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by J. M. Turkiewicz, C. F. Fuechsel, R. G. Martin, F. D. Piazza, and V. L. Krueger

Goddard Space Flight Center
Greenbelt, Md.

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SUMMARY

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The first International Ionosphere Satellite was launched on 26 April 1962, and carried experiments designed to increase knowledge of the ionosphere and its complex relation to the sun. The on-board experiments included ion and electron studies by probes, an electron density experiment, measurements of Lyman-alpha and x-ray solar radiation, and cosmic ray measurements. The program was a joint United States—United Kingdom project. This report describes the spacecraft, its various subsystems, and the techniques used in integration. An evaluation of the performance of the various systems during testing and launching is also given.

Author

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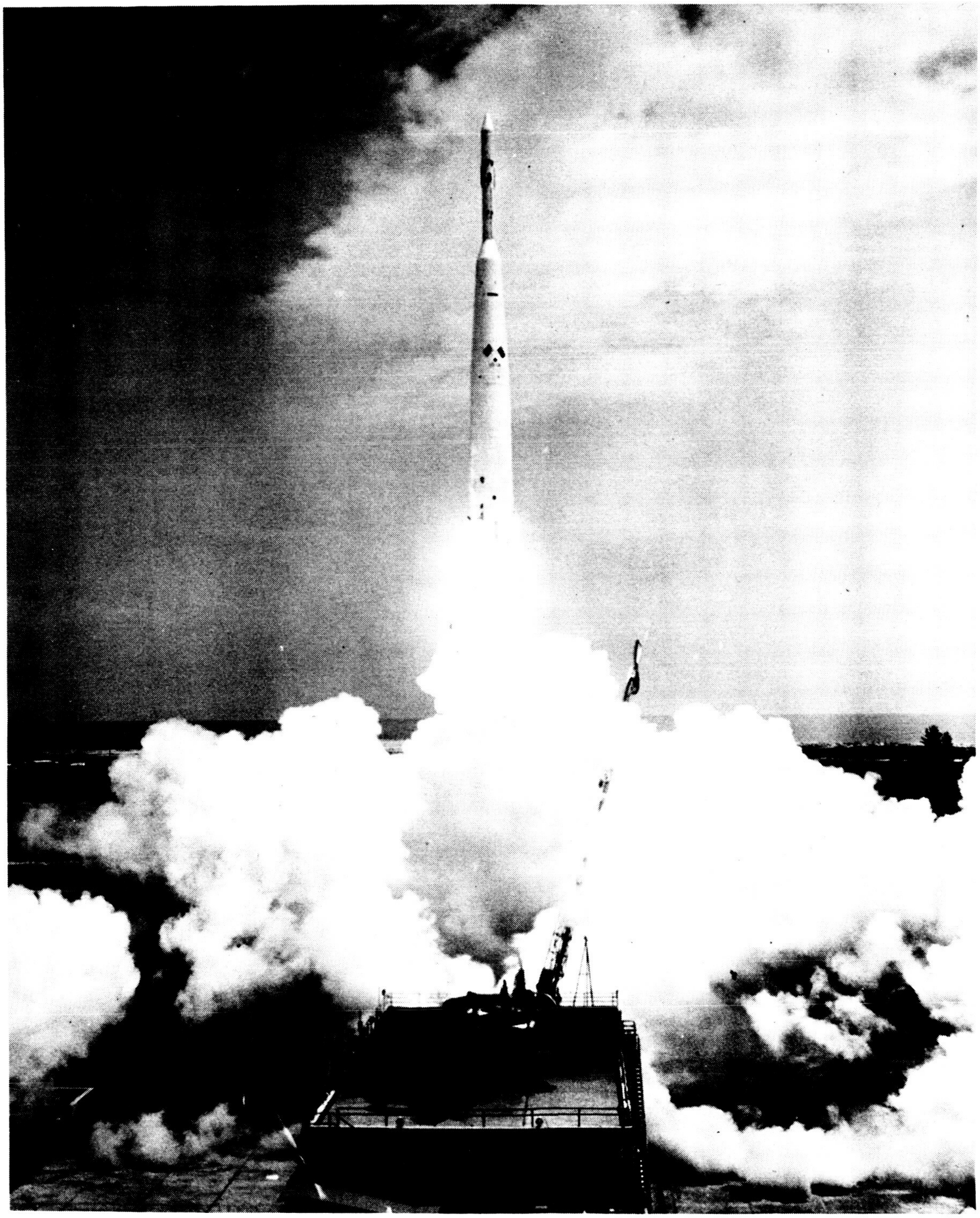

Frontispiece—Launching of the International Ionosphere Satellite.

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INTRODUCTION

The first International Ionosphere Satellite was launched on 26 April 1962. It carried experiments that were designed to increase knowledge of the ionosphere and its complex relation to the sun. The experiments carried on board are as follows: 1) ion and electron studies by probes, 2) an electron density experiment, 3) measurements of solar radiation, both x-ray and Lyman-alpha, and 4) cosmic ray measurements.

The program was a joint United States—United Kingdom project. United States responsibilities included the design, fabrication, and testing of the satellite structure, power supply, telemeter, command receiver, temperature control and data storage (NASA-GSFC), and necessary launch facilities (NASA-AMR). The United Kingdom responsibilities included the design, fabrication, and testing of all experiments; data analysis and interpretation. The United States and United Kingdom shared joint responsibility for launching, tracking, telemetering data, data handling and data processing.

From a system standpoint, the experiments and support electronics (such as encoders, power supplies, transmitter, etc.) are electronic subsystems which must be combined to form a complete electronic system. This combination process, called integration, is the subject of this report.

The integration procedures used on the spacecraft employed techniques, methods and approaches that were developed and used in previous spacecraft programs. New methods in data recording and reduction developed for the integration of the UK-1 spacecraft included use of an optical recording oscillograph for recording, simultaneously, up to 24 independent channels of information. Also, a special test stand data reduction system was built.

List of Experiments

The experiments were the responsibility of the United Kingdom; the electronic subsystems were the responsibility of Goddard Space Flight Center. Physical location of some of the experiments and electronic subsystems can be seen in Figure 1. The functional relationships of the experiments and other subsystems of the satellite are shown in Figure 5.

Electron Temperature and Density (University College of London). This experiment, based on Druyvesteyn's modification of the Langmuir probe, determines the value of the electron temperature and density near the satellite. There are two of these experiments onboard. The detector for one of these experiments is identified in Figure 1 as "Electron Temperature Probe." This boom-mounted detector looks at electrons in the region above the probe and approximately 4 feet away from the satellite. The second detector is mounted below the tape recorder and looks in a direction directly opposite to that of the boom-mounted probe.

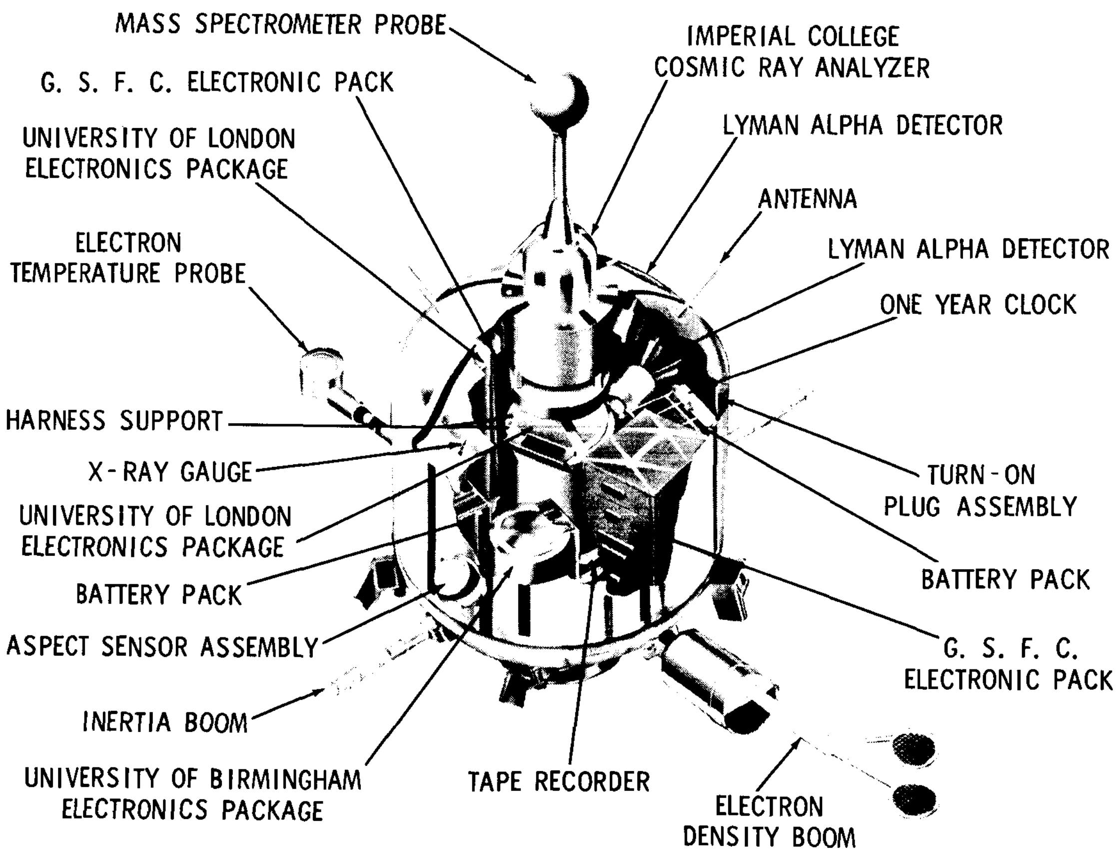

Figure 1—International ionosphere satellite S-51/UK-1.

Ion Mass, Composition, and Temperature (University College of London). This experiment utilizes electronic circuits essentially the same as those used in the electron temperature and density experiment. The detector consists of two spheres (concentric, hollow, perforated metal) which are supported by a shaft attached to the cosmic ray experiment on the top cover of the spacecraft. It is identified in Figure 1 as "Mass Spectrometer Probe."

X-Ray Emission (University College of London). This experiment provides an indication of solar conditions by counting the number of x-rays emitted from the sun. Two detectors, one on the upper satellite cover, identified as "X-Ray Gauge" in Figure 1, and one mounted on the lower cover in a corresponding position, provide almost complete coverage of space in each revolution of the spacecraft.

Solar Aspect Measurement (University College of London). The solar aspect subsystem provides the satellite spin rate, and the latitude and longitude of the sun in the satellite coordinate system. The detector consists of four silicon solar cells mounted in a pyramid configuration under the cover identified as "Aspect Sensor Assembly" in Figure 1.

Cosmic Ray Analyzer (Imperial College). The purpose of this experiment is to make accurate measurements of the primary cosmic ray energy spectrum and the effects of interplanetary magnetic field modulation on this spectrum. A Cerenkov scintillator, located under the dome shaped cover identified as "Imperial College Cosmic Ray Analyzer" in Figure 1, is the detector for this experiment.

Ionosphere Electron Density Measurement (University of Birmingham). The density of electrons in the ionosphere is measured by the change in capacitance between the two circular grids mounted on the boom which is labeled "Electron Density Boom" in Figure 1. Since this experiment also measures electron temperature as a secondary function, it measures the same quantities as the electron temperature and density experiment, but it does so by a different measuring technique. The two experiments therefore complement each other.

List of Electronic Subsystems

Telemetry - The telemetry system uses pulse frequency modulation on a phase modulated carrier.

Data Encoders - Data encoding is done by 2 encoders, a real time high speed encoder which completes a scan of 256 channels every 5.12 seconds, and a storage data low speed encoder which completes a scan of 32 channels every 30.72 seconds.

Tape Recorder - Data storage is accomplished by a single channel magnetic tape recorder in a hermetically sealed pressurized container 7 inches in diameter and 2-1/2 inches high.

Power System - The primary power source is a set of 4 solar paddles. They operate the spacecraft and keep the storage batteries charged.

Spacecraft Parameters - Four of the 256 telemetry channels of the high speed encoder provide information on internal spacecraft conditions such as temperatures and voltages.

GENERAL DESCRIPTION OF SATELLITE

The S-51/UK-1 International Ionosphere Satellite is shown in Figure 1. The satellite was placed in orbit by a Delta launch vehicle on 26 April 1962. The satellite has an elliptical orbit extending from a perigee of 200 statute miles to an apogee of 600 statute miles, with an inclination of 55 ± 3 degrees. The single-orbit period is 100 minutes.

The satellite is 23 inches in diameter, and weighs approximately 150 lbs. The experiment booms extend 48 inches from the body of the satellite. Real time data and recorded data are transmitted to ground stations. Data transmission will be discontinued at the end of one year by an electrochemical timer in the satellite.

A typical electronic subsystem of the satellite is a printed circuit card which is completely encapsulated in foam. Inside the encapsulation material are electronic components which, depending on circuit, space, and weight requirements, are mounted in several ways. Figure 2 shows a view of an electronic printed circuit card with potted modules installed, prior to complete encapsulation of the card.

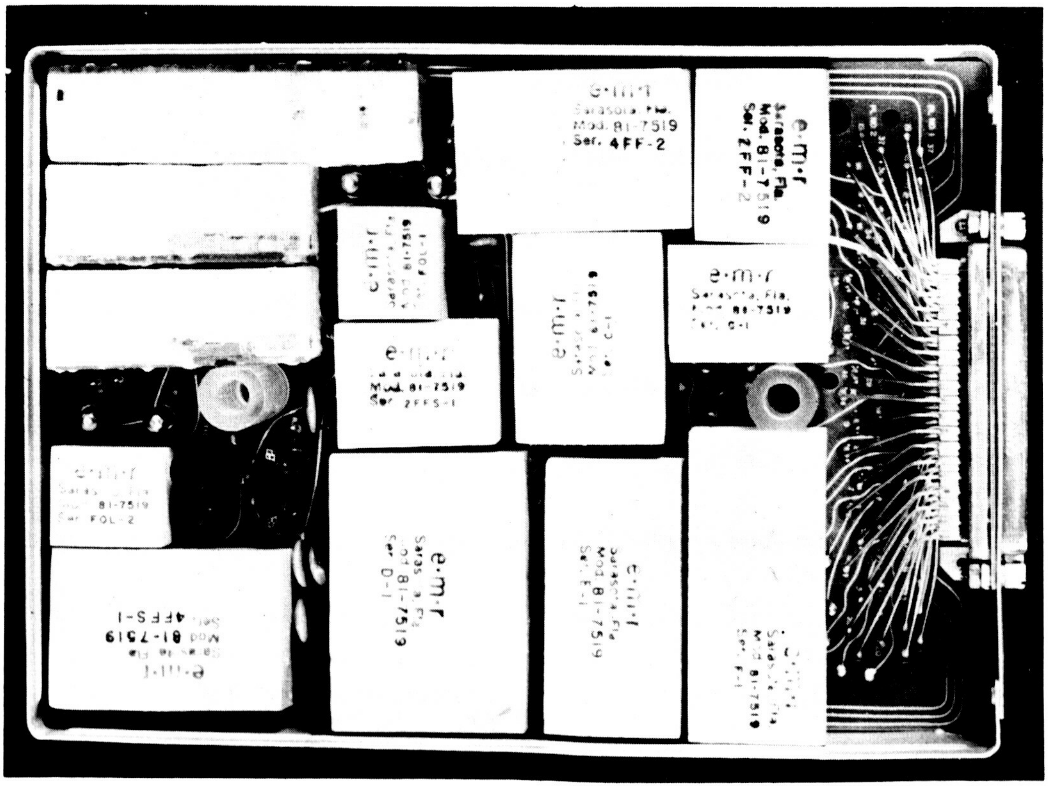

Figure 2—Printed circuit card, high component density, modules encapsulated.

DESCRIPTION OF ELECTRONIC SYSTEM

Block Diagram Discussion

A block diagram of the spacecraft's electronic system is shown in Figure 3. Eight probes are shown with their data outputs processed by signal conditioning circuits. The outputs of the signal conditioning circuits are connected to subcarrier oscillators (analog and digital). The subcarrier oscillators convert analog and digital data to representative ac signals. The frequencies of these signals are dependent upon the value of the input analog voltage and/or digital data, Figure 4. The outputs from the subcarrier oscillators are sequentially gated as high speed (HS) or low speed (LS) data outputs by the respective encoder matrices. The encoder assignments of experimental data outputs are listed in Table 1.

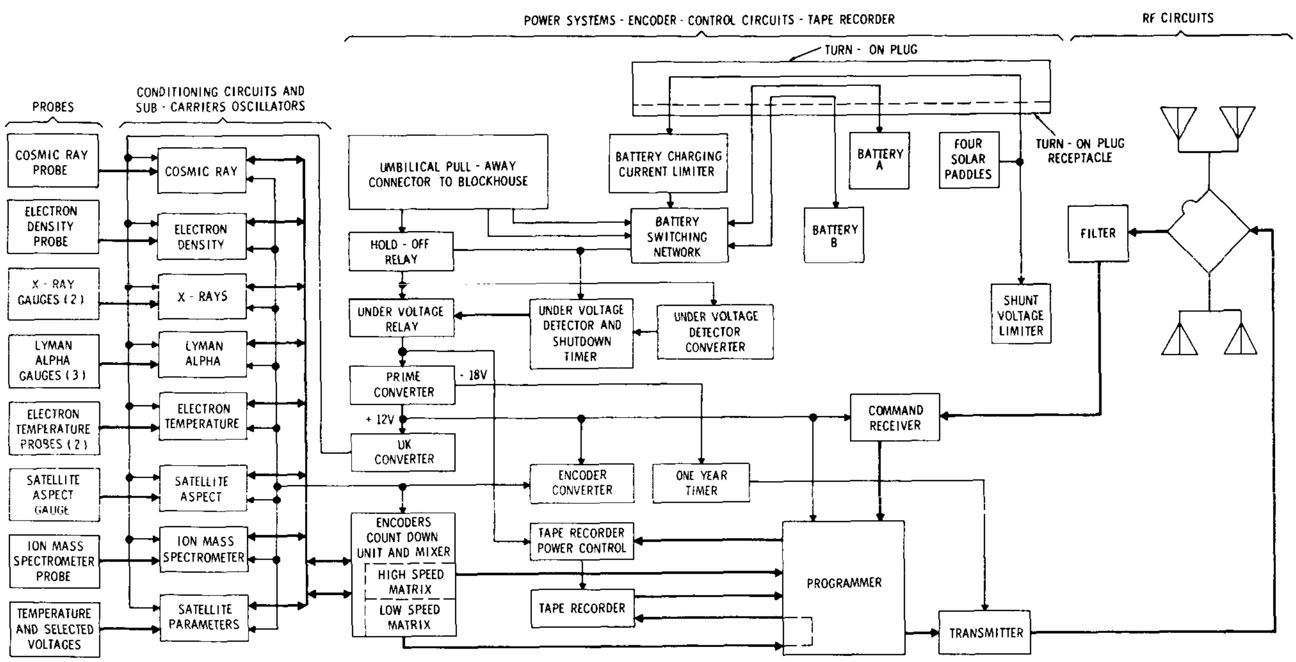

Figure 3—Electronic system block diagram.

The *high speed (HS) encoder* data are transmitted in real time, whereas the *low speed (LS) encoder* data are recorded on magnetic tape. The speed ratio between the HS and LS encoders is 48:1. By playing back the recorded LS encoder data 48 times faster than it was recorded, the low speed data are transmitted at the same bandwidth as the high speed data. The output format of the high speed and low speed encoders is shown in Figure 5; also shown is the telemeter program for almost two complete orbits.

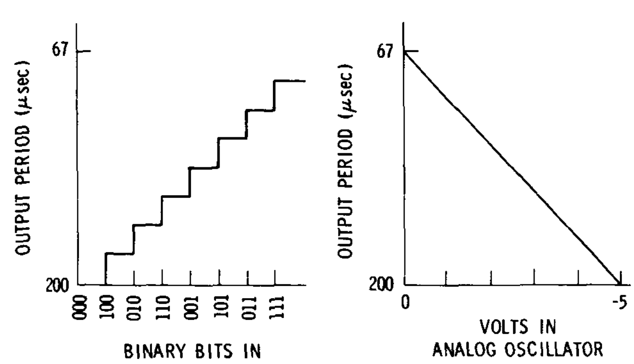

Figure 4—Outputs of digital and analog oscillators.

Table 1

Encoder Input Data.

Experiment	Nomenclature of each output	Experiment	Nomenclature of each output
Electron Temperature #1	U_1 - HS	X-Ray Counter	X_1 - HS
	U_2 - HS		X_2 - HS
	U_3 - HS		X_3 - HS
	U_m - HS		X_4 - HS
	U_1 - LS		X_5 - HS
	U_2 - LS		X_{m1} - HS
	U_3 - LS		X_{m2} - HS
			X_1 - LS
			X_{m1} - LS
Electron Temperature #2	T_1 - HS		
	T_2 - HS	Cosmic Ray	C_1 - HS
	T_3 - HS		C_2 - HS
	T_m - HS		C_3 - HS
	T_1 - LS		C_4 - HS
	T_2 - LS		C_5 - HS
	T_3 - LS		C_6 - HS
			C_1 - LS
			C_2 - LS
Mass Spectrometer	I_1 - HS		C_3 - LS
	I_2 - HS		C_4 - LS
	I_3 - HS		C_5 - LS
	I_{m1} - HS		C_6 - LS
	I_{m2} - HS		
		Electron Density	E_1 - HS
			E_2 - HS
Lyman-Alpha	L - HS		E_3 - HS
	L - LS		E_4 - HS
			E_{m1} - HS
			E_{m2} - HS
Aspect	A_1 - HS		E_{m3} - HS
	A_2 - HS		E_{m4} - HS
	A_3 - HS		E_1 - LS
	A_3 - LS		E_2 - LS
			E_3 - LS
			E_4 - LS
Performance Parameters	P - HS		E_{m1} - LS
	PV_1 - HS		E_{m2} - LS
	PV_2 - HS		E_{m3} - LS
	Pt_1 - HS		E_{m4} - LS

Playback of the *magnetic tape recorder* is initiated by a ground station command to the command receiver in the satellite. The command receiver in turn commands the programmer to switch the tape recorder from record to playback.

The *power system* for the spacecraft consists of four solar paddles, 2 battery packs, shunt voltage limiter, battery-charging current limiter, battery-switching network, and undervoltage detector system.

The four *solar paddles* are mounted with silicon solar cells and together are capable of providing 0.5 to 2 amperes at 15 volts, depending on the spacecraft aspect to the sun. The *shunt voltage limiter* regulates the solar paddle output voltage to 14.5 volts. The *battery charging current limiter* regulates the battery charging current to a value not in excess of 0.5 ampere. The *battery switching network* selects the battery with the highest voltage to operate the spacecraft.

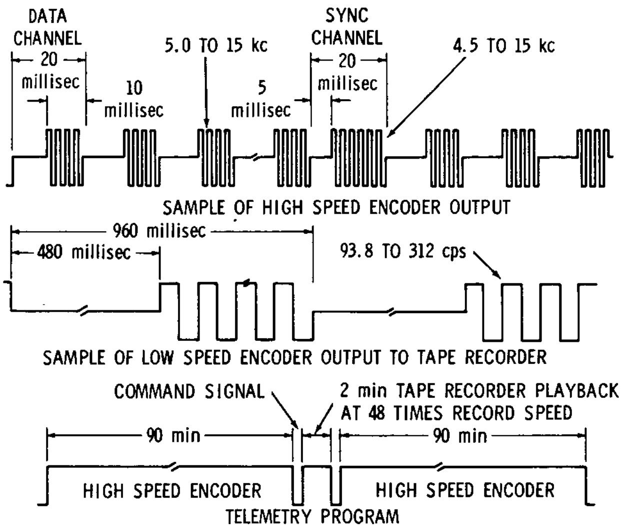

Figure 5—Telemetry program.

The *hold-off relay* provides a means of turning the satellite electronic system on and off after the turn-on plug has been installed in the satellite. (See Figure 3.) It is normally operated from the blockhouse after the satellite has been mounted on the launching vehicle on the launch pad. Since it is in the satellite in the normally on condition, it must be continuously energized for the satellite to remain off after the turn-on plug is in place. If it is not energized, the satellite electronic system is on, this being the desired satellite condition during and after launch.

The *undervoltage detector system* includes an undervoltage detector, converter, relay and shutdown timer. Operation of this system turns off the entire spacecraft electronic system to provide maximum charging current to the batteries. The several *converters* are dc-to-dc converters which supply regulated voltages to the spacecraft's electronic subsystems.

Encoder

The encoder (Figure 6) is operated by a crystal controlled clock. The clock frequency is divided to produce a 50-cps signal for the HS encoder data rate. The 50 cps is divided by 48 to produce the 50/48-cps LS encoder data rate.

The HS encoder format consists of 256 channels arranged in 16 frames (16 channels per frame). A complete set of 256 channels is referred to as an HS encoder sequence. The LS encoder format contains two frames, also with 16 channels per frame. A complete set of 32 channels is referred to as an LS encoder sequence. The channel allocation for both formats is given in Figures 7 and 8. These figures may be compared with the notations on Table 1 for identification of channels.

Operation of the HS encoder is independent of the operation of the LS encoder for increased reliability. However, a loose type of synchronization between the two encoders is provided. Synchronization is defined as being achieved when both encoders start their respective sequences within 20 milliseconds of each other. If the time difference is greater than 20 milliseconds, a "blipper" circuit shortens the LS sequence by 20 milliseconds, causing the next LS encoder

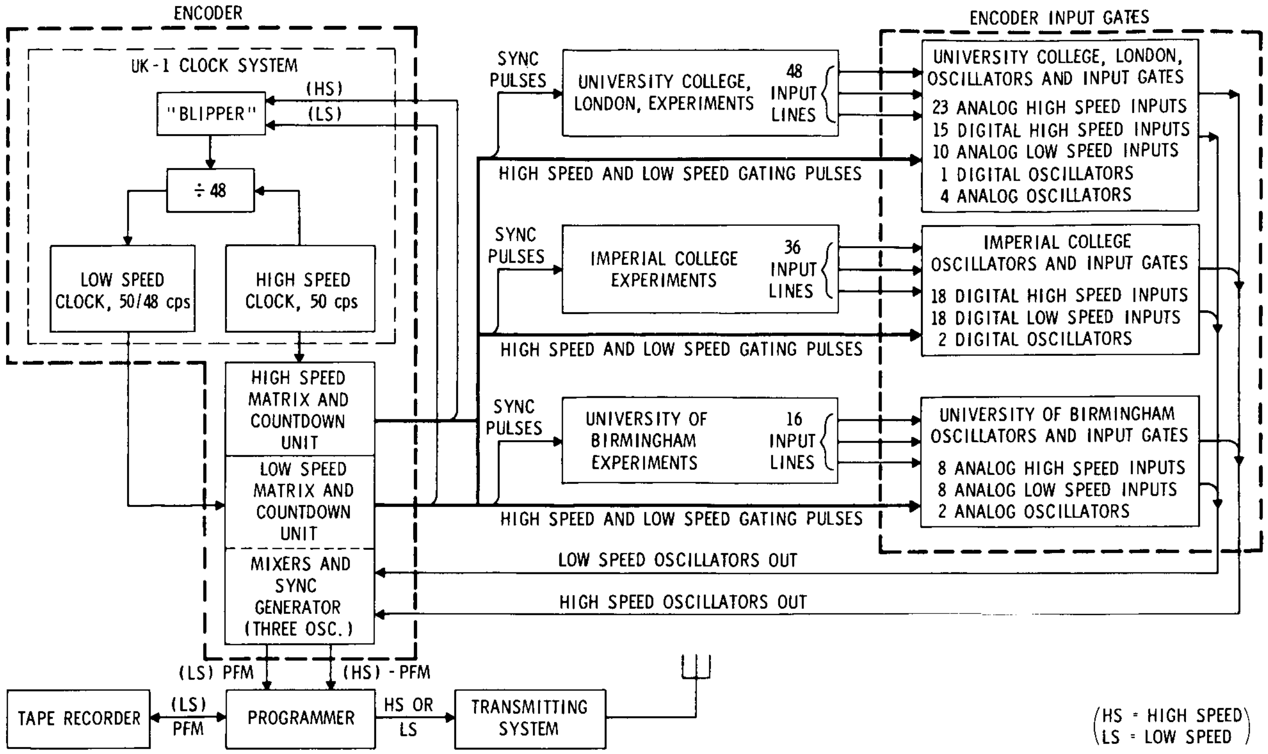

Figure 6—PFM encoder block diagram.

CHANNEL

FRAME	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
0	S	C_1	C_2	C_3	E_3	E_4	E_1	E_2	T_1	T_2	U_1	U_2	I_1	I_2	E_1	E_2
1		C_4	C_5	C_6												
2		T_3	L	E_{m1}												
3		T_m	A_1	E_{m2}												
4		U_3	A_2	E_{m3}												
5		U_m	P_{v2}	P												
6		I_3	X_{m1}	X_{m2}												
7		I_{m1}	I_{m2}	P_{v1}												
8		C_1	C_2	C_3												
9		C_4	C_5	C_6												
10		T_3	L	E_{m1}												
11		T_m	A_1	E_{m2}												
12		U_3	A_3	E_{m4}												
13		P	P_{t1}	P												
14		I_3	X_1	X_2												
15		X_3	X_4	X_5												

Figure 7—High speed encoder telemetry channel allocation.

CHANNEL

FRAME	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
0	S	X_1	X_{m1}	E_4	E_1	E_2	E_3	X_1	C_1	C_2	C_3	C_4	X_1	A_3	C_5	C_6
1	S	X_1	X_{m1}	E_{m4}	E_{m1}	E_{m2}	E_{m3}	X_1	T_1	T_2	T_3	L	X_1	U_1	U_2	U_3

Figure 8—Low speed encoder telemetry channel allocation.

sequence to begin 20 milliseconds earlier. This "blipper" operates once for each LS encoder sequence until synchronization is achieved between the two encoders.

Programmer and Tape Recorder

The main function of the programmer is to control the transmission of LS and HS encoder data to the ground station. A block diagram of the programmer is shown in Figure 9.

To transmit the data recorded by the tape recorder, an rf command is sent to the satellite from a ground station. The command receiver sends a pulse to the programmer, at which time the programmer: (1) disconnects the HS encoder from the transmitter, (2) disconnects the tape recorder input from the LS encoder, and (3) gates a 320.83 cps signal for 2 seconds to the transmitter and tape recorder. The 2 seconds of 320.83 cps is thus simultaneously recorded on the tape recorder and transmitted to the ground station as a signal that the rf command was successful and that tape recorder playback follows immediately.

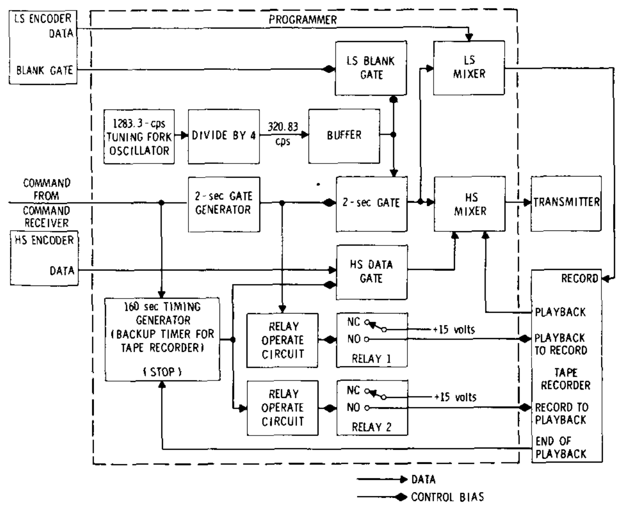

Figure 9—Programmer block diagram.

At the end of the 2 seconds of 320.83 cps the programmer: (1) switches the tape recorder from record to playback and simultaneously shifts the tape speed to 48 times its record speed, and (2) connects the tape recorder playback output to the transmitter.

Transmission of tape recorder playback for a period of 125 to 134 seconds is controlled by a timer in the tape recorder. This period of time is sufficient to play back all stored data. This includes a 15.4-kc pulse, approximately 42 milliseconds long, which is the 2 seconds of 320.83 cps recorded immediately prior to tape recorder playback. This pulse indicates the end of LS encoder data. It also serves as a time reference for correlating data from both encoders.

An electronic back-up timer in the programmer generates a 140 second timing signal which is used to switch the tape recorder from playback to record. This signal is used by the recorder in the event that the tape recorder mechanical playback timer fails.

Figure 10 is a block diagram of the tape recorder. In the record mode, the tape speed is 0.25 in./sec; in the playback mode, the tape speed is 12 in./sec; power consumption over a temperature range of -20°C. to +30°C in record mode is 0.5 watt, and 0.75 watt in the playback mode. Playback flutter is less than 1% p-p from dc to 200 cps; dynamic signal to noise ratio is a minimum of 30db.

Power to the tape recorder drive motor is provided through a dc control circuit. This control circuit removes power from the tape recorder drive motor in the event of a power overload greater than 300 milliamperes lasting for 70 seconds. The dc control circuit receives a reset signal each time the programmer switches the tape recorder from record to playback. This provides a means of testing whether the overload condition is continuous.

Command Receiver

The receiver is basically a double-superheterodyne unit with an amplitude-modulation detector. A decoder, which follows the detector, contains a tone filter, a signal integrating circuit and

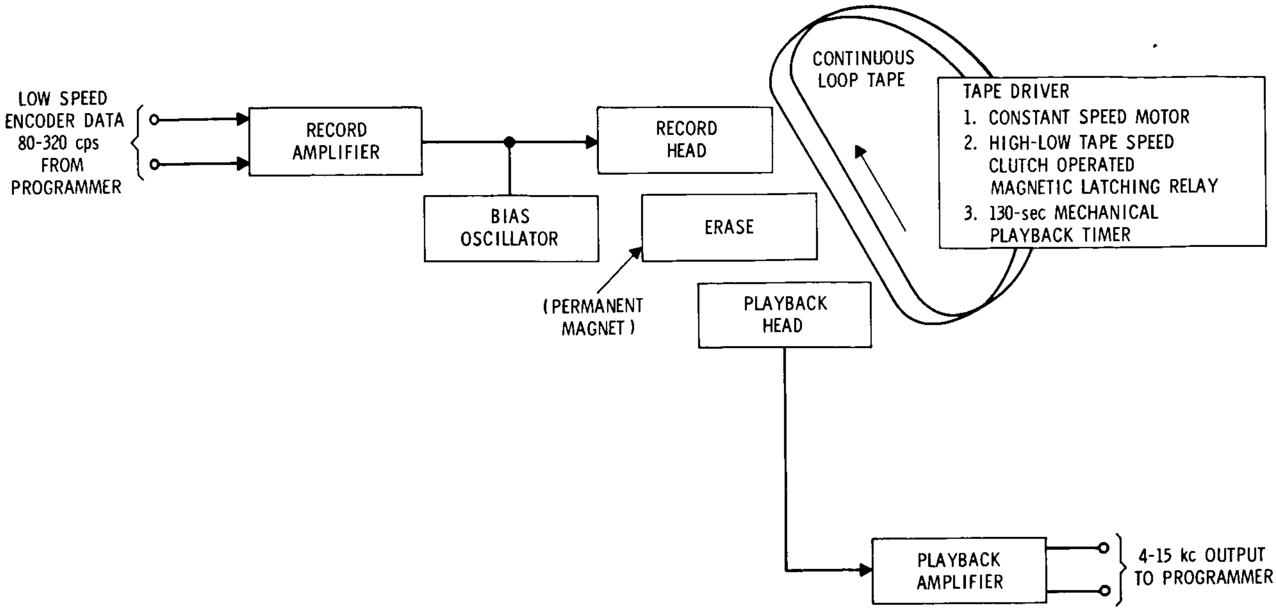

Figure 10—Tape recorder block diagram.

an output trigger circuit to signal the programmer to switch the tape recorder from record to playback. The subcarrier tone that amplitude modulates the received signal must be present for 0.1-second minimum, and the carrier must be modulated at least 30% before the decoder will command. The receiver is powered from the +12-volt line and requires 4.0 milliamperes of current. The receiver sensitivity is -100 dbm or greater.

Transmitter

The phase modulation transmitter is transistorized and delivers 260 milliwatts of power to the antenna system at a frequency of 136.410 megacycles. Frequency stability is ±0.002% or less over a temperature range of -20°C to +60°C. Peak deviation from the square wave modulation is ±1 radian, with less than 2.5 cps incidental frequency modulation and less than 3% amplitude modulation. Harmonic frequencies are 60 db below the carrier power. Total power drain from the -18-volt power supply is about 900 milliwatts. The transmitter consists of a crystal-controlled oscillator operating at a frequency of 68.205 megacycles; a buffer stage to isolate the oscillator from the phase modulator; a phase modulator that employs voltage variable capacitors to shift the phase of the carrier (the instant phase excursions are controlled by the LS and HS encoder output data); a doubler stage which multiplies the output frequency of the modulator to 136.410 megacycles/second; a driver stage that amplifies the doubler output power to about 60 milliwatts; a final amplifier that delivers 260 milliwatts to an antenna load impedance of 50 ohms and; a harmonic filter to prevent radiation of unwanted harmonics of the carrier frequency.

RF Antennas

The spacecraft rf antenna system includes a coaxial hybrid power divider, coaxial phasing lines and a canted turnstile antenna. The hybrid power divider and filter provides about 20-db isolation between the command receiver and the transmitter. The antennas are driven from the base, each pair acting as a dipole, with the dipole pairs in phase quadrature. Radiation in the plane of the turnstile is essentially linear, while circular polarization is obtained along the spin axis. The total power radiated varies with satellite aspect, from +2 db to -4 db, relative to an isotropic radiator. Spacecraft commands are received through the same antenna system and have about the same pattern, but see an additional 2-db loss, because of mismatch at the command frequency.

Power System

Power to the spacecraft electronics is provided by a solar cell array and two battery packs. Power control and regulator circuits include a shunt voltage limiter, a battery charging current limiter, a battery switching network, an undervoltage detector, a hold-off relay and turn-on plug, and several dc-to-dc converters. A block diagram of the power system is shown in Figure 3.

Solar-Cell Array. The solar-cell array consists of four solar paddles arranged in a series-parallel matrix and furnishes 0.5 to 2 amperes at 15 volts, depending on the spacecraft aspect to the sun. The solar cells are p-n type silicon cells that perform as photoelectric converters. The cells are flat mounted, and of gridded construction and exhibit a high efficiency. While the spacecraft is orbiting in sunlight, the solar cells power all electronic subsystems onboard the spacecraft and supply a charging current to the two battery packs.

Battery Packs. Each battery pack consists of ten individual cells connected in series. The cells are nickel-cadmium, sintered-plate type, hermetically sealed in a stainless steel case. Each cell has a nominal potential of 1.3 volts. All ten cells have a nominal output of 13 volts.

Shunt Voltage Limiter. The shunt voltage limiter regulates the solar paddle output voltage to 14.5 volts. Excess power from the solar paddles is dumped through a pair of power resistors located on the arms that extend the solar paddles; the dumping is controlled by a pair of power control transistors.

Battery Charging Current Limiter. The battery charging current limiter regulates the battery charging current to a value not in excess of 0.5 ampere. This circuit prevents the generation of hydrogen in the batteries due to excessive charging currents.

Battery Switching Network. The battery switching network connects the battery with the highest voltage to operate the spacecraft. The other battery is connected to the solar-cell array and receives a trickle charge from the array. When the voltage difference between the two batteries exceeds 1.2 volts, the stand-by battery is connected to operate the spacecraft.

Undervoltage Detector. The undervoltage detector disconnects both battery packs from the spacecraft electronics when the output voltage of both batteries falls below the minimum acceptable level. The detector also activates a recycling timer which, after 18 hours, connects the battery output to the spacecraft electronics. During the 18 hours, the batteries are connected to the solar cell array and are charged. At the end of the 18-hour charging period, the batteries are connected to the spacecraft electronics. If, after the 18-hour charging period, the batteries are still below the minimum acceptable level, the charging cycle is repeated.

Hold-Off Relay. The hold-off relay is controlled from the block house and provides for turning the spacecraft power system on and off when the turn-on plug has been installed in the spacecraft. When the spacecraft is launched, the hold-off relay connects the battery to operate the spacecraft.

DC-to-DC Converters. The dc-to-dc converters supply the different dc voltage levels required to operate the spacecraft electronics. There are four converters in the power system: the prime converter, encoder converter, UK converter, and undervoltage detector converter.

The prime converter is connected to the main power line at the output of the undervoltage relay. The outputs of this converter are +12 volts and -18 volts, regulated to ±1% at 80% efficiency. This converter supplies power to the transmitter, command receiver, programmer, encoder converter, and UK converter.

The encoder converter furnishes the +1.9 volts, -4.0 volts, -6.2 volts, -2.7 volts, and +6.7 volts to the encoder. The -2.7 volts are regulated to ±0.25%, all the remaining voltages are regulated to ±5%. This converter has an efficiency of about 30%.

The UK converter supplies all power to the experiments. The output voltages are, +6.5 volts, +15 volts, -6.5 volts, all regulated to ±5%; -9.0 volts regulated to ±10%; 12 volts regulated to ±8%; -15 volts regulated to ±7%; and 24 volts regulated to ±8%. The overall efficiency is about 60%.

The undervoltage detector circuit converter is connected to the main power line; this converter furnishes 15 volts and -18 volts, both regulated to ±5% and at an efficiency of approximately 50%. This converter is disconnected from the batteries only when the hold-off relay is energized.

One-Year Timer. A one-year timer incorporated in the spacecraft removes power from the transmitter at the end of one year. Two timers are used in a parallel redundant hookup. The timers employ an electrochemical deplating process which has a timing accuracy of plus or minus ten percent (10%).

Orbital Injection Programmer

The orbital injection programmer consists of a battery operated electronic timing system. Its function is to program the events associated with the ignition and separation of the third stage. The orbital injection programmer is activated when the third stage motor is ignited. At this time two pressure switches provide signals to start a 900-second timer. At the end of this timing period two pulses are applied to the silicon controlled rectifier circuit which in turn fires one pair of squibs. This time is designated t_1. The t_1 output from the 900-second timer starts a 60-second timer. At intervals of 60 seconds from time t_1, pulses are provided to the silicon controlled rectifier circuit to fire three additional pairs of squibs. The times of these firings are designated t_2, t_3 and t_4. The orbital injection programmer is redundant throughout, and is operated from two separate batteries. Failure of either programmer will not prevent the squib firing sequence from being completed. Accuracy of the timing cycles is 5% or better. The events at the squib firing times are: despin (t_1), boom release (t_2), paddle release (t_3), and separation (t_4).

INTEGRATING THE SUBSYSTEMS

Electronic Integration Task

The task of integrating the subsystem included the molding of the subsystems into a compatible system, testing the system and evaluating the results, solving interface problems, and monitoring the spacecraft's signals until it was launched. It was thus necessary for project personnel to: (1) have a thorough knowledge of all subsystems (mechanical and electronic); (2) design and fabricate a wiring harness; (3) design and fabricate a test stand for evaluating the performance of the spacecraft's electronics; (4) originate test procedures for each electronic subsystem, as well as the integrated system.

Electronic Integration Plan

The step-by-step plan for the integration of the spacecraft subsystems was as follows: (1) To collect design and performance specifications for all electronic subsystems of the spacecraft. (2) To evaluate the subsystem specifications with the dual purpose of gaining knowledge of the operation of the subsystems, and to design a data-processing system capable of monitoring the performance of all the electronic subsystems as an integrated electronic system within a reasonable time of 1 to 3 hours. (3) To obtain electronic subsystem wiring information for building a spacecraft wiring and test harness. (4) To build a power-control panel to be used with the spacecraft wiring and test harness for integration and interference testing. (5) To design and build test panels for bench testing the electronic subsystems as required. (6) To work out with the mechanical engineers, who are assembling the spacecraft, any problems that may arise from wiring requirements; e.g., accessibility of connectors, holes in the spacecraft structure for routing of wires, supports and tie-down fixtures for the wires. (7) To write a test procedure for the complete electronic system of the spacecraft from information and experience gained with electronic subsystems as they operate individually on the test bench. (8) To detect and assist in resolving problems due to electrical interference or incompatibilities between electronic subsystems as they are connected into the spacecraft. (9) To arrange for supply of the exciters for the experiment sensors. (10) To arrange for the necessary wiring connections through the walls of the environmental test chambers. (11) To monitor the spacecraft performance as it undergoes all phases of the integration testing, environmental testing, and launch-site testing. (12) To report to the project coordinator and the subsystem designers all failures observed in an electronic subsystem during the three test phases.

Electronic Integration History

January 1961. The integration group was formed, beginning with one engineer and one technician. After reviewing the integration schedule and determining the integration tasks, orders were placed for cable connectors, wire-crimping tools, polyvinyl-chloride insulated wire, and rack-mounted test equipment. A conference was held with the cable connector manufacturer to discuss the magnetic characteristics of the connectors.

A standard wiring-list format was developed to speed up the integration process. A copy of the list was forwarded to each experimenter in the United Kingdom.

Three complete sets of test equipment were ordered; one set for the use of the experimenters during the prototype spacecraft integration, and the other sets for the integration of the prototype and flight unit 1. In reality, the experimenters required the constant use of three or four sets of test equipment throughout most of the integration program. Temperature chambers were not included in the initial order of test equipment; however, three were borrowed and were frequently used by the experimenters. Another factor which was not taken into account in the original purchase order of test equipment was the number of pieces of equipment required in building the two test stand telemeter data reduction systems.

February 1961. During February several meethings were held to discuss the format of decoded and decommutated telemetered data, and the decoding and decommutating equipment. Some of the problems resolved during these meetings included the format of decoded and decommutated data during prelaunch go-no-go tests of the spacecraft electronic system, and the test-stand equipment needed for reducing the telemetered data received from the spacecraft. During one of these meetings, it was decided that Goddard Space Flight Center would furnish all the special decommutating equipment.

The spacecraft functional diagram was reviewed and a schedule made for obtaining the subsystem schematics, block diagrams and wiring diagrams. During the review of the functional diagram, the question of battery protection in the event of a battery undervoltage was raised. This matter was resolved at a later date by incorporating a battery undervoltage protection circuit into the spacecraft's power system.

The first returns of the wire-test data sheets, mailed to the experimenters last month, were received.

March 1961. A method for cataloging experiment output data was proposed by the integration engineer and approved by the United Kingdom experimenters.

Preliminary specifications and requirements were established for data handling and processing during environmental testing and prelaunch checkout of the spacecraft. These requirements and specifications were given to the engineer who was assigned the responsibility for the design of the data-reduction test stand. A high-speed, in-line printer (IBM 407)was considered for printing the decoded data received from the spacecraft. After a lengthy discussion, it was decided that special capabilities (ability to store binary data from selected channels and operate on it to produce a decimal equivalent) were more desirable than a high volume of print-outs.

A discussion with the experimenters on the type of triaxial and coaxial connectors to be used with the wire harness led to a decision whereby, the United States would supply all connectors except for the special high-voltage connectors which would be supplied by the United Kingdom.

April 1961. During the month of April, the preliminary requirements for the telemeter decoding and decommutating equipment were forwarded to the Goddard Space Flight Center (see Appendix A).

Wooden mock-ups of the experiment electronic packages were built and mounted in the spacecraft. Using the wooden mock-ups as a guide, construction of the wire harness was started.

A method of turning off the power system in the event of an undervoltage condition was proposed and submitted to the United Kingdom experimenters for comment.

May 1961. The power control panel for monitoring all power supply line voltages and currents was completed and checked out during this month. Also, the automatic voltage and current recording system to be used with the power control panel was checked for its compatibility with the control panel.

A bench checkout procedure for the cosmic-ray experiment was submitted to the experimenter for review. This procedure, after being approved, was circulated among all concerned as a model for the remaining bench test procedures.

A tape recording of the analog and digital data channels was made and forwarded to the design engineer for the test stand telemeter data reduction system. It was used in designing and testing the test stand.

A fusing system was developed for the spacecraft electronics. The fusing system consisted of two 37-pin connectors used as a fuse block. Other wiring developments included the design of a terminal block system for the purpose of attaching leads to the solar paddles.

Mechanical supports for the main wiring harness were developed and installed in the spacecraft.

June 1961. The integration group was expanded by two electrical engineers and one summer employee. The engineers worked with the experiments and subsystems, and the summer employee, a graduate student in physics, wrote the experiment test procedures. All experiments were checked for electronic compatibility with the encoder, and the encoder was checked for compatibility with the programmer. It was found that the programmer had to be redesigned. The incompatible design was directly attributed to poor communications between design groups. Bench test procedures for the transmitter, command receiver, electron-density experiment, cosmic-ray experiment, and x-ray experiment were released. The system test procedure was nearing completion and was completed in July.

July 1961. The prototype was operated as a system through two hot and cold temperature cycles. Where they were available, prototype subsystems were used, but in some cases, breadboard subsystems had to be used to form a complete system. Tests were delayed several times because of broken leads at the wire/connector interfaces. These connectors were not encapsulated.

The prototype was also tested in the thermal-vacuum chamber. While most of the spacecraft wire harness was enclosed by the spacecraft structure, some 150 leads of the test cable were directly exposed to the environment. Polyvinyl-chloride insulated wire was used to construct the test cable and spacecraft wire harness. Considerable outgassing resulted during the thermal-vacuum tests which resulted in the condensation of a viscous liquid in the chamber, and consequent contamination of the vacuum pumps. This problem led to a series of thermal-vacuum tests on polyvinyl-chloride insulated and other types of insulated wire. The results of the test are reported by Mr. F. LeDoux (Reference 1). The end result was a new wiring harness of Teflon insulated wire.

August 1961. The data-reduction system was modified to provide synchronization lock-in capability. This capability enables the testing technicians to receive synchronized data when one or more subsystems were removed from the spacecraft. Prior to this capability, holes in the telemeter format (caused by the removal of one or more subsystems from the spacecraft) prevented the data-reduction system from synchronizing with the output data of the spacecraft.

Starting transients were measured and recorded on film; special test circuits were used to trigger the spacecraft circuit under test. The special test circuits included relays with mercury wetted contacts which provided fast rise-time waveforms free of contact bounce. Modification of the prototype spacecraft wiring harness consisted of new wiring for a battery switching network. Other modifications included pin assignments for coaxial cables on the United Kingdom experiment stacks; adjacent connectors were so close together that the overhang due to coaxial termination made it impossible to mate the cable connectors to the fixed connectors.

Calibration of the experiments was started by the experimenters. During this time, the experiment test procedures were checked. Ordinary radio type fuses were submitted for vibration testing; levels of 60 g's up to 1800 cps did not destroy any fuses of size 0.031 amperes or smaller.

September 1961. Responsibility for integration of prototype and flight unit 1 satellites was assigned to one engineer for each satellite, by the integration group leader. Environmental testing of the prototype was started. The tests performed and the results of these tests are documented in References 2 and 3.

October 1961. During this month, temperature and humidity tests, vibration tests, and shock tests were completed. The electron-density experiment exhibited performance degradation below 0°C. The cosmic-ray geiger tube was damaged at +60°C and had to be replaced, and the time of the tape recorder two-second tone burst increased to five seconds. Vibration test data indicated that the dutchman (spacer required for mounting the spacecraft on the launch vehicle) had a "Q" along the thrust axis of approximately 9 at 80 cps. This means that the payload separation mechanism was subjected to 90 g's at 80 cps; during this time, the cosmic-ray photomultiplier tube was damaged. An x-ray counter tube, which failed shortly after the vibration tests was probably damaged during the shake tests. A connector shorted during acceleration tests; it was carefully sawed open and examined. The cause of failure was found to be an improper assembly technique. The faulty procedure was revised.

A meeting was held with representatives of the launch vehicle manufacturers for the purpose of establishing the electrical requirements of the separation mechanism. Assembly of flight unit 1 was completed, and is undergoing experiment calibration. Flight unit 2 was two weeks behind schedule because of late delivery of the encoder.

November 1961. Three new wire harnesses were built; these harnesses had fewer test points, improved utilization of wiring redundancy, and all test-point connections were made at the skin instead of within the spacecraft's interior (Figure 10).

The prototype underwent hot and cold thermal-vacuum tests. During these tests, problems were revealed in the following areas: encoder clock; solar-paddle shunt regulator; spacecraft tape recorder; corona in the x-ray experiment high-voltage lines; Lyman-alpha, aspect, and cosmic-ray experiments. The new harness was partially assembled for flight unit 1. Flight unit 1 was installed in the systems integration trailer and the new wire harness placed in the spacecraft. Responsibility for integrating flight unit 2 was assigned to a third electrical engineer by the integration group leader.

December 1961 and January 1962. A circuit for protecting the spacecraft power system in the event the tape-recorder motor should stall was breadboarded. The tape-recorder motor draws 0.5 ampere on starting. This is equal to the total spacecraft load current.

Facility and personnel requirements at Cape Kennedy* were reviewed.

The prototype was tested in the thermal-vacuum chamber. During these tests the UK converter was dissipating 0.7 watts power without any heat sink. The converter was modified and performed satisfactorily on retest. The failures of experiment components, such as Aspect and Electron Density, during a hot test at 60°C did not appear on retest. The power system operated with batteries at 70°C during a simulated 100% sunlight orbit. This condition is undesirable due to reduced expected battery life and because of the additional heat dissipated within the spacecraft. A redesigned wiring harness was installed in the prototype and subjected to a vibration test. During this time, the environmental test committee reviewed the results of the prototype thermal-vacuum tests to determine what portions of the spacecraft are qualified.

Flight unit 1 completed experiment calibration and preliminary tests in the temperature chamber. The cosmic-ray experiment was missing, and a mock-up was used in the spin and balance test scheduled for January 9, 1962. Flight unit 2 was behind schedule. It was missing several experiments including cosmic-ray, x-ray, and some of the experiment probes and gauges. Power was applied successfully to the electronics on hand.

February 1962. A battery charging-current limiter and a tape recorder overload relay were added to the spacecraft power system, and the wiring harness for all three spacecraft modified accordingly. Both circuits have operated satisfactorily in flight unit 1 in room temperature tests.

*Formerly Cape Canaveral.

Prototype thermal-vacuum tests were completed early this month. However, the review of environmental tests by the Environmental Committee resulted in scheduling retests of prototype to qualify the following redesigned and modified system components: prime converter, UK converter, programmer, electron-density experiment, x-ray experiment, aspect experiment, electron-energy experiment, Lyman-alpha experiment, and command receiver. The environmental retests will also qualify the battery charging-current limiter and tape recorder relay mentioned above.

Flight unit 2 was spin and balance tested. The spacecraft was scheduled for calibration of experiment thermistors in the temperature chamber.

March 1962. The prototype spacecraft's antenna pattern and solar radiation were tested with a thermal-vacuum retest scheduled next to qualify the tape recorder in a hot test. The thermal-vacuum tests and final calibration of flight unit 1 were completed. Antenna pattern tests and final balance remained to be completed before shipment to Cape Kennedy for launch preparation. Thermal-vacuum tests with a spare cosmic-ray unit, replacing the unit which failed vibration testing, were started on flight unit 2. After final testing, all three spacecraft were delivered to Cape Kennedy.

CABLE HARNESS

One of the early steps of the integration process was to plan the construction of the cable harness. The cable harness design was based on information from the experimenters and instrumenters. This information included a wiring table which listed the origin, function, destination, size, type, and color of each wire. All pin assignments were made by the individual experimenters and electronic subsystems designers.

In the original cable harness all power lines to the experiments were fused, the solar paddles were connected to a terminal strip, the main power lines were brought through the turn-on plug, and the high voltage lines for the x-ray experiment were brought through banana plugs and sockets which were insulated by lucite covers. These covers were threaded and could be screwed together. Polyvinyl-chloride insulated wire and miniature coaxial cable was used in the construction of the harness.

During the initial testing of the prototype spacecraft it was found that the starting or turn-on transients in the power lines to the experiments were large enough to blow the fuses when power was applied to the experiments, the use of the terminal strip to connect the solar paddles made it difficult to remove the cable harness when disassembling the satellite, and the connectors in the high-voltage line of the x-ray experiment were subject to corona in vacuum. The polyvinyl-chloride wire insulation outgassed to an objectionable extent, and much trouble developed in the harness due to breaks in the miniature coaxial cable that was used for shielded leads. The coaxial cable depended on physical clamping of its outer conductor for most of its tensile strength. When used as shielded leads, the outer conductor was not mechanically clamped, and all stresses were transferred to the inner conductor, which is relatively weak. Thus, breaks in the wire occurred frequently.

In the original cable harness (Figure 11), all of the power lines were brought through a pair of connectors located inside the satellite which could be disconnected and a power meter panel inserted. The use of the meter panel was a great aid in the integration of the electronic subsystems, however, this method of inserting the power meter panel in series with the power lines resulted in excessive handling of the cable harness, and severe stress on the harness.

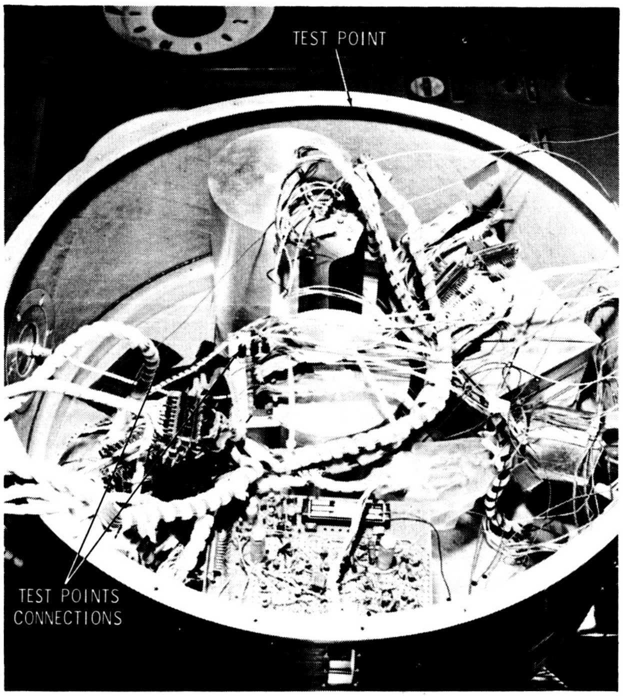

Figure 11—Test point connection before redesign of wire harness.

Because of these problems with the original cable harness, it was decided to modify the design of the harness and build three new ones. Teflon insulated wire was chosen for the new harnesses on the basis that its outgassing properties were negligible compared to polyvinylchloride. For shielded leads, No. 22, shielded, Teflon insulated wire was used, which decreased the number of broken wires from handling and testing. The fuses were eliminated from the harnesses, as well as the connectors for the meter panel. Since enough information on power consumption by the subsystems of the satellite had been collected during prototype testing, the meter panel was no longer needed. It was also decided to bring all of the test points out to the turn-on plug instead of having a separate harness for test points. The number of test points brought out was reduced (from 120 to 37) to only those that were absolutely essential for checkout of the spacecraft. The solar-paddle terminal strip was replaced by a mating 37-pin connector pair, with the common voltages bussed together on connectors at the solder pot end of the contacts. This made it possible to remove the harness when the satellite was disassembled for inspection after tests. Also, the high voltage lines for the x-ray experiment were soldered and the joint encapsulated to eliminate the corona problem. To disassemble the x-ray detector from its associated printed circuit card, it was necessary to cut off the encapsulated solder joint in the high-voltage (1600v) line.

The cable harnesses supplied by the British for their experiments were not suited for encapsulating. When the new cable harnesses were being built it was decided to rebuild the British harnesses also. Thin walled, no. 28 teflon wire was used, the harnesses were laid out in a manner that made the connectors more accessible and thus easier to encapsulate and install. Several wiring changes were made to add redundancy to the power lines.

SPACECRAFT EVALUATION AND FLIGHT CHECKOUT

General Performance

The task of the integration group was to mold the electronic subsystems into an operational spacecraft. To accomplish this task, the integration group tested each subsystem to determine normal operating parameters, tested the integrated system for compatibility, and monitored the spacecraft performance during the environmental tests and launching. For the convenience of presentation, the various test phases will be referred to as integration testing, environmental testing, and launch-site testing.

Integration Testing

Integration testing consisted of developing bench test procedures for each subsystem, developing an overall system test procedure, connecting all subsystems to the wire harness, and testing the spacecraft subsystems individually and as an integrated operational spacecraft.

To develop and perform bench tests, the integration group had to gather specifications, information on operating characteristics, and input/output data for each subsystem. This was accomplished using a wiring and test procedure checklist which was distributed to the experimenters. The experimenters completed the checklists and returned them to the integration group. A bench test procedure for the cosmic-ray experiment was developed using the data received from the experimenter. After approval by the experimenter, the procedure was circulated among all concerned as a guide for the development of the remaining bench test procedures.

After each subsystem had been properly tested on the test bench, it was installed in the spacecraft. Test points were incorporated in the wiring harness and test cables connected these to a test stand. The test stand was used to check the compatibility of the integrated subsystems. Transients, power consumption, automatic controls, and operation of each subsystem were checked. The spacecraft was tested for magnetized components in the Magnetics Laboratories of the Naval Ordnance Laboratory, Silver Spring, Maryland. If the results of any test were unsatisfactory, the integration group reported this to the Project Manager. It also sometimes recommended changes to correct the malfunctions. After much testing, redesign, and effort, the integration group developed a completely integrated electronic system. During this time the system test procedure was developed.

Environmental Testing

Environmental testing includes subjecting the spacecraft to all conditions it will encounter from the time of launching to its final position in orbit. Some of these conditions are acceleration, vibration, temperature, humidity and vacuum. Each of these conditions must be simulated before launching, and the spacecraft tested for normal operation. If the spacecraft failed during any of these tests, the integration group reported the failures so that appropriate steps could be taken to correct them.

The Goddard Space Flight Center supplied simulating devices for the vibration, temperature, humidity, and vacuum tests. The Naval Research Laboratory in Washington, D. C. supplied the acceleration simulator (Figure 12). The integration group supplied the test stand and monitored the operation of the spacecraft during the environmental tests.

Figure 12—Spacecraft in centrifuge at the Naval Research Laboratory.

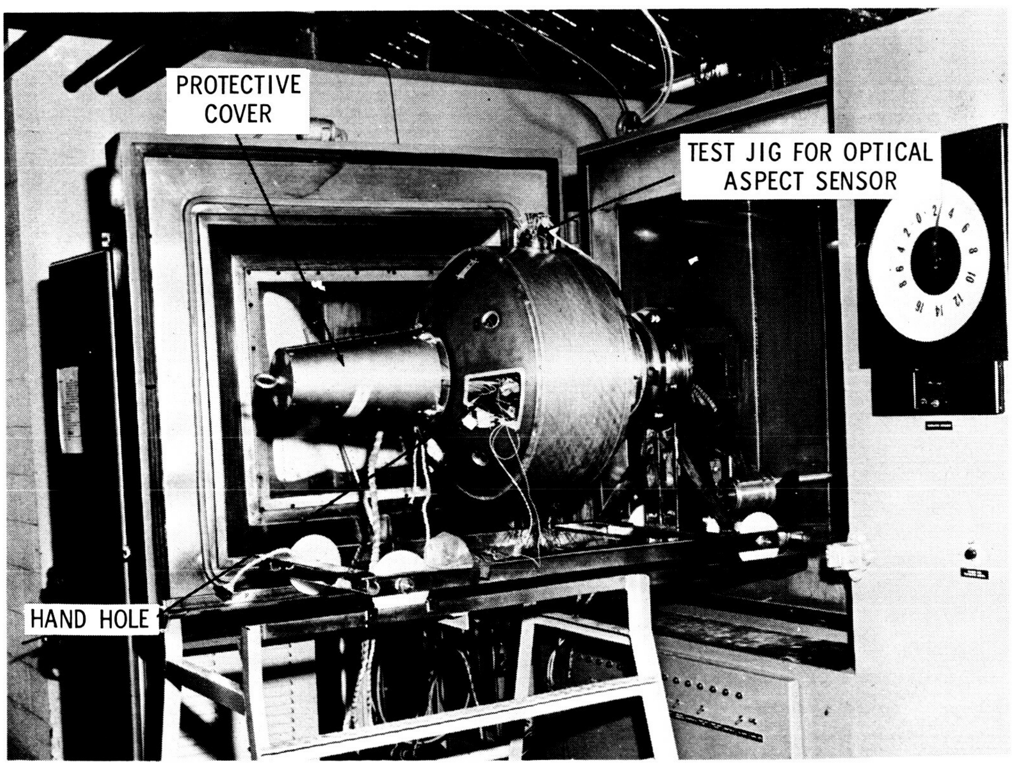

Figure 13—Spacecraft mounted for thermal and humidity tests.

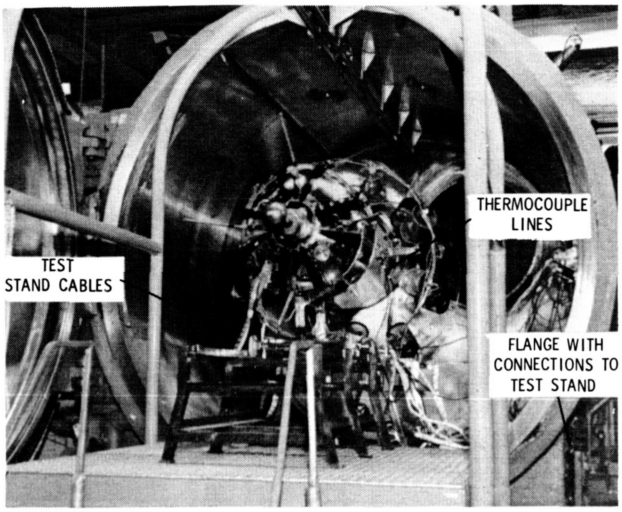

Figure 14—Spacecraft in thermal-vacuum test chamber.

The vibration tests were conducted at the Goddard Space Flight Center. No serious malfunctions in flight units 1 and 2 occurred during these tests. Reference 1 contains the tests results.

The temperature and humidity tests were performed at the Goddard Space Flight Center. Figure 13 shows the spacecraft mounted on a dolly outside the test chamber. During the test, several difficulties were encountered in the spacecraft and its associated monitoring equipment. A complete report on this test will be found in Reference 3.

The thermal-vacuum tests were performed at the Goddard Space Flight Center. Figure 14 shows the spacecraft mounted in the test chamber. During this test, it was discovered that the polyvinyl-chloride insulated wire outgassed considerably. This led to a thorough investigation of different types of insulated wire. The end result was a new wire harness using Teflon insulated wire.

Launch Site Testing

Two complete ground stations, each containing a special telemeter radio receiver, were used to monitor the spacecraft performance at the launch site. The main station was located inside a trailer and the backup station was placed in a hangar. The back-up station was operated in parallel with the main station during testing and launch. The special telemeter radio receivers were fed by two 136 megacycle preamplifiers connected to a 9-element yagi antenna. A frequency counter and signal generator were used to measure the spacecraft transmitter frequency. Launch site testing included a complete system test of the spacecraft in the trailer upon arrival at Cape Kennedy. All other tests were conducted exclusively by rf link from the spin facility and launch pad gantry. Each test or checkout included a complete analysis of test data. Several practice runs of the final prelaunch countdown were performed.

Description of Test Stand

The performance evaluation test stand is shown in block diagram form in Figure 15. This equipment can be classified into functional groups as follows: (1) power control system, (2) rf system, (3) sensor exciter panel, (4) data reduction system, and (5) data recorders.

Power Control System

The power control system simulates the power output, internal impedance of the solar array, and the sunlight/darkness cycle that the spacecraft will experience in orbit. The internal impedance

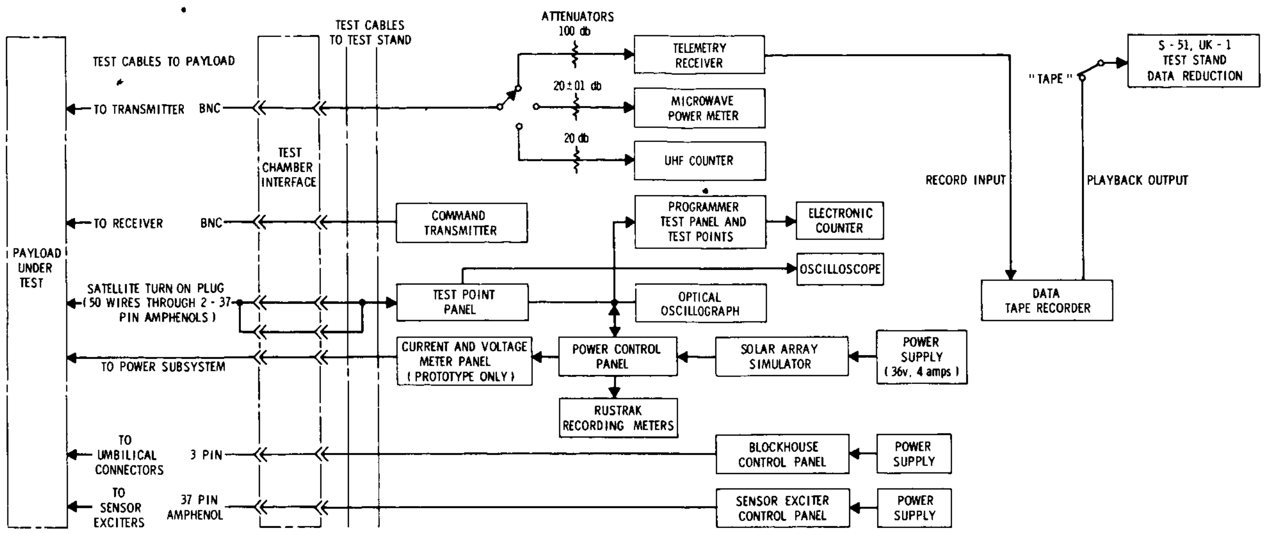

Figure 15—Spacecraft test stand block diagram.

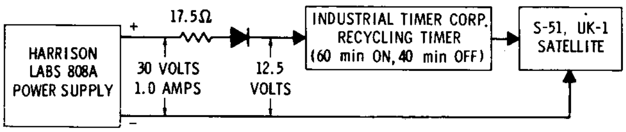

Figure 16—Solar cell array simulator block diagram.

and power output of the solar array is simulated by a 17.5-ohm resistor in series with a 30-volt power source, arranged as shown in Figure 16. The resistor and power supply form the Thevenin equivalent of the solar array. The diode, connected between the resistor and the timer, prevents reverse current from flowing through the power supply from the spacecraft battery system should the power supply voltage accidentally fall below the battery voltage. Throughout the testing phase, the spacecraft was powered electrically through the solar array simulator.

The recycling timer is on for 60 minutes and off for 40 minutes; this simulates the least favorable sunlight to darkness ratio expected during the 100-minute orbit.

During the prototype tests a meter panel was used to measure the voltages and currents used by each subsystem (Figure 15). Each current meter is shunted by a shorting switch. This enables the operator to insert or remove the meter from the circuit. Individual current meters were used for each line that was monitored. A single voltmeter was switched to the voltage point to be measured. The voltage readings were normalized by placing a miniature potentiometer at each voltage test point; the potentiometers were adjusted so that the voltmeter would show a midscale indication if the voltage was correct, which permitted rapid voltage checking. Each power line to a subsystem was switch-controlled from the meter panel. This facilitated deactivation of individual subsystems. Also, each power converter was monitored at the input, and switch controlled at the output; this made it possible to remove any converter from its load and switch in external power. These switching arrangements greatly simplified many tests.

RF System

The rf system receives and demodulates the rf signal from the transmitter, measures transmitter power and frequency, and presents an rf load to the transmitter and the command receiver.

The rf system consists of a simulated command transmitter, an rf power monitor, a uhf counter, and a phase tracking telemetry receiver. The simulated command transmitter consists of a crystal oscillator followed by an amplitude-modulated rf stage. The modulating signal is generated from an L-C audio oscillator adjusted to the command subcarrier frequency. The total power output from the transmitter is about 80 milliwatts. At the launch site, it was necessary to provide a power amplifier (10-db gain) to boost the power output because the spacecraft was several miles from the test stand.

Sensor Excitor Panel

This panel provides artificial stimulation to the aspect sensor, the electron density sensor, and the Lyman-alpha sensors. The aspect stimulator is a transistor driver relay circuit that intermittently illuminates three flashlight bulbs. The flash rate and flash separation of the bulbs corresponds to a specific spin and aspect angle. The electron-density stimulator consisted of an 8-inch plastic disc with a pie-shaped piece of brass shim stock attached to it; this was mounted on the electron density boom and rotated by a 1-rpm synchronous motor. Two notches on the rim of the disc enabled a microswitch, with a roller-cam follower, to position the disc in one of two positions: (1) maximum proximity between brass and plates, and (2) minimum proximity between brass and plates. This is illustrated in Figure 17.

The *Lyman-alpha sensor* excitor consists of a hydrogen-discharge tube and a high-voltage power supply. The discharge tube was used in the vacuum chamber, and illuminated one of the Lyman-alpha sensors. The Lyman-alpha line (for ultra-violet light) is readily absorbed in air by water vapor, and can only travel a few millimeters.

The *x-ray sensor,* the *cosmic-ray sensor* and the *electron-temperature probes* were not excited from the sensor excitor panel. A 50 millicurie source of iron (Fe 55) was used to provide 5-kv x-rays. A radioactive source was used to excite the cosmic ray experiment. A diode-resistor dummy load was applied to the electron-temperature probe sensors to simulate free electrons with a specific average energy. A diode-resistor dummy load was applied to the ion mass spectrometer to simulate ions of a specific mass.

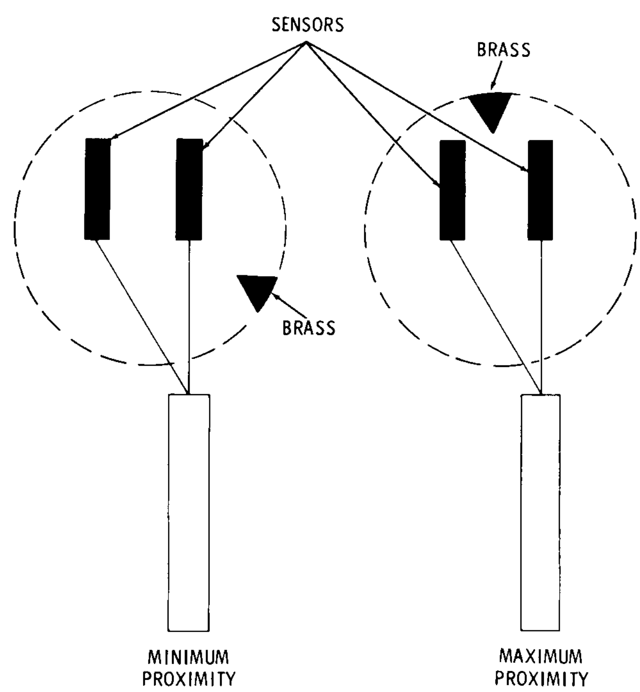

Figure 17—Electron density boom exciter.

Data Reduction System

The integration group performed a study on the decoding equipment requirements, described in Appendix A.

Data Recorders

Printers in the test stand data reduction system provided printed records of test stand data output. (See Appendix A for discussion of format.) An instrumentation data magnetic tape recorder was used to record the complete demodulated rf signal from the spacecraft.

LAUNCH OPERATIONS

General Information

On March 12, 1962, system integration of the spacecraft was drawing to a close and preparations for launch operations at Cape Kennedy started. Four electronic integration personnel were selected to go to Cape Kennedy with the prototype, three with flight unit 1, and two with flight unit 2.

Prototype Launch Operations

On March 15, the prototype arrived at Cape Kennedy. The spacecraft was given a complete system test, using the test stand mounted in the trailer. In the following days, the spacecraft was taken to the spacecraft level on the gantry, the spin facility, and the airframe and engine (A&E) hangar.

At the gantry, the rf link between the trailer test stand and the spacecraft was tested. This assured that the trailer test stand could receive the telemetered signals, and that the tape recorder in the spacecraft would play back on command from the trailer.

At the spin facility, the spacecraft was spun and balanced. Also, another rf telemeter test was conducted. When the rf link test was completed, the spacecraft was taken to the A&E hangar. A back-up test stand had been installed here and it was checked by conducting another system test on the spacecraft.

Between March 21 and 30, the spacecraft was mounted on the third stage of the launch vehicle and given a complete system test. After the system test, the complete third stage and spacecraft were enclosed in a metal casing and mounted on the second stage of the launch vehicle. At this time, further rf tests were conducted, and a practice countdown performed. During the practice countdown a spacecraft engineer and technician were stationed at the spacecraft level of the gantry. The spacecraft engineer coordinated the countdown. At the completion of the practice countdown, the prototype spacecraft was removed from the launch vehicle and moved to the spin facility and placed in storage.

Flight Unit 1 Launch Operations

On March 20, flight unit 1 arrived at Cape Kennedy. A system test and antenna pattern test were conducted. The spacecraft experiments were calibrated and retested. Between March 27 and 29, flight unit 1, the separation mechanism, and dutchman were spun and balanced at the spin facility. From March 30 to April 4 the spacecraft received a final spin balance; and the spacecraft, separation mechanism, dutchman, and third stage motor was assembled. A final checkout of all systems was performed prior to mounting the assembled third stage and spacecraft on the second stage of the launch vehicle. From April 5 to April 9, more practice countdowns were performed.

A final countdown began at 0100 on April 10. The experience gained during the practice countdowns provided a smooth countdown as far as the integration responsibilities were concerned. However, a launch vehicle malfunction prevented the launching. The spacecraft was removed from the launch vehicle and moved to the spin facility. The spacecraft was spun and balanced again. On April 23, the spacecraft was remounted on the launch vehicle. The batteries were charged on April 25, and final countdown began at 2400 hours on April 26.

Contrary to the smooth operation of the previous countdown, trouble developed when the low and high speed encoders failed to maintain synchronization. The malfunction was attributed to rf interference, possibly originating from electrical equipment on the gantry. To eliminate the trouble, all electrical equipment, which had no primary influence on the countdown, was turned off. The actual source of the rf interference was not determined. The malfunction could have caused a slip in the launch time, but because of one hour hold period included in the countdown, the spacecraft was launched on time.

Flight Unit 2 Launch Operations

Flight unit 2 arrived at Cape Kennedy on March 27. The spacecraft was given a system test, calibrated, and spin balanced. After the complete checkout, the spacecraft was placed in standby readiness to replace flight unit 1 in the event of a malfunction during the launch countdown.

CONCLUSIONS AND RECOMMENDATIONS

Wiring Harness Recommendations

1. Work out wiring requirements (accessibility of connectors, holes in the spacecraft structure for routing of wires, supports and tiedown fixtures for wires) with the mechanical engineers who are assembling the spacecraft.

2. It is extremely difficult to make any changes in a harness which has all of its connectors potted. It is easier to rebuild the entire harness rather than attempt any extensive modifications on a harness with potted connectors.

3. Single conductor stranded wire with polyvinyl-chloride insulation was ordered for the construction of the wiring harness. Subsequent operation of the spacecraft in the thermal-vacuum chamber caused the insulation of this wire to outgas. The outgassing collected on the window of the thermal-vacuum chamber as an oil and contaminated the vacuum pumps. As a result, GSFC performed tests to determine the outgassing properties of various insulating materials. The

results of these tests are documented by Mr. F. LeDoux (Reference 1). The wiring harness was redesigned using Teflon insulated wire, which eliminated the outgassing problem.

4. Several failures occurred during the thermal test, which were directly attributed to broken wires at the wire/connector interface. This problem was remedied by potting the connectors. It is recommended that all connectors be potted prior to testing.

5. Place all test points within easy access of hand holes. This will eliminate the problem of removing the spacecraft cover each time a new test is performed. Figure 12 shows the spacecraft prior to placing the test points within easy access of the hand holes.

Test Equipment and Testing Recommendations

1. The initial order of test equipment was based on the premise that two sets would be sufficient for the needs of the experimenters during the integration of the prototype, and one set would be needed by the time flight unit 1 was assembled. It was anticipated that no test equipment would be required by the experimenters by the time flight unit 2 was scheduled for integration testing. However, the experimenters required almost constant use of three to four sets of test equipment up to the delivery time of flight unit 1 to Cape Kennedy. It is recommended that a careful evaluation of test equipment requirements be made on future projects.

2. When selecting or designing equipment to decode and reduce telemetered data from a spacecraft, capabilities such as storing binary data from selected data channels and operating on this data to produce a decimal equivalent is more desirable than a high volume of individual printouts.

3. Bench test procedures for all subsystems should be developed and circulated early in the program.

4. Solid state fuses were used to protect the transistorized circuits in the prototype spacecraft during testing. These fuses would blow when subjected to short duration overloads. In addition, they appeared to remember small overloads, and would blow after the same small overload had been applied many times (excess of 100 times). In the re-designed wiring harness the fuses were eliminated. During the prototype testing a meter panel was wired to the spacecraft to measure load currents and voltages of the individual subsystems. Also, power could be turned on or off to the individual subsystems.

5. A twinax connector shorted during mating. After replacing, the defective connector revealed that it had been improperly assembled. All faulty components should be carefully examined to determine the reason for malfunction.

Prelaunch Recommendations

1. Several practice prelaunch countdowns before the launch date are suggested. This is a prerequisite for a smooth countdown on the launch date.

2. During final countdown, trouble developed in the low and high speed encoders. It was determined that the difficulty was caused by rf interference getting into the spacecraft through the test leads. The actual source of the rf interference was not determined. To eliminate the trouble, all electrical equipment which had no primary influence on the countdown was turned off.

3. A reliable communications system between everyone involved in the integration countdown is necessary. A channel and a channel box at the spacecraft level of the gantry should be permanently assigned to the integration group and marked.

4. The experience gained in performing the practice prelaunch countdowns proved that it is essential to limit the personnel in the spacecraft area on the gantry to an absolute minimum. The reasons for this are: the working space is extremely limited, personnel in close proximity to sensitive experiment sensors can cause the transmission of incorrect data, and reducing the

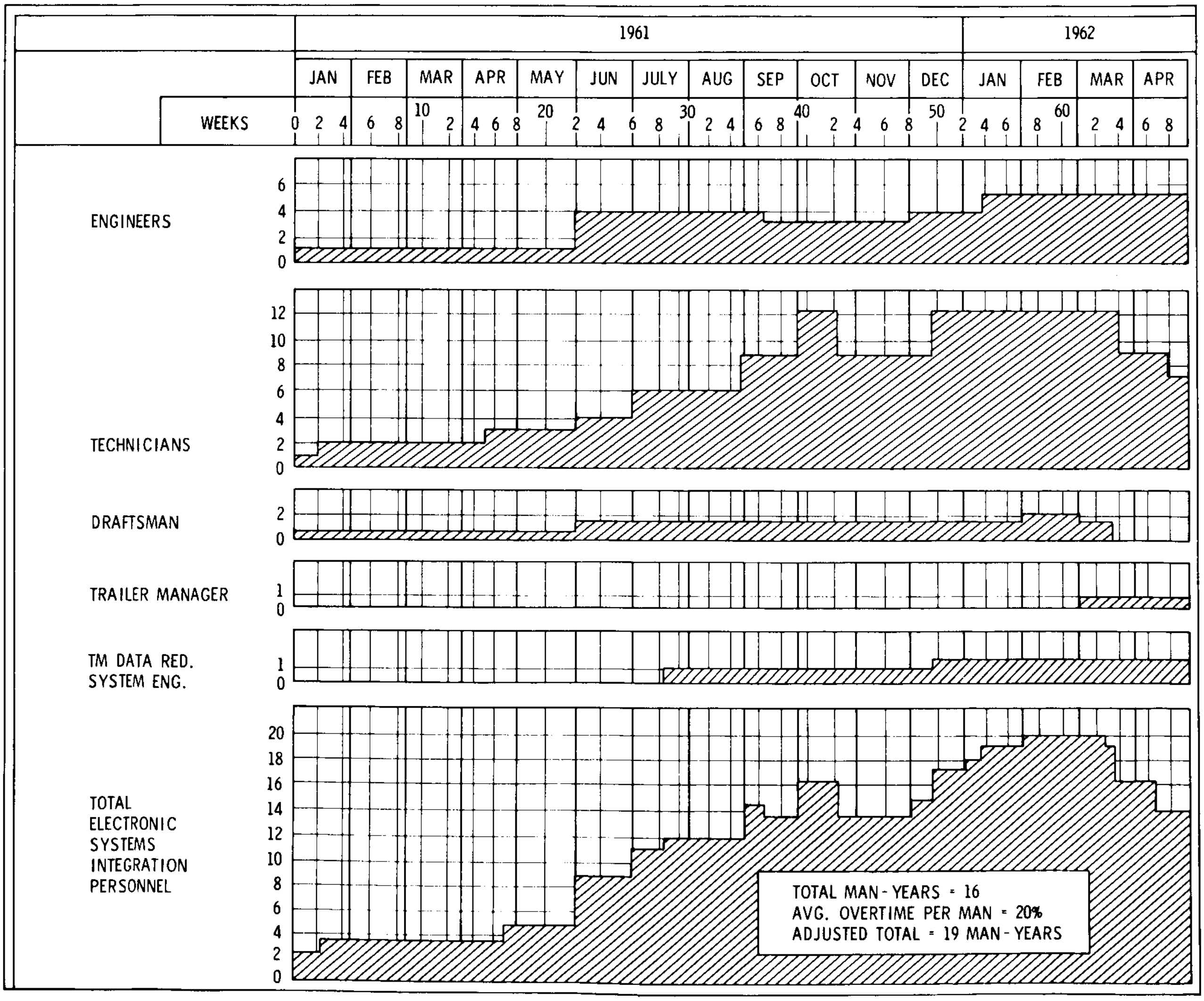

Figure 18—Electronic system integration manpower requirements.

number of persons reduces the possibility of someone accidentally bumping or brushing against a delicate part of the spacecraft and thereby causing a malfunction.

Manpower Recommendations

Figure 18 illustrates the manpower requirements of the integration group. The figure shows, that, the build-up in technicians was rather rapid. Technicians and engineers are difficult to find; therefore, on future projects steps necessary to obtain these people should begin early in the program.

(Manuscript received November 17, 1964)

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Appendix A

PROPOSED REQUIREMENTS FOR A SATELLITE TEST STAND TELEMETER DATA REDUCTION SYSTEM

Introduction

This proposal describes the encoder and data transmission system for the satellite. It discusses the significance of the individual data channels, the reduction of the data and its presentation. It sets forth requirements for equipment to perform the reduction and presentation of this data. The terminology used in this proposal to discuss the reduction of telemeter data is defined in Reference 4.

Description of Operation

Data from the experiments is processed simultaneously by an HS encoder and an LS encoder. The output of the LS encoder operates at 1/48 of the speed of the HS encoder; all of its data channel frequencies are divided by 48, and its output is recorded on a tape recorder. The tape recorder uses an endless tape which moves in the same direction for record and playback. During playback, however, the tape speed is 48 times the record speed. Thus the playback has data channels whose width and frequency range are identical to those of the HS encoder.

To transmit the data stored by the tape recorder, an rf command is sent to the satellite from a ground station. The command receiver sends a pulse to the programmer, at which time the programmer: (1) disconnects the HS encoder from the transmitter, (2) disconnects the tape recorder input from the LS encoder, and (3) gates a 320.83 cps signal for two seconds to the transmitter and tape recorder. The two seconds of 320.83 cps is thus transmitted and at the same time recorded on the tape recorder. The transmitted pulse signals receipt of the command and the tape recorder playback follows immediately. At the end of the two seconds of 320.83 cps the programmer (1) switches the tape recorder from record to playback and simultaneously switches the tape speed to 48 times its record speed, and (2) connects the transmitter to the tape recorder.

Transmission of tape recorder playback for a period of 125 to 134 seconds is controlled by a timer in the tape recorder. This period of time is sufficient to play back all stored data. This must include a 15.4 kc pulse, approximately 42 milliseconds long, which is the 2 seconds of 320.83 cps recorded immediately prior to tape recorder playback. This pulse indicates end of LS encoder data. It also serves as a time reference for correlating data from both encoders.

Telemeter Data

The telemetered rf signal will go through a phase-modulation receiver, a tracking filter, and into the pfm decoder. The input to the decoder will be a series of pulses in the frequency range from 4.5 kc to 15.4 kc. The length of the pulses is 10 milliseconds duration and the pulses are separated by 10 milliseconds, except for the frame sync pulses which last for 15 milliseconds and are preceded by a space of 5 milliseconds. The frame sync pulses, in addition to being coded by an additional 5 milliseconds of pulse width, are also coded in frequency as shown in Table A1.

Channel zero in each frame for both HS and LS encoder is used for frame sync pulses. All the remaining channels are either digital data channels or analog data channels. The pulse frequency of the analog data channels varies from 5 kc to 15 kc. Table A1 also correlates the digital data channel pulse frequencies with bit inputs to the digital oscillator.

General Requirements for Test Stand Data Reduction Equipment

The HS and LS encoders are synchronized and the data from the two encoders is related. Since the data from the two encoders is transmitted at different times, it is necessary to reconstruct the time relationship. Therefore in addition to other requirements, the test stand data

Table A1

Frame Sync Pulses and Digital Data Channel Pulses.

FRAME SYNC PULSES				Digital Data Channel Pulses
High Speed Encoder		Tape Recorder Playback of Low Speed Encoder		
Channel O, Frame	Frame Sync Pulse Frequency (cps)	Channel O, Frame	Frame Sync Pulse Frequency (cps)	Binary Bit Input to Digital Oscillator
0	4,500	0	4,500	---
1	5,140	1	---	000
2	4,500			---
3	5,600	---	---	001
4	4,500			---
5	6,230	---	---	010
6	4,500			---
7	6,960	---	---	011
8	4,500			---
9	8,015	---	---	100
10	4,500			---
11	9,250	---	---	101
12	4,500			---
13	11,190	---	---	110
14	4,500			---
15	14,120	---	---	111

reduction equipment must have a four place decimal counter to count each encoder sequence (HS or LS), independently of whether or not a data print-out occurs during the sequence.

Significance of Digital Data Channels and the Required Reduction and Presentation of their Data

The cosmic ray and x-ray experiments have HS encoder digital data channels.

Cosmic Ray Experiment: The cosmic ray experiment has two separate binary counters. Binary counter #1 has HS encoder channels C_1(1-0), (1-8),* C_2(2-0)(2-8), C_3(3-0)(3-8) and C_4(1-1)(1-9) which represent binary bits (2^1, 2^2, 2^3); (2^4, 2^5, 2^6); (2^7, 2^8, 2^9); and (2^{10}, 2^{11}) respectively. This is shown in Figure A1.

In particular it should be noted that the most significant binary bit of C_4, called "Sensitivity Indicator 0 or 1" does not come from counter #1, and thus should be treated as a separate piece of data. In addition, the binary number stored in counter #1 is two times the number represented by C_1, C_2, C_3, and C_4 because binary 2^0 is not one of the digital data inputs.

Binary counter #2 has HS encoder channels C_5(2-1)(2-9) and C_6(3-1)(3-9) which represents binary bits (2^0, 2^1, 2^2); and (2^3, 2^4, 2^5) respectively. This is shown in Figure A2.

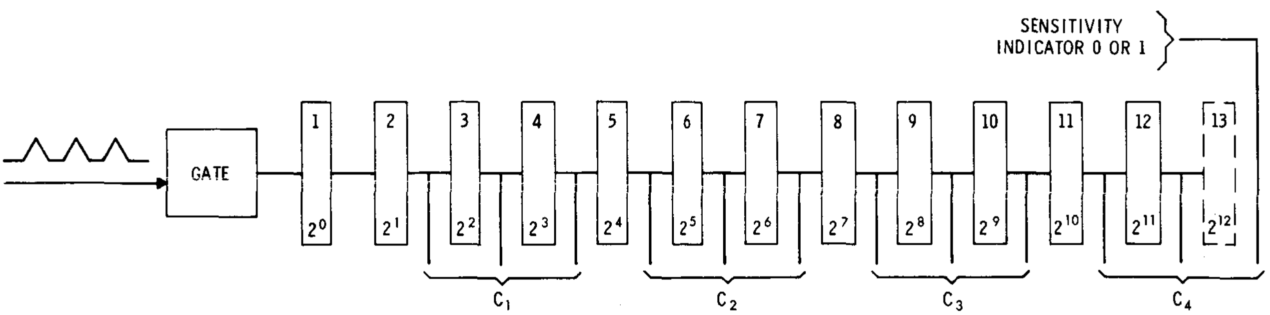

Figure A1—Binary counter #1 (HS Encoder) digital data channel connections.

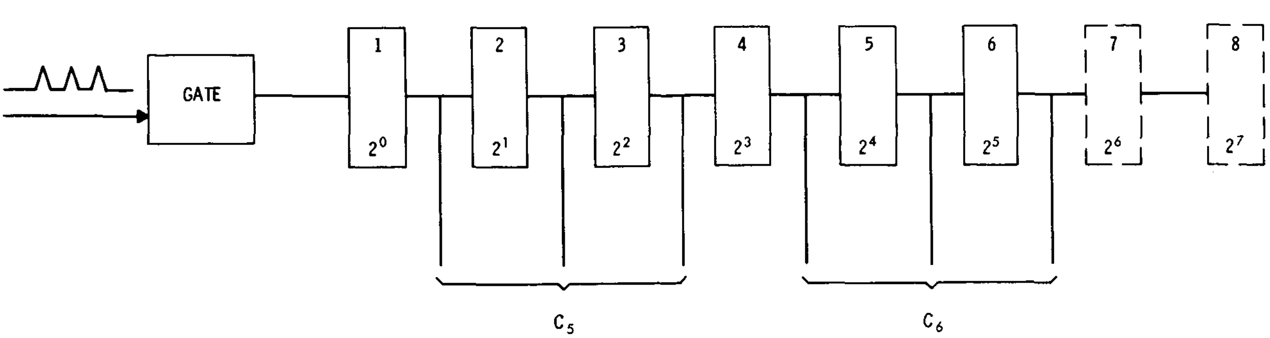

Figure A2—Binary counter #2 (HS Encoder) digital data channel connections.

*See Table 1 of Reference 4 for explanation of numbers following C_1, etc.

In the experiment "Sensitivity Indicator 0 or 1" indicates the sensitivity of the discriminator which sends pulses to counter #2, and so it is desirable to print "Sensitivity Indicator 0 or 1" and the number stored in counter #2 adjacent to each other.

Binary counter #1 also has LS encoder digital data channels C_1 (8-0), C_2 (9-0), C_3 (10-0), and C_4 (11-0) which represent binary bits (2^3, 2^4, 2^5); (2^6, 2^7, 2^8); (2^9, 2^{10}, 2^{11}); and (2^{12}, 2^{13}) respectively. This is shown in Figure A3.

In particular it should again be noted that the most significant binary bit of C_4, called "Sensitivity Indicator 0 or 1" does not come from counter #1, and thus should be treated as a separate piece of data. In addition, the binary number stored in counter #1 is 4 times the number represented by C_1, C_2, C_3, and C_4 because binaries 2^0 and 2^1 are not used as digital data inputs.

Binary counter #2 has LS encoder channels C_5 (14-0) and C_6 (15-0) which represent binary bits (2^3, 2^4, 2^5) and (2^6, 2^7, 2^8) respectively. This is shown in Figure A4.

In the experiment "Sensitivity Indicator 0 or 1" indicates the sensitivity of the discriminator which sends pulses to counter #2, and so it is desirable to print "Sensitivity Indicator 0 or 1" and the number stored in counter #2 adjacent to each other.

X-Ray Experiment: The x-ray experiment has one 15-stage binary counter. HS encoder channels X_1(2-14), X_2(3-14), X_3(1-15), X_4(2-15) and X_5(3-15) represent binary bits (2^0, 2^1, 2^2); (2^3, 2^4, 2^5); (2^6, 2^7, 2^8); (2^9, 2^{10}, 2^{11}) and (2^{12}, 2^{13}, 2^{14}) respectively. This is shown in Figure A5.

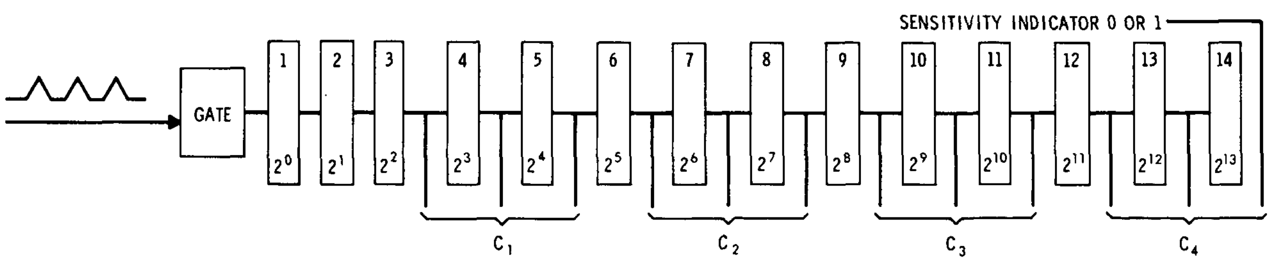

Figure A3—Binary counter #1 (LS Encoder) digital data channel connections.

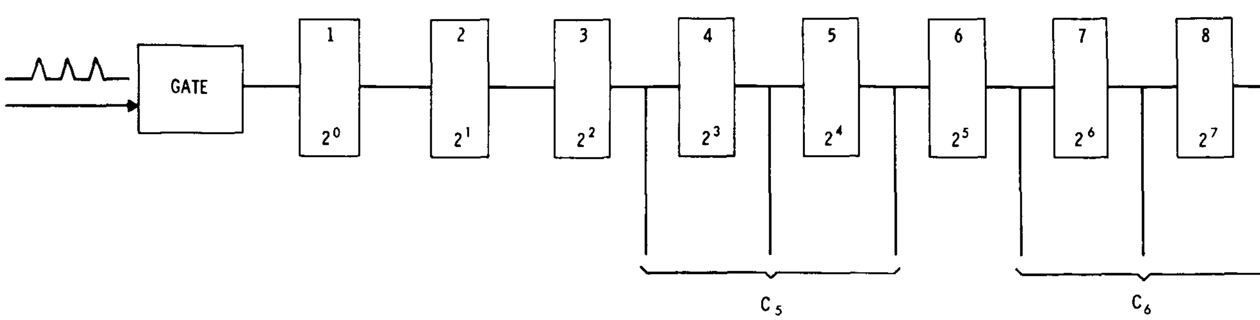

Figure A4—Binary counter #2 (LS Encoder) digital data channel connections.

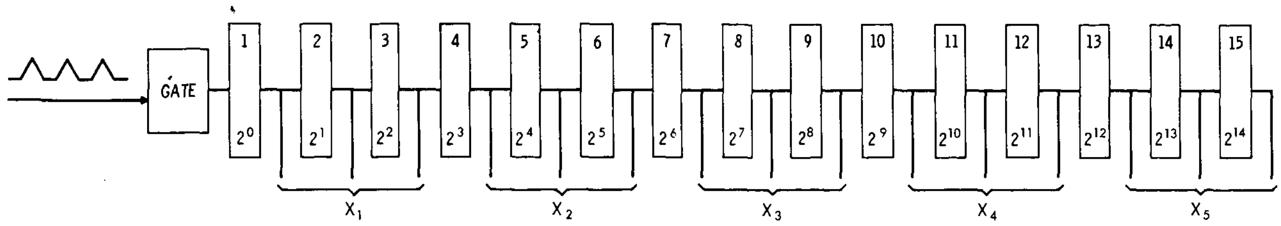

Figure A5—X-ray binary counter.

General Requirements: All numbers are to be converted to decimal before print-out. Each decimal number is to be identified as to the experimental data it represents. It will be sufficient to print out the number stored in one counter of one experiment for each HS or LS encoder sequence, and to have a switch which will select which counter store will be printed. This means it would be sufficient to print C_1 (1-0), C_2 (2-0), C_3 (3-0), C_4 (1-1) as one number per HS encoder sequence; C_5 (2-1), C_6 (3-1) and "Sensitivity Indicator 0 or 1" of C_4 (1-1) as one number per HS encoder sequence; C_5 (2-9), C_6 (3-9) and "Sensitivity Indicator 0 or 1" of C_4 (1-9) as one number per HS encoder sequence; X_1 (2-14), X_2 (3-14), X_3 (1-15), X_4 (2-15) and X_5 (3-15) as one number per HS encoder sequence; and C_1 (8-0), C_2 (9-0), C_3 (10-0) and C_4 (11-0) as one number per LS encoder sequence. Each encoder sequence is to be counted and its number printed whenever a decimal number is printed from any of its channels.

The x-ray experiment does not have any LS encoder digital data channels.

Significance of the HS Encoder and LS Encoder Analog Data Channels and the Required Reduction and Presentation of their Data

Analog data channels are exactly what their name implies. The only reduction required for these channels is to count the frequency of the pulse. The presentation required is a print-out of pulse frequency, data channel identification and number of encoder sequence during which the frequency pulse occurred.

It would be sufficient if the selectivity of the "Single Channel/Frame Selector Section" were limited to selecting one of 16 channels to be printed once for every frame of the encoder sequence. The fact that this arrangement would print the pulse frequency of any digital data channels that happen to be in the column of selected channel "X" is not a drawback, since it will allow for checking the digital oscillator pulse frequency for any drift which might begin to occur. The same considerations apply equally well to frame sync channel 0.

For channels 1, 2, and 3, the channel of interest may occur only once per HS encoder sequence, and the rest of the print-out is an inconvenience. However, for channels 0 and 4 through 15 the print-out for each frame is a distinct advantage. In the case of the LS encoder data this type of selection is acceptable with no particular advantages or disadvantages.

It is also required that an "output sync pulse" be made available on one "Output Sync Pulse" terminal. It shall have one distinct output at time of channel 0, frame 0 (0-0) at the beginning of each encoder (HS or LS) sequence. It shall also have another distinct output which will occur during the time of telemetering of the channel selected for print-out.

These pulses are for marking one channel of a strip chart recorder. Another channel of the strip chart recorder will be connected by a wire to a test point in the satellite which monitors the voltage input to the encoder oscillator. The "Output Sync Pulses" will thus (a) identify the beginning of each encoder sequence and (b) will mark that portion of the test point trace during which telemetering took place. Thus the performance of the Encoder oscillator may be checked at the same time that performance of an experiment parameter is being checked.

Correlation of HS Encoder Data with LS Encoder Data

If an auxiliary tape recorder is used to record the demodulated playback of the satellite tape recorder, the matter of correlation of HS and LS encoder data becomes a straightforward procedure. It consists of the following steps:

(a) Operate the satellite for a little over 1-1/2 hours while recording the HS encoder digital channel data on one printer and the selected channels "X" on the other printer.

(b) Arrange for the switch which initiates the rf command signal to the satellite to also put the auxiliary tape recorder into the record mode. Operate the switch and record the two seconds

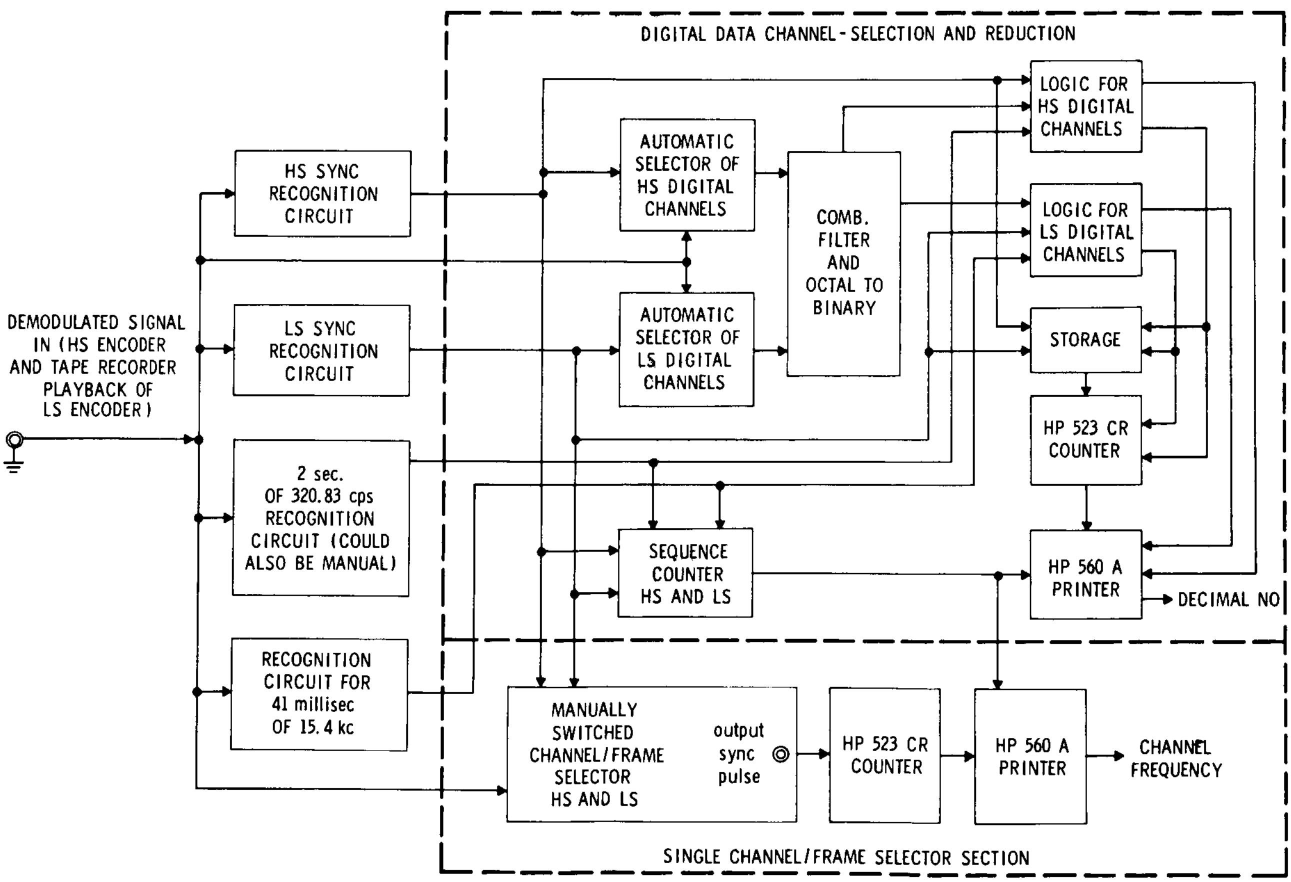

Figure A6—Proposed spacecraft test stand data reduction system block diagram.

of 320.83 cps, and the full playback of the tape recorder which must include the 41 milliseconds of 15.4 kc time marker.

(c) Rewind the auxiliary tape recorder and play it back while obtaining digital data on one printer and selected channel X data on the other printer. Repeat 16 times, once for each column of channels.

(d) Correlate the data by counting back six sequences from the 320.83 cps time mark on the HS encoder data records for each sequence counted back from the 15.4 kc time mark on the LS encoder data records.

The telemeter data reduction system which was built in response to the proposed requirements outlined in the Appendix is documented in Reference 4. A circuit block diagram is shown in Figure A6.

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