Cone Pressure Pnc
27	Calibration step 5 - 5 volts
28	Fin Base Pressure Pfb

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 4 (T/M 4 - 30 Kilohertz) - Thermocouples and Miscellaneous Monitors

Segment	Function
1	Calibration step 1 - Ground
2	Calibration step 2 - 5 volts
3	Tgb (regulator)
4	Tgr (regulator)
5	T_{CI} (II)
6	T_{CI} (IV)
7	T_{CO} (I)
8	T_{CO} (II)
9	T_{CO} (III)
10	T_{CO} (IV)
11	Tgb
12	Tgr
13	T_{CI} (II)
14	T_{CI} (IV)
15	T_{CO} (I)
16	T_{CO} (II)
17	T_{CO} (III)
18	T_{CO} (IV)

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 4 (T/M 4 - 30 Kilohertz) - Thermocouples and Miscellaneous Monitors (continued)

Segment	Function
19	Calibration step 3 - 3 volts
20	Command Receiver Channel Monitor
21	T/M 1 + 30-volt monitor
22	T/M 2 + 30-volt monitor
23	T/M 3 + 30-volt monitor
24	T/M 4 + 30-volt monitor
25	T/M 5 + 30-volt monitor
26	Cutoff valve position monitor
27	Booster Tail Switch monitor
28	Calibration step 4 - 1 volt

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 5 (T/M 4 - 22 Kilohertz) - Forward Voltage Monitors

Segment	Function
1	Calibration step 1 - Ground
2	Calibration step 2 - 5 volts
3	BLH Amplifier 19
4	BLH Amplifier 20
5	BLH Amplifier 21
6	BLH Amplifier 22
7	BLH Amplifier 23
8	BLH Amplifier 26
9	Gulton Amplifier 1
10	Gulton Amplifier 2
11	Gulton Amplifier 3
12	Gulton Amplifier 4
13	Gulton Amplifier 5
14	Gulton Amplifier 6
15	Gulton Amplifier 7
16	Gulton Amplifier 8
17	Gulton Amplifier 15
18	Gulton Amplifier 16

Commutator 5 (T/M 4 - 22 Kilohertz) - Forward Voltage Monitors (continued)

Segment	Function
19	6-volt monitor 1
20	6-volt monitor 2
21	6-volt monitor 3
22	6-volt monitor 4
23	5-volt monitor
24	Command Receiver 1 Signal Strength
25	Command Receiver 2 Signal Strength
26	±18-volt monitor (forward)
27	Signal Ground
28	BLH Amplifier 24

17.01 TELEMETRY ALLOCATIONS (continued)

Commutator 6 (T/M 3 - 52.5 Kilohertz) - Forward Temperature

Segment	Function
1	Calibration step 1 - 130 ohms
2	Nose Ogive skin temperature (forward, inside) T 1
3	Nose Ogive skin temperature (middle, inside) T 2
4	Nose Ogive skin temperature (aft, inside) T 3
5	Nose Ogive skin temperature (aft, outside) T 4
6	Forward payload rack strut temperature (forward) T 5
7	Calibration step 2 - 125 ohms
8	Forward payload rack strut temperature (aft) T 6
9	Cylindrical payload extension skin temperature (middle, inside) T 7
10	Sustainer forward skirt (inside, adjacent to shroud II forward cap) T 8
11	Body shroud skin temperature (forward, inside) T 9
12	Helium bottle skin temperature (outside) T 10
13	Calibration step 3 - 100 ohms
14	Regulator section skin temperature (forward, inside) T 11
15	Regulator section skin temperature (aft, inside) T 12
16	Fuel tank skin temperature (forward, outside) T 13

Commutator 6 (T/M 3 - 52.5 Kilohertz) - Forward Temperature (continued)

Segment	Function
17	Fuel tank skin temperature (aft, outside) T 14
18	Sustainer fin skin temperature (mid-chord, forward, inside) T 30
19	Sustainer fin skin temperature (root-chord, forward, outside) T 31
20	Calibration step 4 - 75 ohms
21	Sustainer fin skin temperature (tip-chord, forward, outside) T 32
22	Sustainer fin skin temperature (root-chord, middle, inside) T 33
23	Sustainer fin skin temperature (mid-chord, middle, inside) T 34
24	Sustainer fin skin temperature (mid-chord, aft, inside) T 35
25	Sustainer fin skin temperature (tip-chord, aft, inside) T 36
26	Sustainer fin trailing edge temperature (inboard, under cork insulation) T 37
27	Sustainer fin trailing edge temperature (outboard, under cork insulation) T 38
28	Calibration step 5 - 50 ohms

17.02 TELEMETRY ALLOCATIONS

Telemetry 1 - 256.2 Megahertz (FM/FM)

IRIG Channel	Function
G	Minnesota Experiment Receiver Output
18	Commutator 1 Forward Voltage & Miscellaneous Monitors
17	Commutator 2 Temperature and Miscellaneous Monitors
16	Commutator 3 Aft Voltage Monitors
15	(PcI) Chamber Pressure
14	(Poj-I) Pressure Oxidizer (Injector Manifold)
13	(Pfj-I) Pressure Fuel Injector (upstream of Nike Valve)
12	Pnk Pressure Nike Booster
11	Payload Accelerometer Thrust (Z) (+ Axis Forward)
10	Payload Accelerometer Lateral (Y) (+ Axis toward Fin I)
9	(Pol-I) Line Pressure Gas Bottle (Upstream of Nike Valve)
8	Stable Platform (Roll)
7	Strain Gage 1 (Extension between Fins I and II)
6	Strain Gage 2 (Extension between Fins II and III)
5	Nike Valve Monitor I
4	Minnesota Experiment (Sweep/Bias voltage)

17.02 TELEMETRY ALLOCATIONS (continued)

Telemetry 2 - 244.3 Megahertz (FM/FM)

IRIG Channel	Function
G	Ionosphere Experiment (Course Phase)
18	Ionosphere Experiment (Fine Phase)
17	Ionosphere Experiment (Course Current)
16	Pc-IV Chamber Pressure
15	Pc-II Chamber Pressure
14	(Poj-II) Pressure Oxidizer Injector (Manifold)
13	(Pfj-II) Pressure Fuel Injector (upstream of Nike Valve)
12	(Pfj-IV) Pressure Fuel Injector (upstream of Nike Valve)
11	(Poj-IV) Pressure Oxidizer Injector (Manifold)
10	Payload Accelerometer Lateral (X) (+ Axis toward Fin IV)
9	VSWR Monitor (TM 3)
8	Stable Platform (Pitch)
7	Strain Gage 3 (Extension between Fins III and IV)
6	Strain Gage 4 (Extension between Fins IV and I)
5	Nike Valve Monitor II

17.02 TELEMETRY ALLOCATIONS (continued)

Telemetry 3 - 240.2 Megahertz (FM/FM)

IRIG Channel	Function
18	Solar Aspect (5 sensors)
17	Antenna Bias Voltage (Ionosphere Experiment)
16	Ionosphere Experiment (Course Voltage)
15	Pc-III Chamber Pressure
14	Poj-III Pressure Oxidizer Injector (Manifold)
13	Pfj-III Pressure Fuel Injector (upstream of Nike Valve)
11	Payload Accelerometer Thrust (Z)
10	Pgr Pressure Gas Regulator (Regulator Outlet He Manifold)
9	Pgb Pressure Gas Bottle
8	Stable Platform (Yaw)
6	Nike Valve Monitor IV
5	Nike Valve Monitor III

Telemetry 4 - 231.4 Megahertz (PPM)

IRIG Channel	Function
1	Liquid Level Monitor (Oxidizer)
2	Liquid Level Monitor (Fuel)
3	X-Ray Experiment (3 of 3)
4	X-Ray Experiment (2 of 3)
5	Tco-I (Zone Box 2)
6	Tco-II (Zone Box 3)
7	Tco-III (Zone Box 4)
8	Tco-IV (Zone Box 5)
9	Ogive (Pitch)
10	Ogive (Yaw)
11	Pes II Vehicle Extension Surface Press II (near Fin II line)
12	Pes IV Vehicle Extension Surface Press IV (near Fin IV line)
13	Pei Vehicle Extension Internal Press (Rack)
14	Magnetometer X (In line to Ionosphere Experiment Antennas, + Axis near Fin III line)
15	Magnetometer Y (+ Axis near Fin II line)
16	Magnetometer Z (+ Axis forward)

Commutator 1 (T/M 1 - 70 Kilohertz) - Forward Voltage and Miscellaneous Monitors

Segment	Function
1	Calibration Step 1 (Ground)
2	Calibration Step 2 (+5 volts)
3	Tgb
4	+5 Volts (Ogive)
5	Squib Firing Current Monitor
6	BLH 5 Monitor
7	Command Receiver 1, Channel 4, Monitor
8	Command Receiver 2, Channel 4, Monitor
9	T/M 1, 30 volts
10	T/M 2, 30 volts
11	T/M 3, 30 volts
12	T/M 4, 30 volts
13	±18 volt Monitor
14	Command Receiver 1 Signal Strength
15	Command Receiver 2 Signal Strength
16	BLH 19 Monitor
17	BLH 20 Monitor
18	BLH 21 Monitor
19	Calibration Step 3 (3.0 volts)

Commutator 1 (T/M 1 - 70 Kilohertz) - Forward Voltage and Miscellaneous Monitors (continued)

Segment	Function
20	Gulton Amplifier 1 Monitor
21	Gulton Amplifier 2 Monitor
22	Gulton Amplifier 3 Monitor
23	Gulton Amplifier 4 Monitor
24	BLH 22 Monitor
25	BLH 23 Monitor
26	BLH 9 Monitor
27	Signal Ground
28	Calibration Step 4 (4.0 volts)

Commutator 2 (T/M 1 - 52.5 Kilohertz) - Temperature and Miscellaneous Monitors

Segment	Function
1	Calibration Step 1 (130 ohms)
2	T-1 Ejectable Nose Fairing, Forward
3	T-2 Ejectable Nose Fairing, Aft, Between Fins III and IV
4	T-3 Ejectable Nose Fairing, Aft, Between Fins I and II
5	T-4 Minnesota Probe Circuit Board
6	T-5 On Gamma Ray Detector Collimator
7	Calibration Step 2 (125 ohms)
8	T-6 Solar Aspect Doubler
9	T-7 N/C, Forward, Below Aspect Doubler
10	T-8 N/C, Forward, Between Aspect Doublers
11	T-9 Internal Payload, 1/16" aluminum plate
12	T-10 N/C, Aft
13	Calibration Step 3 (100 ohms)
14	T-11 (Z) Box, Ion Experiment
15	T-12 (Z) Box, Ion Programmer
16	T-13 Payload Extension Skin, Forward
17	T-14 Space General 1 (Extension)
18	Pin Puller/Tip Eject Monitor

17.02 TELEMETRY ALLOCATIONS (continued)

Commutator 2 (T/M 1 - 52.5 Kilohertz) - Temperature and Miscellaneous Monitors (continued)

Segment	Function
19	Antenna Door Eject Monitor
20	Calibration Step 4 (75 ohms)
21	Hayden Timer Monitor
22	(Z) Antenna 1 deploy Monitor
23	(Z) Antenna 2 deploy Monitor
24	T-15 Space General 2 (Extension)
25	T-16 Space General 3 (Extension)
26	T-17 Space General 4 (Extension)
27	T-18 Tank Assembly Forward Skirt
28	Calibration Step 5 (50 ohms)

17.02 TELEMETRY ALLOCATIONS (continued)

Commutator 3 (T/M 1 - 40 Kilohertz) - Aft Voltage Monitors

Segment	Function
1	Calibration Step 1
2	Calibration Step 2
3	BLH 1
4	BLH 2
5	BLH 3
6	BLH 4
7	BLH 15
8	BLH 6
9	BLH 7
10	BLH 8
11	Calibration Step 3
12	BLH 10
13	BLH 11
14	BLH 12
15	BLH 13
16	Spare
17	(Tci-IV) (Gulton 9) Zone Box 6
18	BLH 16
19	BLH 17

17.02 TELEMETRY ALLOCATIONS (continued)

Commutator 3 (T/M 1 - 40 Kilohertz) - Aft Voltage Monitors (continued)

Segment	Function
20	BLH 18
21	Cutoff Valve Position Monitor
22	BLH 14
23	-30 Volts-dc Monitor Ionosphere Experiment
24	(Tci-IV)
25	Spare
26	(Tci-IV)
27	Spare
28	(Tci-IV)

APPENDIX B

TYPICAL INTEGRATION TEST FOR THE AEROBEE 350
17.02 COUNTDOWN

NOTE: Before Nose Cone On; Short Altitude Switches

T-5 hr.	Perform command safety check (Use match squibs or test boxes depending on test) (30 minutes)
T-4 hr.	Perform instrumentation checks - including lift-off checks: 1. Include battery charge test (2 minute charge) Pack 1 2 3 4 5
T-85 min.	1. Command safety console on. 2. Pull S/A arming pin. 3. Start pyro battery charging. (3 amperes for 60 minutes) (VENT EVERY 5 MINUTES)
T-75 min.	Vent
T-70 min.	Vent
T-65 min.	Vent
T-60 min.	Vent
T-55 min.	Vent
T-50 min.	Vent

17.02 COUNTDOWN (Continued)

T-45 min.	Vent
T-40 min.	Vent
T-35 min.	Vent
T-30 min.	Instrumentation "warm up" on. (Record Voltages)
	1. Vent
	2. Record currents:
	I 1
	2
	3
	4
	3. Experiments on Record Currents ("warm up") (Record Voltages)
	I 1
	2
	3
	4
	II Read and record ledex (payload) position
	4. Read and Record Haydon Timer Monitor
T-25 min.	1. Vent pyro batteries
	2. Telemetry ground station ready.

17.02 COUNTDOWN (Continued)

T-25 min.

3. Pyro battery load check:

   (3 amperes for 30 seconds.)

   No Load

   Full Load

T-20 min.

1. Commutators ON
2. Telemeters ON - read power supply voltages and currents:

   T/M 1

   T/M 2

   T/M 3

   T/M 4

   Tip Eject

3. Command safety check

   a. Read and record cutoff valve position.

   b. FRW-2 carrier ON.

   c. Turn on Command Receivers 1 and 2

      Read and record Monitors A and B:

      1) Monitor A

         Ledex 1, Receiver 1

         Ledex 2, Receiver 2

         Ledex 3, Receiver Signal Strength

T-20 min.

Ledex 4, Receiver Signal Strength

Ledex 6, Receiver 1

Ledex 7, Receiver 2

Ledex 8, Receiver 1, Channel 4

Ledex 9, Receiver 1, Channel 4

AN/FRW Transmit Channel 4

Ledex 8, Receiver 1, Channel 4

Ledex 9, Receiver 1, Channel 4

2) Monitor B:

(+) Destruct

(-) Destruct

(+) Cutoff

(-) Cutoff

(+) Pyro Battery

(-) Pyro Battery

4. Ground station confirm RF

T-17 min. Gyro External power ON

T-15 min.
1. Read and record "G" switch position
2. Complete battery charging

T-14 min. Channel 4 OFF

Ground station read and record (T/M 1, 70 kilohertz) position: ______________

T-13 min.	FRW-2 Send Channel 4
T-12 min.	Ground station read and record (T/M 1, 70 KHz) TM segments 7 and 8 (Channel 4 ON)
T-11 min.	FRW-2 Channel 4 OFF
T-10 min.	1. Read and record power supply voltages and currents: T/M 1 T/M 2 T/M 3 T/M 4 2. Read and record Tip Eject Pyro Supply
T-7 min.	Uncage Gyro
T-6 min.	1. Cage Gyro 2. Tape recorders ON 3. Paper recorders ON a. F/M at 1 minute per second b. PPM at 0.2 minute per second
T-4 min.	All systems internal. FRW-2 Send Channel 4 TM Confirm Channel 4 Receiver 1 - Receiver 2 On
30 sec.	T/M calibrate Perform pyro battery "no voltage" checks:

30 sec.	(+) Destruct
	(-) Destruct
	(+) Cutoff
	(-) Cutoff
	(+) Pyro Battery
	(-) Pyro Battery
T-2 min.	1. Read and record instrumentation voltage (current should be zero) T/M 1 T/M 2 T/M 3 T/M 4
	2. Read and record Tip Eject Pyro Supply voltage
	3. Read and record Ledex (payload) position
	4. Remove tail can door and tape switch on Liquid Level Unit to 2 position. (Place door back on tail section)
	5. "Arm" S/A Destruct
T-60 sec.	1. T/M Calibrate
T-50 sec.	"Arm" S/A relay cutoff
T-45 sec.	Read "G" Switch monitor
T-35 sec.	1. T/M Calibrate

17.02 COUNTDOWN (Continued)

T-35 sec.	2. Read and record valve position monitor voltage (Leave in position for continuous monitoring)
T-15 sec.	1. Command safety to launch position
	2. Uncage Gyro
T-5 sec.	Paper recorder to fast speed
	1. F/M 10 minutes per second
	2. PPM 4 minutes per second
000	1. Pullaways (2) out
	2. Control Deck 6 "C" switches (2) by pass
T+3 sec.	Booster pulled away
T+5 sec.	FRW-2 Turn Channel 4 OFF, then ON, five times; ON for rest of Flight
T+10 sec.	Start ogive test: Move ogive toward Fin I from center position
	Hold for 5 seconds.
T+15 sec.	Return ogive to center position. Hold for 5 seconds.
T+20 sec.	Move ogive towards Fin III from center position. Hold for 5 seconds.
T+25 sec.	Return ogive to center position. Hold for 5 seconds.
T+30 sec.	Move ogive towards Fin IV from center position. Hold for 5 seconds.
T+35 sec.	Return ogive to center position. End of ogive test.
T+40 sec.	VSWR Test: Place hands on T/M 3 antennas for 5 seconds.

17.02 COUNTDOWN (Continued

T+45 sec.	VSWR Test completed
T+60 sec.	FRW-2 Give cutoff (Channels 1 and 3) 5 seconds
T+65 sec.	1. Cutoff OFF 2. Liquid Level OFF 3. Tip eject test: Lamp for tip eject ON (timer). (Squibs may be used for this test. If so prepare to catch tip and disconnect pullaway.)
T+70 sec.	1. FRW-2 Give Channels 1 and 5 2. Antenna Door Eject Test: Prepare to catch doors. 3. Antenna Deployment Test: Lamps for antenna deployment ON. CAUTION If squibs are used care must be taken to avoid possible injury to personnel. (This function is armed at this time, T+70 seconds, but circuit should not function until approximately T+80 seconds)
T+75 sec.	FRW-2 Give Destruct Channel 2 ON Release 5—leave Channel 1 ON (Should have Channels 1 and 2 ON) Leave ON for 5 seconds
T+80 sec.	Antennas Deployed (See item 3 in T+70 seconds.)

T+90 sec.	Magnetometer (longitude) test: Move magnet back and forth along the thrust axis for 10 seconds. (Probe located in nose cone)
T+100 sec.	1. Magnetometer (longitude) test complete
	2. Haydon timer monitor indicates end of cycle
T+105 sec.	Magnetometer (roll) test: Move magnet around body of the vehicle near the magnetometer for 10 seconds
T+115 sec.	Magnetometer (roll) test complete
T+120 sec.	Paper recorder to fast speed
T+125 sec.	Solar Aspect Test:
	Remove cover from Eye 1 and illuminated for 10 seconds. Replace cover
T+135 sec.	Remove cover from Eye 2 and illuminate for 10 seconds. Remove cover
T+145 sec.	Remove cover from Eye 3 and illuminate with sun gun for 10 seconds. Replace cover
T+155 sec.	Remove cover from Eye 4 and illuminate with sun gun for 10 seconds. Replace cover
T+165 sec.	Remove cover from Eye 5 and illuminate with sun gun for 10 seconds. Replace covers on Eyes 4 and 5
T+170 sec.	End of Solar Aspect Test.
T+175 sec.	Accelerometer Tap Test:
	Tap forward ring of 30 inch extension in line with Fin I for 10 seconds
T+185 sec.	Tap forward ring of 30 inch extension in line with Fin II for 10 seconds

17.02 COUNTDOWN (Continued)

T+195 sec.	End of accelerometer Tap test.
T+200 sec.	Start Gyro and (1) "G" test:
	Roll vehicle 360 degrees in 40 seconds.
	Paper recorders to slow speed.
T+240 sec.	End of vehicle roll
T+245 sec.	Move vehicle nose, with Fin I down, towards Fin II as far as possible, and hold for 10 seconds
T+255 sec.	Move vehicle nose back to center and hold for 10 seconds
T+265 sec.	Move vehicle nose toward Fin IV and hold for 10 seconds
T+275 sec.	Move vehicle nose back to center and hold for 5 seconds
T+280 sec.	Rotate Fin I to horizontal, hold for 10 seconds
T+290 sec.	Move vehicle nose towards Fin I and hold for 10 seconds
T+300 sec.	Move vehicle nose back to center and hold for 5 seconds
T+305 sec.	Move vehicle towards Fin III and hold for 10 seconds
T+315 sec.	Move vehicle nose back to center. Test completed
T+320 sec.	System test complete. All systems standby
	When all systems ok. (Pullaways IN) then:
	FRW-2 OFF

17.02 COUNTDOWN (Continued)

T+320 sec.	Safe Command System - (Secure)
	Recorders OFF
T+325 sec.	Payload power on External Cage Gyro
T+330 sec.	Standby to run Strain tests: If no tests complete count.
T+335 sec.	Payload to warmup

Additional Test:

Strain Test (8 seconds per position)

Position 1 Fin III down; Gyro uncaged

2 Fin II down

3 Fin I down

4 Fin IV down

5 Fin III down

Deployment using FRW-2 as back-up

Same as T-5 hours to T-1 seconds

Pullaway (2)

T+3 seconds	Booster Pulled away
T+60 sec.	FRW-2 Give Cutoff (Channels 1 and 3)
T+65 sec.	FRW-2 Cutoff OFF
	Give Tip eject Channel 7 for 5 seconds (Tip eject lamp ON) (Command back-up). If squibs are used for this test prepare to catch tip and disconnect pullaway connector for 5 seconds.

T+70 sec.	FRW-2 Give channel 10 for 5 seconds (Antenna Deployment) Lamps for antenna deployment ON. This function is armed at this time T+70 seconds but circuit should not operate until T+80 seconds.
	CAUTION
	If squibs are used care must be taken to avoid possible injury to personnel.
T+80 sec.	Antenna deployed. (See item 2 additional test T+70 seconds).
T+85 sec.	Additional test completed.
T+90 sec.	All Systems standby
T+100 sec.	When all systems OK (pullaways IN) then:
	FRW-2 OFF
	Safe Command
	Recorders OFF
T+105 sec.	Payload power ON external Cage Gyro
T+110 sec.	Standby to run additional tests; if no tests complete count
T+115 sec.	Payload to warmup
T+120 sec.	Experiments OFF
T+125 sec.	All power OFF