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Technical Memorandum

DODGE SATELLITE POWER SYSTEM

BY L. WILSON

THE JOHNS HOPKINS UNIVERSITY • APPLIED PHYSICS LABORATORY

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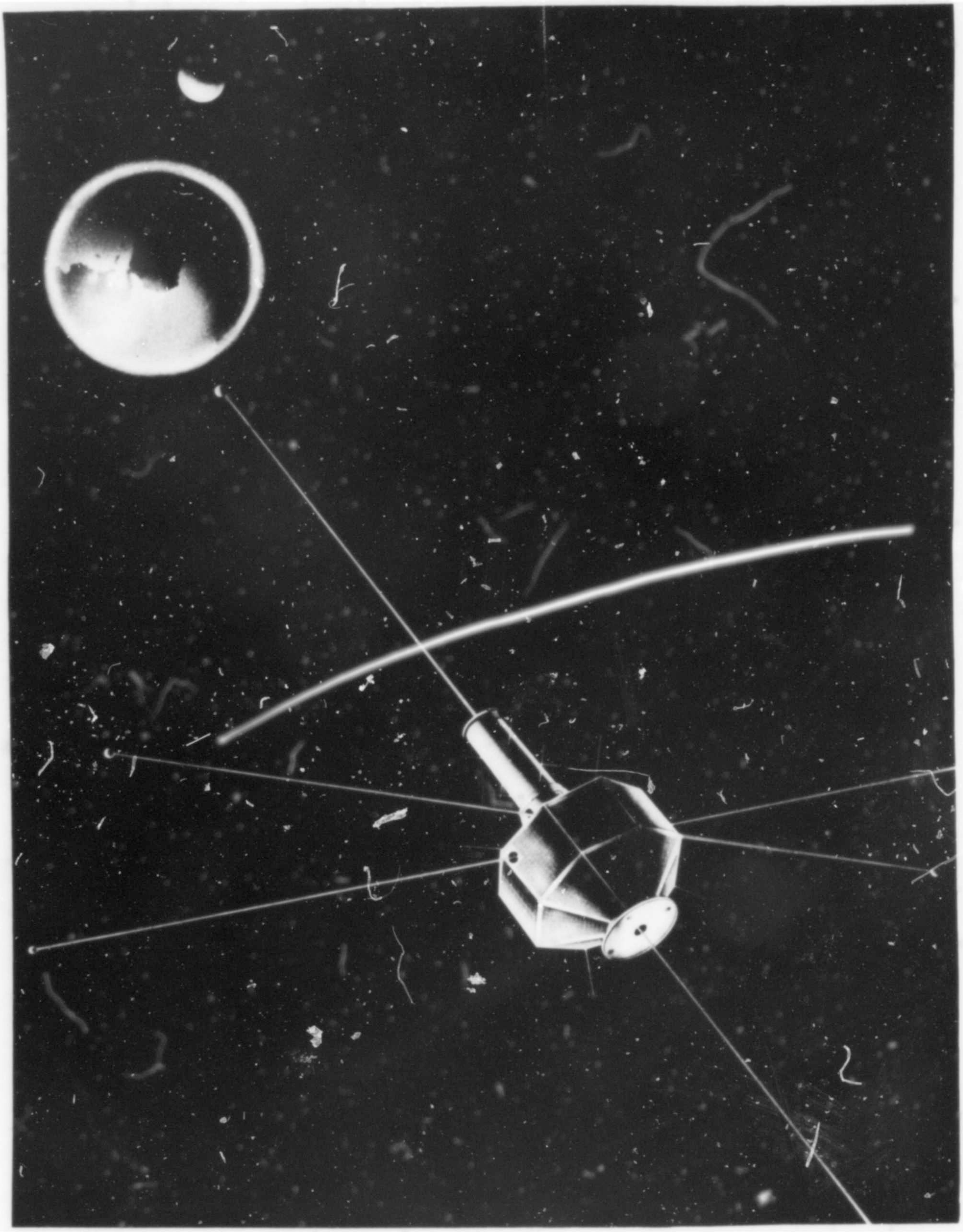

ARTIST'S CONCEPT OF DODGE SATELLITE

ABSTRACT

The DODGE (Department of Defense Gravity Experiment) satellite was launched on July 1, 1967 to an altitude of 18,000 statute miles into a near-synchronous orbit. The satellite was designed, developed, and built by the Applied Physics Laboratory as an experiment to demonstrate two- and three-axis gravity-gradient stabilization using magnetic and hysteresis damping techniques. A solar-cell/battery power system furnishes all of the electrical energy required to operate the satellite subsystems. The solar power generating capability is 45 watts (average). A 6-ampere-hour 10-volt nickel-cadmium battery operates peak electrical loads during light and dark orbits, as required. DC-to-AC inverters and DC-to-DC converters transform the solar/battery voltage to the proper levels required to operate the on-board electrical loads. The power system contains protective devices for safeguarding the battery against excessive discharge currents and charging voltages.

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I. INTRODUCTION

The design objectives of the DODGE power system are summarized as follows:

> To operate telemetry, television, transmitters, and attitude detection systems in an automatically programmed 1:6 duty-cycle mode,
>
> To provide continuous power to attitude control, doppler, and battery, as required, and
>
> To provide reset capability for achieving proper power load balance over the operating temperature range (+45 to 80°F) of the battery.

The power system contains conventional and mechanical parts proved flightworthy in previous APL-launched satellites. The performance of each component was specified and experimentally verified prior to its installation in the satellite. Also, to ensure compatibility and reliability, the entire power system was tested in thermal vacuum under specified simulated orbital conditions.

In-flight power data obtained during the first four months of orbit of the DODGE satellite indicate that the system is operating efficiently and is meeting the specified design objectives.

II. SUMMARY

The DODGE power system has been furnishing the necessary power to operate the satellite loads in all their programmed and manual modes.

During the first 120 days in orbit, the solar-cell array had experienced output degradation of about 20% of its initial capability. This was an expected degradation, however, and the solar array had been designed to include an excess power generating capability to compensate for it.

At the time of this writing the battery continues to produce stable end-of-charge voltages, currents, and temperatures. Analysis of the battery data indicates that the electrical capacity of the battery continues to remain high and that the battery is adequately charged during each charge cycle. This conclusion is deduced from the fact that the orbital data do not indicate any voltage degradation at nominal battery loads. The battery is operating within its design limits of +45 to 80°F. For short periods of time these limits are exceeded by about ±15°F, but this larger excursion is then reduced by commanding in solar power to heat the battery or by adjusting the power and loads to the satellite bus to permit the temperature to decrease to below +80°F. The command system is receiving uninterrupted power at levels which are in close agreement with design values.

III. POWER SYSTEM DESCRIPTION

The DODGE power system contains a solar power generating capability of 45 watts (average); a 6-ampere-hour, 10-volt hermetically-sealed nickel-cadmium battery for operating peak electrical loads during light and dark orbits; DC-to-DC converters; AC-to-DC inverters; and several safety devices and disconnects to minimize effects of excessive battery drain during discharge and of over-voltage during charge. The basic power elements and the DC power distribution of the satellite loads are schematically represented in Figs. 1 and 2. Each of these elements will be described briefly.

A. Solar-Cell Array

About seven thousand 2 x 2 cm solderless N/P silicon solar cells 14 mils thick are distributed on the satellite body to utilize effectively the gravity-gradient orientation. Twenty aluminum honeycomb panels coated with glass-fiber insulation serve as a skin for the satellite and as substrate for the solar array. The slant and equatorial sections of the satellite contain eight solar-cell panels each, and four solar-cell panels are contained in the polar sections. The total solar power of the satellite is divided into four separate and independent circuits, each wired in a series-parallel arrangement to enhance reliability. The solar-cell arrays making up these circuits are identified in Fig. 2 as: main, auxiliary 1, auxiliary 2, and command. The main solar-cell array is permanently connected to the satellite's power bus. On command, the auxiliary solar-cell arrays can be connected to the satellite's power bus to supply increased power to the bus, as required, for uninterrupted operation of on-board electrical loads or for automatic temperature-control circuitry.

The solar-cell arrays are designed to operate at a nominal voltage of 0.32 volt per cell. Each string in the

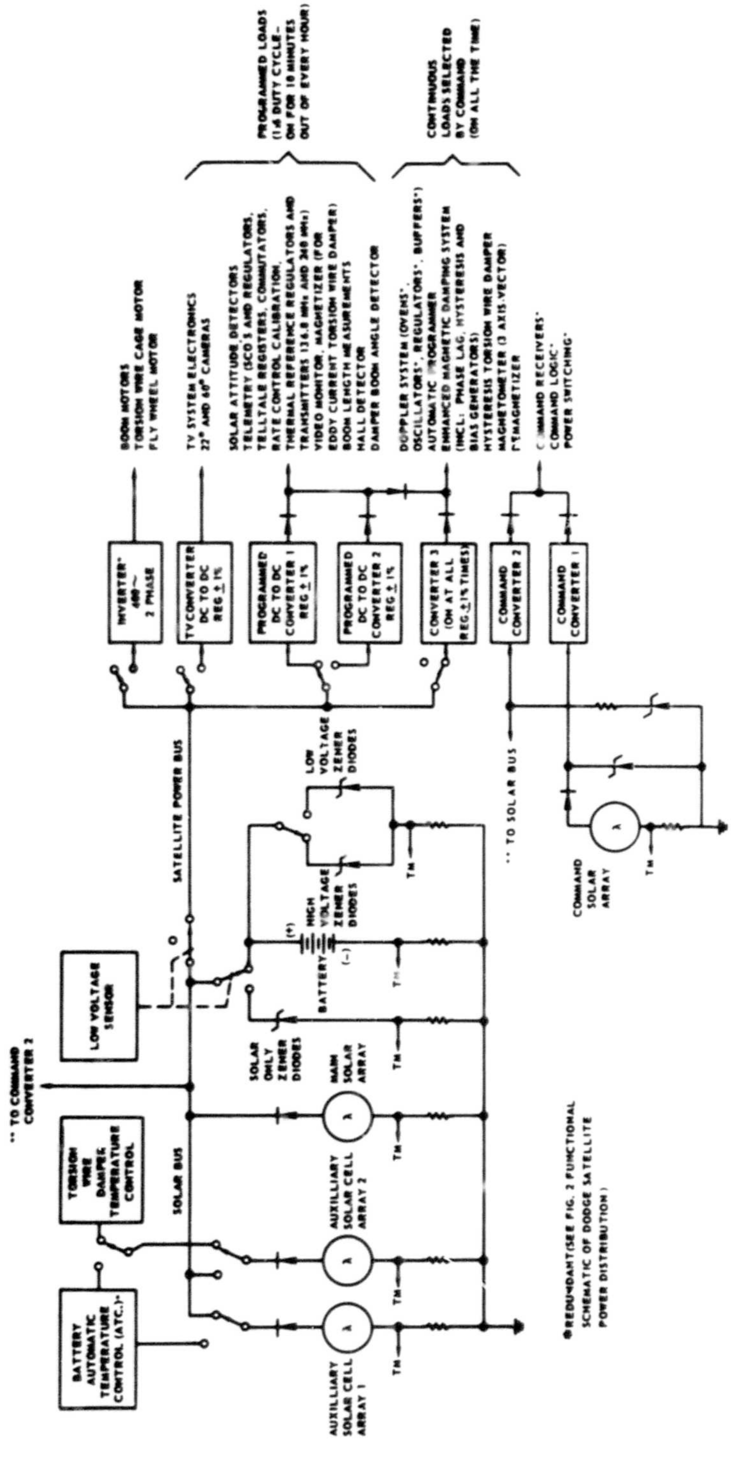


Fig. 1 BLOCK DIAGRAM OF SATELLITE ELECTRICAL SYSTEM

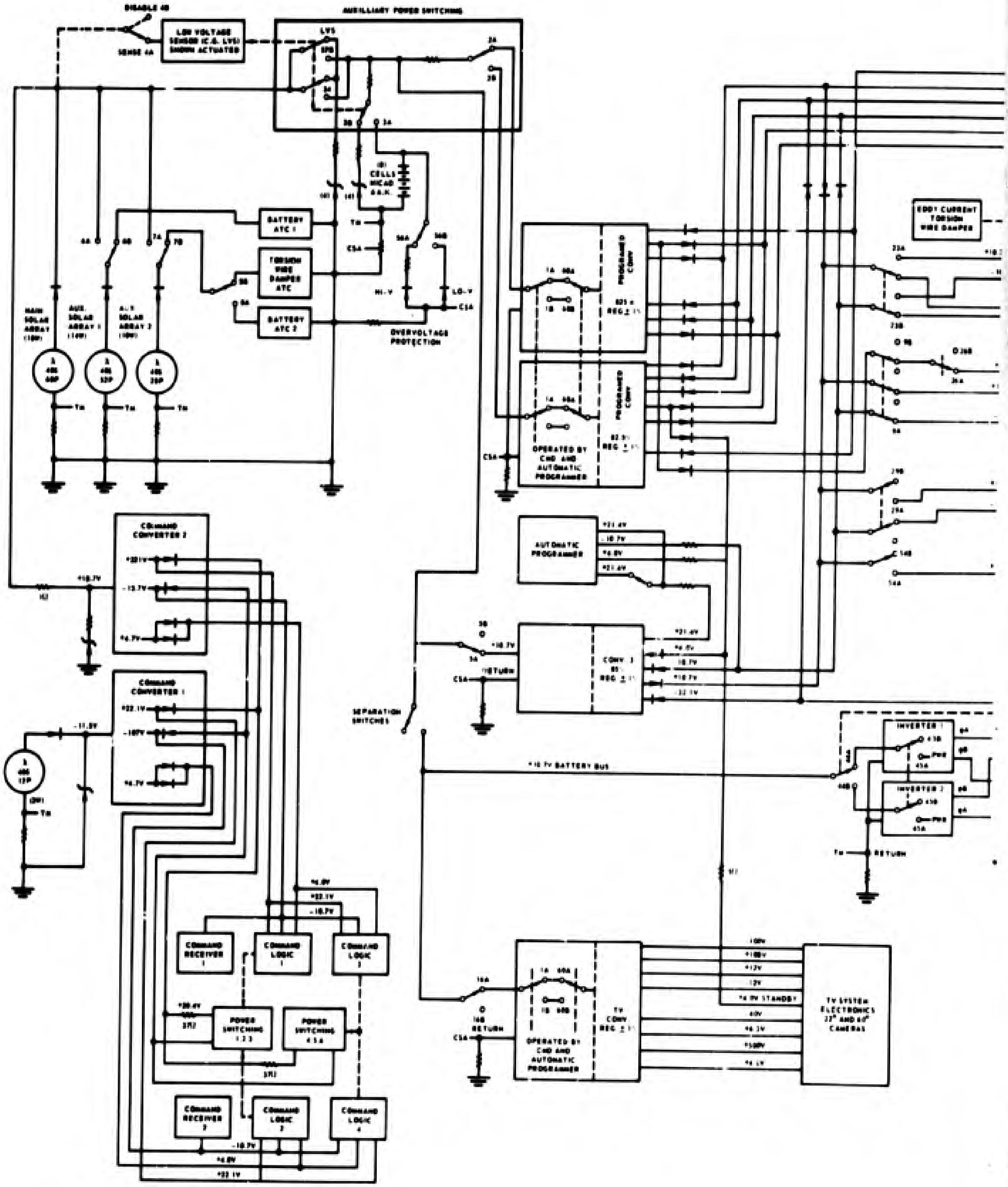


Fig. 2 FUNCTION

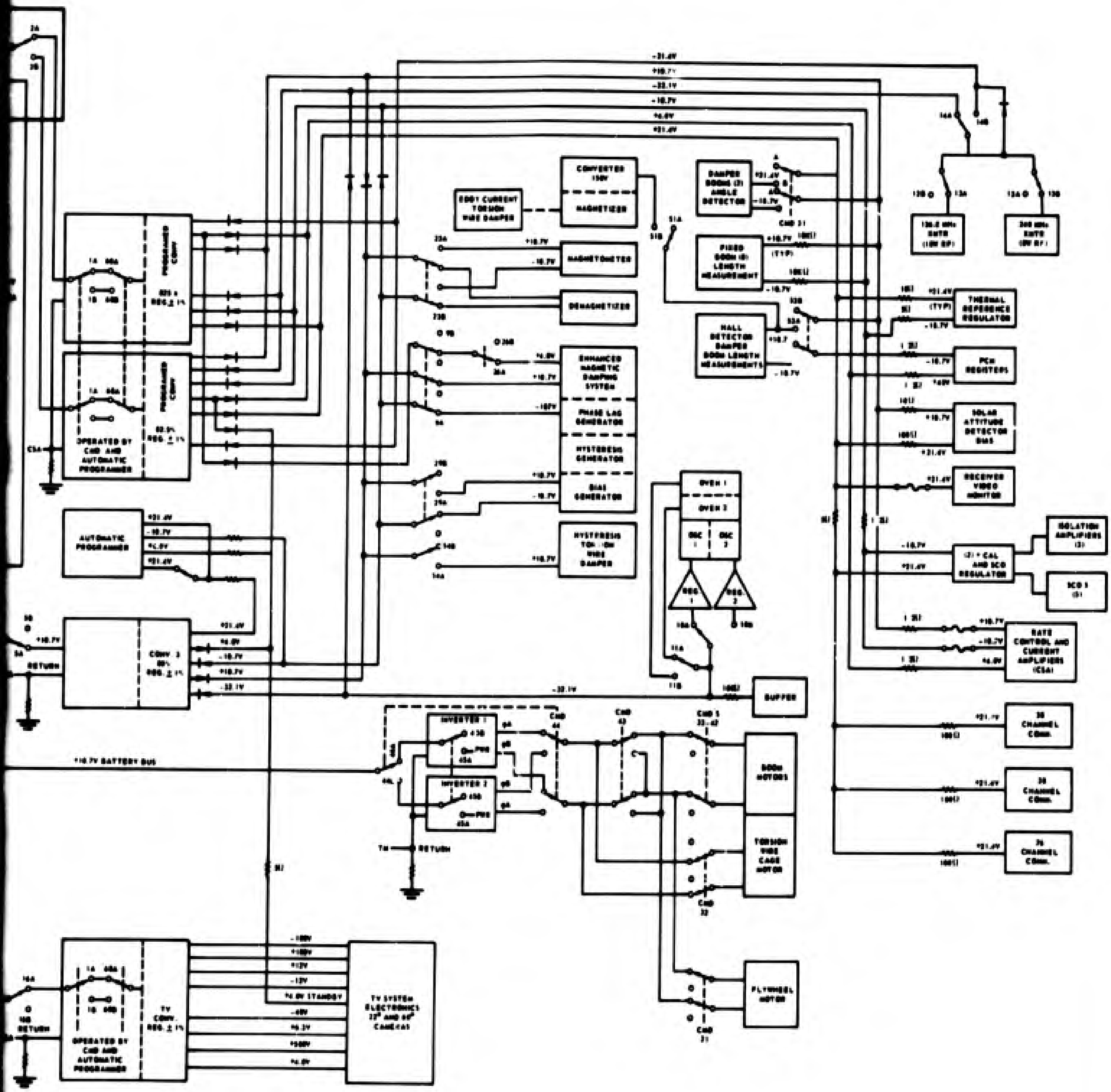


Fig. 2 FUNCTIONAL SCHEMATIC OF DODGE SATELLITE POWER DISTRIBUTION

array contains 40 cells electrically connected in series so as to produce a nominal 12.8 volts. The power capability of the various arrays is adjusted by paralleling a number of solar-cell strings. This redundancy feature significantly enhances the reliability of the array. The current output of the various arrays is a function of the attitude of the satellite with respect to the sun. The average solar-cell current and power output are plotted in Fig. 3 as a function of the angle (ψ) that the sun's rays make with the symmetry axis of the satellite. The approximate solar-cell power available to the battery and loads is indicated in Table I.

Table I

Initial* Power Available from Each Solar-Cell Array

Solar-Cell Array	Available Power (watts)
Main	18
Auxiliary 1	14
Auxiliary 2	10
Command	3

*Averaged over a 24-hour period.

B. Battery

The battery is composed of eight cylindrical, hermetically-sealed, nickel-cadmium cells arranged electrically in series so as to produce a nominal 10.7 volts at 77°F. Each battery cell is 1.3 inches in diameter and 3.5 inches long, and weighs 0.5 pound. A fully instrumented battery weighs 6 pounds and contains, in addition to its power cells, automatic temperature control circuitry, housings and flanges, zener diodes for voltage control, electrical leads, and connectors. A photograph of the physical arrangement of the cells and other electrical parts of the battery is shown in Fig. 4.

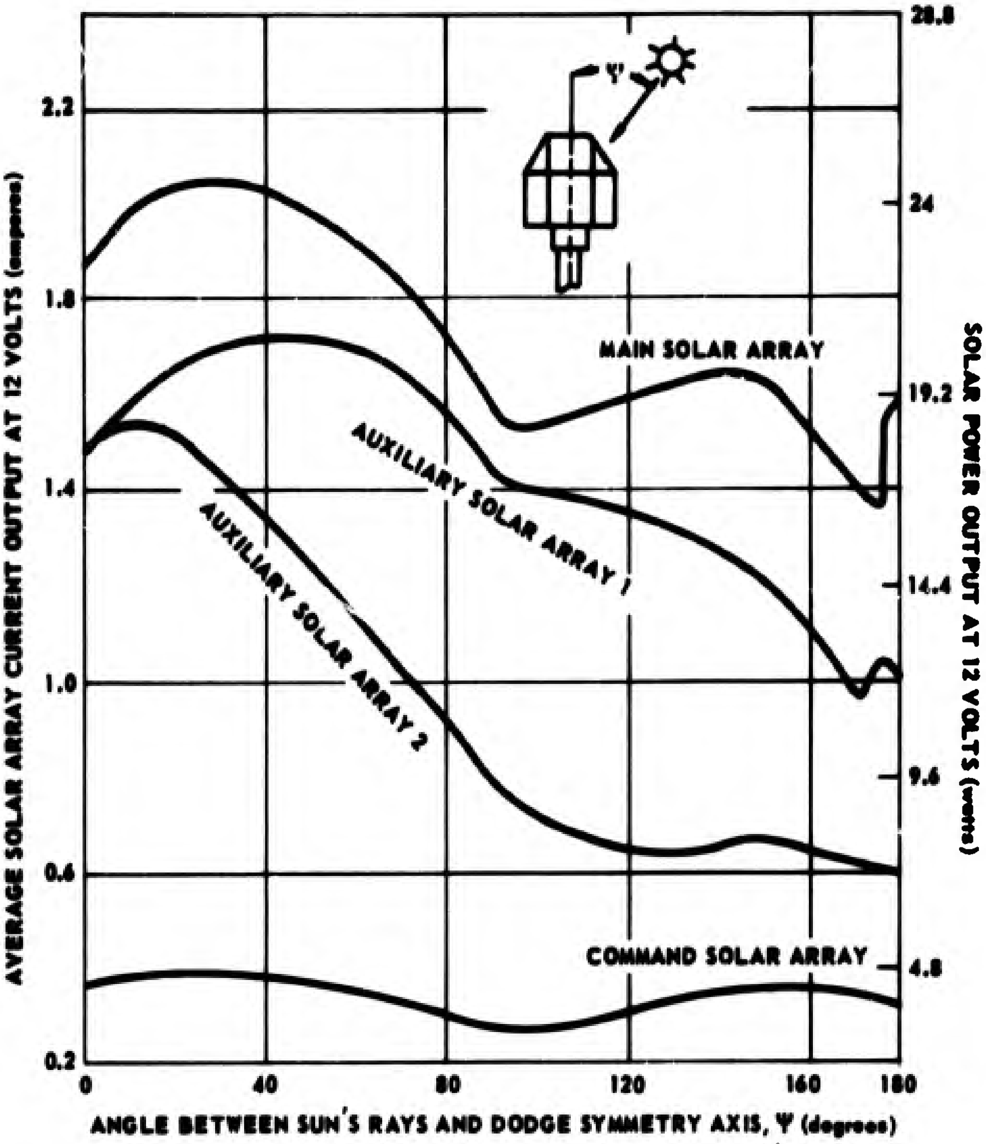
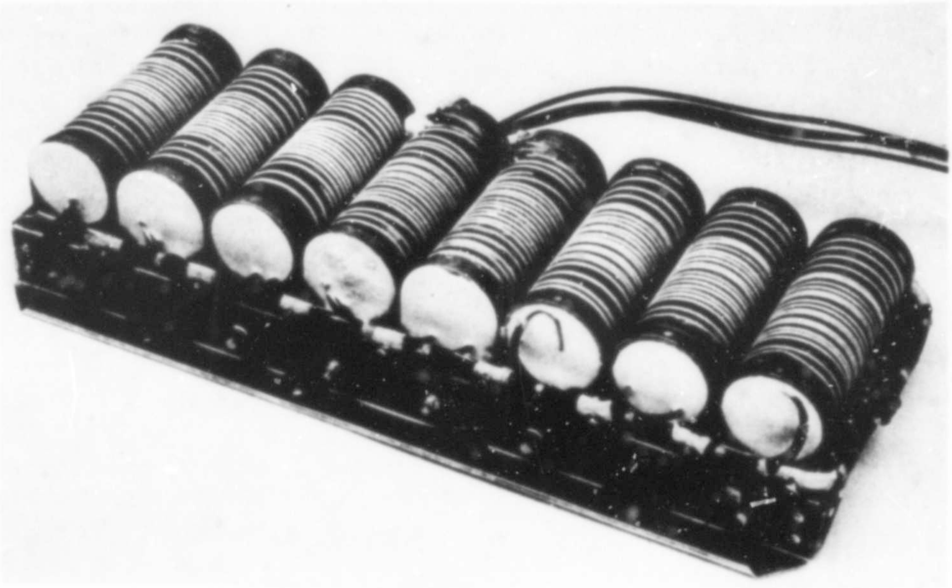
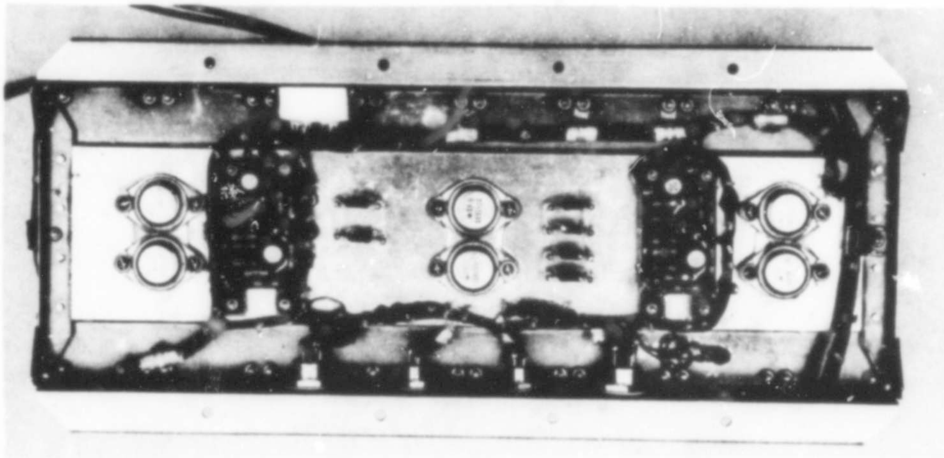


Fig. 3 PREFLIGHT DATA FOR DODGE SOLAR CELL ARRAY



TOP VIEW



BOTTOM VIEW

Fig. 4 DODGE SATELLITE BATTERY

The experimentally determined relationship of the total battery capacity of the discharge voltsge is summarized in Fig. 5. These values apply over the operating temperature range of +45 to 80°F. Four and one-half ampere-hours are delivered by the battery when it is discharged at a constant rate of 3 amperes to a voltage level of 8.8 volts. At a 6-ampere discharge rate the same battery delivers 3 ampere-hours. In the DODGE satellite, discharge of the battery is automatically discontinued when its terminal voltage decreases to 8.8 volts.

The battery is designed to undergo continuous charge/discharge cycles. Each discharge portion of the cycle extracts approximately 10% of the battery's available charge. The battery voltage is decreased to about 9.5 volts by the end of the discharge period. During the remaining portion of the cycle, the charge of the battery is replenished. Battery voltage during charge increases to about 12 volts, a value that varies somewhat with the ambient temperature.

There are two power-system components incorporated into the satellite to ensure that the battery voltage will stay within the limits of 8.8 to 12.5 volts. For the lower limit, a low-voltage sensor switch automatically disconnects the loads from the battery should the battery voltage decrease to 8.8 volts. The removal of the loads from the satellite bus and the connecting of the full complement of solar-cell arrays and the battery to the bus (on command) allows the battery to recharge. For the upper limit, several voltage zener diodes are connected electrically in parallel to the battery to bypass excess charging current that would otherwise tend to cause the battery to overvoltage. This action for voltage stabilization of the battery is continuous and automatic. By mounting the zener diodes in the same housing with the battery cells, good thermal coupling between these two components is achieved. The energy dissipated as heat through the zener diodes helps to stabilize the battery temperature.

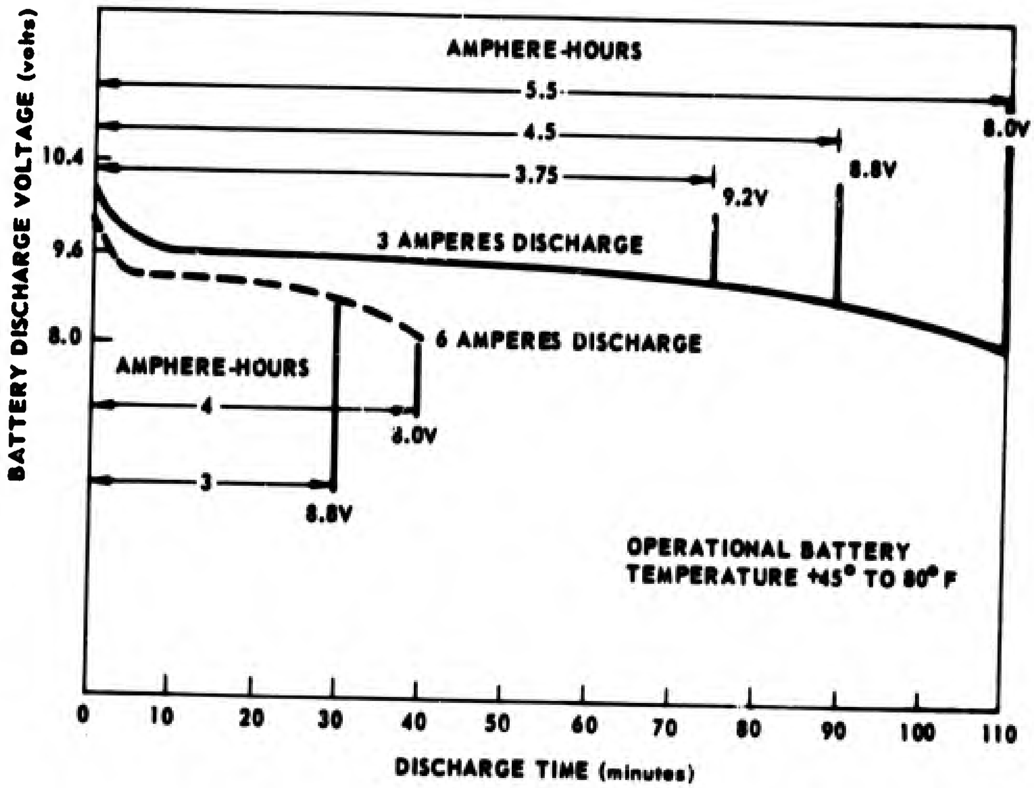


Fig. 5 BATTERY CAPACITY MEASUREMENTS

C. Converters and Inverters

The satellite contains seven DC-to-DC converters and two DC-to-AC inverters. The use of these converters and inverters in the power system and the voltage lines they furnish to their respective loads are shown schematically in Fig. 2. The programmed DC-to-DC converter furnishes five regulated voltages (±10.7, +4.0, -32.1, and +21.4 volts) to operate several programmed loads (telemetry, transmitter, and attitude-detection circuitry). An additional programmed converter may be made to perform these same functions on command, if it should be desired or required. Overall converter efficiency is approximately 85%. The satellite contains a separate converter (converter 3) to supply regulated voltages (±10.7, -32.1, and +21.4) and to continuously operate the attitude control equipment (i.e., magnetometer, demagnetizer, and damping systems), the automatic programmer, and doppler circuitry. The TV converter is designed to produce ±1% regulated voltages (±100, ±12, ±4, +500, and +6.3 volts) to operate the 22° and 60° cameras and electronic systems. Two independent command converters (one powered by the battery bus and the other by the 3-watt solar array) supply three voltages (+22.1, +10.7, and +4.7 volts) to the command loads (i.e., redundant receivers, logic, and power-switching circuitry). The 150-volt converter produces the high voltage used to charge a capacitor that, in turn, actuates the magnetizer coils. Either one of the two DC-to-AC inverters may be selected to supply power (2 phase, 400 cycles, 52 volts peak to peak) to the satellite booms, torsion wire, and flywheel motors.

D. Satellite Electrical Loads

The DODGE satellite contains an assortment of equipment and instruments designed to meet the mission objectives and to provide electrical information on the performance quality of the systems. Although about 100 watts are required to operate all of the satellite loads simultaneously, the use of all the on-board equipment at any one time is extremely unlikely; however the power system can sustain this level of operation for several minutes. In-flight and ground power measurements indicate that approximately 55 watts of power is sufficient to perform most of the satellite measurements and controls. Figure 2 provides detailed information on the satellite loads. A brief description of each load follows, and a summary of these loads is given in Table II.

1. Command System. The command system includes redundant receivers, TADEX (Tone Address Execute) logic, and power switching circuitry. The logic converts the command receiver audio outputs into signals suitable for controlling relays in the power switching matrix.

2. Telemetry System. The telemetry system contains two 38-channel commutators, one 76-channel commutator, modulation index control, calibration and SCO regulators, thermal reference regulators, telltale registers, rate control circuits, and 136.8- and 240-MHz transmitters. This system provides analog telemetry data of the electrical input and output of the satellite loads. Telemetry rate control furnishes clock signals for the commutators and telltale registers. The latter is a device that accepts electrical input information and produces a go/no-go indication of its input.

The 136.8- and 240-MHz transmitters produce 10 and 8 watts of RF power, respectively. The transmitters may be set to operate in high- and low-power modes. The 136.8- and 240-MHz transmitters require 21 and 26.5 watts, respectively, to operate in high-power mode. In the low-power mode each requires about 7 watts to operate.

3. Attitude Detection System. Two television cameras are used to sense changes in satellite attitude, and several solar aspect detectors determine the aspect of the sun relative to the spacecraft along three of its mutually perpendicular axes.

One TV camera provides a 60° x 60° field of view of the earth and the second camera provides a narrower field of view, 22° x 22°. By measuring the displacement of the

Table II

Summary of DODGE Electrical Loads

Loads	Operating Power (watts)
Command	0.75
Telltale function and telemetry (excluding transmitters)	3.3
136.8-MHz transmitter	21.0 max. 6.5 min.
240-MHz transmitter	26.5 max. 7.5 min.
TV cameras	8.6
Flywheel operation	8.0
Two-boom operation	13.5
Enhanced magnetic damping (including magnetometer)	23.0 max. 4.2 min.
Hysteresis and eddy-current torsion-wire dampers, Hall and damper boom-angle detectors	5.8
Low-voltage sensor, programmer, auxiliary power switching, magnetizer, demagnetizer, video monitor and solar-attitude detectors	2.0
Doppler components (i.e., oscillators, ovens, regulators, buffer)	1.3

earth's image from the center of the TV picture (as taken by the 22° and 60° cameras) one may determine the pitch and roll angles of the satellite. During programmed operation, the TV system is turned on for 10 minutes out of every hour. Various sequences of TV operation are described by Mr. T. Thomson in his publication "The DODGE Television System," APL Technical Digest, May-June 1967. Briefly, the 10-minute operational period consists of a 200-second charge conditioning cycle followed by exposure of the 60° camera, a 200-second read cycle, exposure, and a 200-second read cycle of the 22° camera. At the end of this sequence the programmer automatically disconnects the TV system from the satellite bus. The TV DC-to-DC converter requires 9.0 watts of power from the battery to perform its function.

Solar attitude detectors detect the satellite yaw. They are mounted in both directions along each of the three principal axes of the satellite, and they produce an output that is proportional to the cosine of the angle of incidence. Circular solar cells housed inside a cylindrical well are employed to produce outputs that vary linearly with the angle of incidence from 0 to 45°. Two additional solar cells mounted behind narrow circular apertures provide calibration information that is used to correct the outputs of the linear and cosine detectors. The detectors require +10.7 and +21.4 volts as bias potentials only: they require virtually no power to operate.

4. Attitude Control System. The DODGE satellite contains a series of motorized booms with attached end masses and several damping schemes, all geared to achieve and to maintain attitude stabilization. AC hysteresis synchronous motors retract or extend single booms or oppositely paired booms precise distances from the satellite. Provision is made in the satellite design for measuring boom length. The boom motors are powered from battery-operated inverters, the output of which is 52 volts peak to peak, 400 cycles, 2 phase. About 13.5 watts are required from the battery to operate two booms simultaneously. The eddy-current torsion-wire damper

provides a range of damping to suit various boom lengths and configurations. A vane of copper moves between the poles of a horseshoe alnico magnet. The latter may be charged to different levels of magnetization. The vane is fixed to the gimballed structure of the damper boom and the magnets are mounted to the satellite structure. This damper requires +10.7 volts and about 0.5 watt.

The hysteresis torsion-wire damper receives regulated ±10.7 volts from converter 3 and consumes a maximum of 3.5 watts to perform its damping function.

The satellite contains an 8000-rpm flywheel, the spin axis of which is aligned with the x-axis of the satellite. The flywheel motor receives power from the inverter (400 cycle, 2 phase) and requires 8 watts. The flywheel is designed to provide additional yaw stability for the satellite.

5. **Enhanced Magnetic System.** The enhanced magnetic system (Fig. 2) provides the magnetic damping capability for the satellite. The system receives the required regulated voltages from converter 3. It may be turned off on command when desired or when required as in the case of power shorts. Additional command functions are available in the satellite to vary the output signal voltages as required by the satellite's electromagnetic coils in order to neutralize the magnetic field of the earth as sensed by the three-axis vector magnetometer. The voltage and power requirements of the enhanced magnetic system and associated equipment are summarized in Table III.

The low-voltage sensor is a unidirectional electronic unijunction oscillator that oscillates only after a given bias voltage is removed. When the battery voltage decreases to 8.8 volts (due to excessive usage or to load failure), a 10-millisecond pulse actuates power switching. The latter disconnects the battery from all loads. The battery then remains open-circuited until it is manually reset by ground command.

Table III

Ground Electrical Measurements of the Enhanced Magnetic Damping System

Magnetic Component	Function	Voltage Required from Converter 3 (volts)	Approximate Power Required to Operate (watts)
Three-axis vector magnetometer	To produce three electric signals that are proportional to the three orthogonal components of the earth's field.	±10.7	2.3
Transconductance amplifier	To amplify voltages produced by the various parts of the enhanced magnetic system.	±10.7	16.0 max. 0.5 min.
Hysteresis generator	To sense effects of hysteresis by moving hysteresis material in a magnetic field. (A Hall detector is used.)	±10.7 + 4.0	3.0 max. 0.2 min.
Phase-lag generator	To receive three DC voltages from the magnetometer, to sample and hold this information according to command, and then to produce fixed voltage signals for the satellite's electromagnetic coils.	±10.7 + 4.0	2.1 max. 1.2 min.
Bias generator	To supply preselected (by command) fixed voltages that are used by the magnetic coils to produce fixed magnetic dipoles. This action will tend to counteract satellite yaw.	±10.7	negligible
Electromagnetic coils	To produce magnetic dipoles (receives power from a 150-volt DC converter).	+150	1.5 ave.

The programmer automatically positions command 1 (Fig. 2) to the on-mode for 10 minutes out of every hour. During the on time, the programmer interrupts telemetry every 200 seconds by a time burst 390 milliseconds in duration. At the end of the third time burst, the programmer switches the satellite to OFF position.

The doppler components provide the tracking capability for the satellite. In-flight measurements indicate that the satellite has achieved an altitude of about 18,000 miles with an orbital period of about 22 hours. The doppler electronics (oscillators, ovens, regulators, and buffer) require -32 volts from the standby converter and 1.3 watts from the battery.

E. Operational Modes

The satellite may be operated in automatically programmed, manual, and solar-only modes.

1. Automatic Modes. The DODGE power system is designed to operate primarily in automatically programmed modes. Selected loads (i.e., on-board programmer, doppler, and attitude control) receive continuous power from converter 3 (Fig. 2), and are normally turned on at all times. Programmed loads, that is, attitude detection, telemetry (including transmitters), television cameras, and electronics operate only for 10 minutes out of every hour. The on-board programmer automatically controls the on and off times of these loads.

Table IV lists the power and load requirements for the six principal programmed modes. The DODGE satellite was tested in a thermal-vacuum environment to the power conditions specified for most of these modes. Thermal and electrical power-load balance was achieved for all cases tested. Figure 6 (a through e) illustrates data obtained while the satellite was in the thermal-vacuum environment and the power system was programmed to operate according to mode 6 (Table IV). The conditions simulated are those experienced by the spacecraft during maximum eclipse at vernal equinox. The test assumed that the satellite is

Table IV

Examples of Automatically Programmed Modes

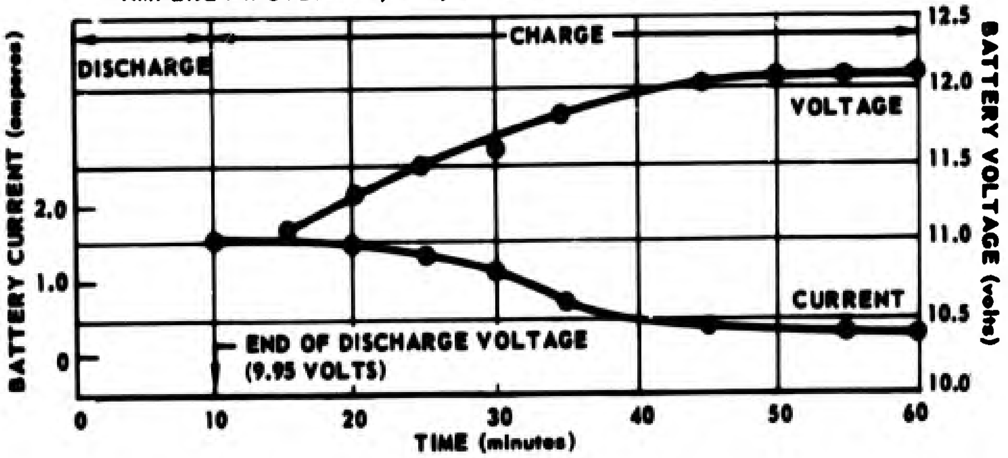
Mode	Mode Function*	Power (watts) Programmed Loads**	Continuous Loads***	Equipment**
1	Magnetic despin	51	8.0	Auxiliary solar array 1 on battery automatic temperature control (ATC), auxiliary solar array 2 on satellite bus, high-voltage zener diodes, telemetry, 240-MHz transmitter (high-power state), television, solar-attitude detectors, receiver video monitor, doppler, programmer, magnetometer, transconductance amplifier, command.
2	Magnetic despin with hysteresis generator	65	19.5	Same as Mode 1 loads except that auxiliary solar array 1 is on satellite bus, auxiliary solar array 2 is on battery ATC, and hysteresis generator is turned on.
3	Magnetic stabilization	64.5	16.5	Same as Mode 2 loads except that bias generator is turned on.
4	Gravity stabilization using magnetic timing damping	59.5	11.5	Same as Mode 2 loads except that bias generator and phase-lag generator are turned on and the hysteresis generator is turned off.
5	Gravity stabilization using torsion-wire and eddy-current damping	54	5.5	Auxiliary solar array 1 on battery ATC, auxiliary solar array 2 on satellite bus, high-voltage zener diodes, telemetry, 240-MHz transmitter (high-power state), television, solar-attitude detectors, receiver video monitor, damper boom angle and Hall detectors, eddy-current torsion-wire damper, doppler, command.
6	Gravity stabilization using hysteresis torsion-wire damper	56	8.0	Same as Mode 5 loads except that hysteresis torsion-wire damper is turned on and the Hall detector and the eddy-current torsion-wire damper are turned off.

*Function name indicates attitude control objective.

**Average power required to operate DODGE loads for 10 minutes out of every hour. These loads (listed in last column) are equal to the sum of selected programmed and continuous loads. Programmed loads are underlined for easy identification.

***Average power required to operate continuous loads of DODGE for 50 minutes out of every hour. During this time the programmed loads (underlined in last column) are turned off.

(a) TEST DATE: 11 APRIL 1967
POWER SYSTEM PERFORMANCE (MEASURED): SIMULATED INPUT TO SATELLITE BUS, 2.24/2.25 AMPERES; BATTERY TEMPERATURE, +64°F; 10-MINUTE LOAD, 5.7 AMPERES; 50-MINUTE LOAD, 0.73 AMPERE; AMPERE-MINUTES OUT, 34.5; AMPERE-MINUTES IN, 43.6



(b) TEST DATE: 11 APRIL 1967
BATTERY VOLTAGE DISTRIBUTION BEFORE, DURING, AND AFTER ECLIPSE IS SHOWN. POWER SYSTEM PERFORMANCE: SOLAR INPUT CURRENT, DURING ECLIPSE, 0; NOMINAL SOLAR INPUT CURRENT, 2.15 AMPERES; BATTERY TEMPERATURE (MINIMUM), +36.3°F; 10-MINUTE LOAD, 5.80 AMPERES; 50-MINUTE LOAD, 0.70 AMPERE; AMPERE-MINUTES OUT (DURING ECLIPSE), 98.5; AMPERE-MINUTES IN (DURING ECLIPSE), 0

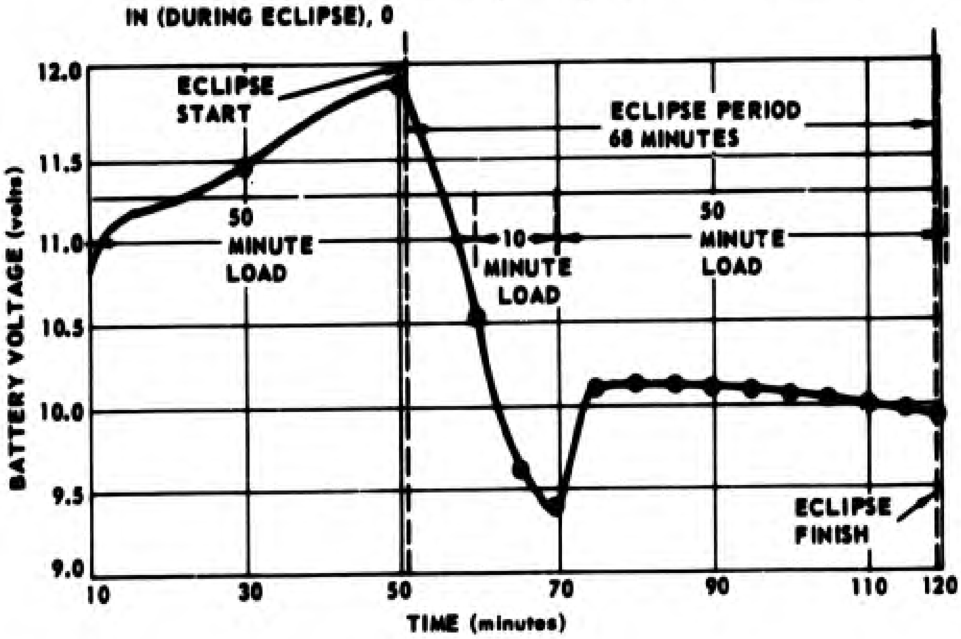
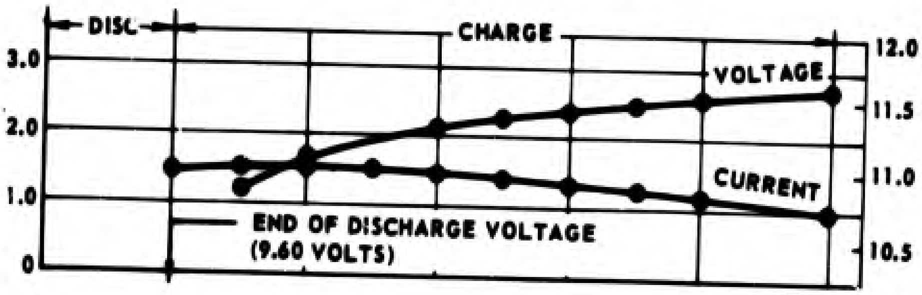
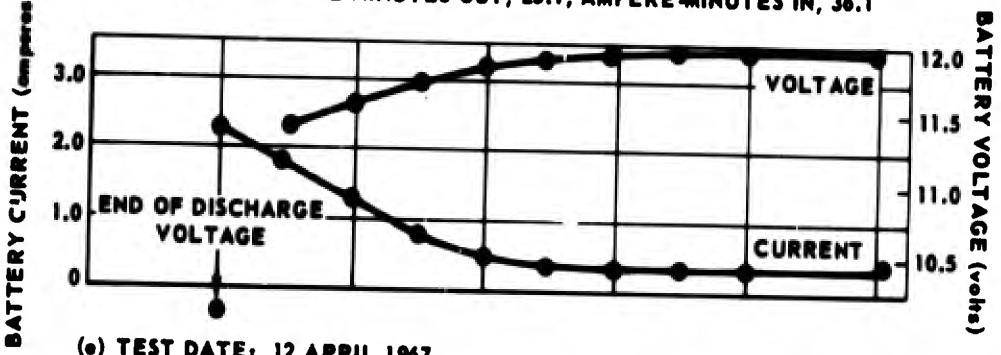


Fig. 6 THERMAL-VACUUM TEST FOR GRAVITY-GRADIENT STABILIZATION (VERNAL EQUINOX). DATA ARE TAKEN FOR OPERATIONAL MODE 6 (TABLE IV)

(c) TEST DATE: 11 APRIL 1967
POWER SYSTEM PERFORMANCE (MEASURED): SIMULATED INPUT TO SATELLITE BUS, 2.20/2.25 AMPERES; BATTERY TEMPERATURE, +40 TO 46° F; 10-MINUTE LOAD, 5.7 AMPERES; 50-MINUTE LOAD, 0.75 AMPERE; AMPERE-MINUTES OUT, 35.2; AMPERE-MINUTES IN, 67 (BATTERY NOT FULLY CHARGED 2 HOURS AFTER ECLIPSE)



(d) TEST DATE: 12 APRIL 1967
POWER SYSTEM PERFORMANCE (MEASURED): SIMULATED INPUT TO SATELLITE BUS, 3.03/3.25 AMPERES; BATTERY TEMPERATURE, +62 TO 65° F; 10-MINUTE LOAD, 5.50 AMPERES; 50-MINUTE LOAD, 0.70 AMPERE: AMPERE-MINUTES OUT, 25.1; AMPERE-MINUTES IN, 36.1



(e) TEST DATE: 12 APRIL 1967
POWER SYSTEM PERFORMANCE (MEASURED): SIMULATED INPUT TO SATELLITE BUS, 2.17/2.50 AMPERES; BATTERY TEMPERATURE, +70° F; 10-MINUTE LOAD, 5.7 AMPERES; 50-MINUTE LOAD, 0.70 AMPERE; AMPERE-MINUTES OUT, 33.2; AMPERE-MINUTES IN, 40.5

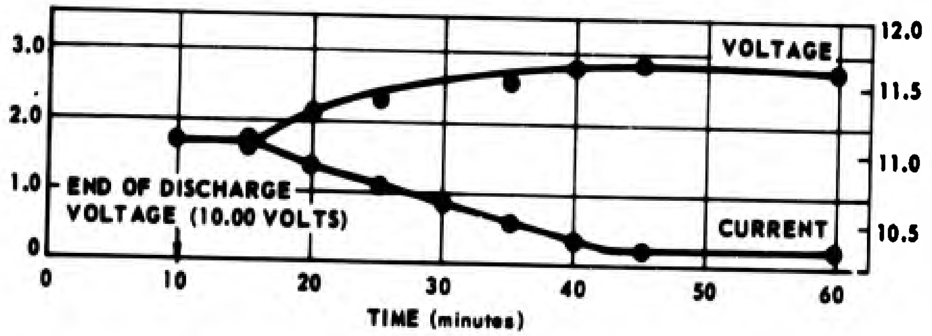


Fig. 6 (cont.)

three-axis gravity-gradient stabilized by use of hysteresis torsion-wire damping. The satellite was operated continuously for 22 hours. Loads are listed in the last column of Table IV. The power normally produced by the main and auxiliary 2 solar-cell arrays is simulated in this experiment by automatically varying (with time) the current output of a power supply. A simulated current-time curve of the auxiliary 1 solar-cell array was used to power the automatic temperature control of the battery. An analysis of the results obtained for these test cases indicates the following:

1. The power-load profile balance achieved throughout the entire 22-hour simulated orbit is very satisfactory.

2. The average battery capacity ratio (ampere-hours in/ampere-hours out) is greater than 1.2, an operational criterion known to be "healthy" and safe for the solar-battery power system.

3. The battery temperature decreased to about +35°F toward the end of the eclipse period. (No overvoltage was observed at this time or during the charging period that followed.)

4. By the time the battery reached its full-charge state, the temperature of the battery had risen to about +45°F, accompanied by a diminution of overcharge current to acceptable values.

5. The minimum value of the end-of-discharge voltage for the battery during the eclipse period was 9.3 volts. (This value is well above the low limit (8.8 volts) specified for the DODGE battery.)

2. Manual Mode. In some attitude experiments it is desirable to operate the satellite continuously. This is accomplished by commanding the programmer to OFF and the preselected electrical loads to ON. The satellite was

operated under manual control several times during the first 120 days of orbit. In-orbit data indicate that the battery will deliver 1 ampere of current for at least 4 hours without developing low voltages. The power delivered is approximately 9.5 watts. Care must be exercised in manual operation to maintain a proper balance between the power available and the loads selected. Battery temperature excursions should be limited to the +45 to 80°F range, and the number of operations that require the battery to deliver more than 20% of its nominal capacity must be kept to an absolute minimum if long battery life is to be achieved.

3. Solar-Only Mode. It may be desired, and in the case of battery failure it is required, to operate with the battery circuit open. The maximum power available is limited in this operation to the output capability of the main, auxiliary 1, and auxiliary 2 solar-cell arrays. It is estimated that most attitude experiments can be operated for about 50% of the time in this mode.

III. IN-ORBIT DODGE POWER DATA

Because the DODGE satellite is in a near-synchronous orbit, it is within "visible" range of APL for intervals of approximately four days at a time; these periods of visibility are listed in Table V. During the periods of visibility

Table V

Periods when DODGE Satellite was "visible at APL Tracking Station

From		To	
Zulu Time	Day	Zulu Time	Day
1450	182 (1 July 1967)	1400	185 (4 July 1967)
0600	192	1800	196
0200	203	0000	208
0900	214	1000	219
1300	225	1130	230
0230	237	1430	241
0430	248	0330	253
0830	259	0630	264
1130	270	0930	275
0130	282	1330	286
0330	293	0230	298
0630	304	0530	309
1930	315	0730	320
0030	327	2130	331

from APL, experimental studies of the attitude of the satellite for various magnetic, damping, and boom inputs are conducted. In-orbit power data is telemetered, recorded, and analyzed continuously. Included in this study are the temperature and current output of the solar array and the voltage, current, and temperature of the battery while it is delivering energy to the satellite loads. The in-orbit power data summarized in this paper provide information on the performance capability of the power system. Sample performance data on the solar-cell array and battery are examined.

1. Solar-Cell Array. Figure 7 presents data taken on the current output distribution of the main and command solar-cell arrays for various values of ψ (the angle the sun's rays make with the symmetry axis of the satellite). After about 120 days in orbit, the solar-cell arrays produced about 20% less current than they did immediately after launch. The temperature excursion of the solar-array panels ranged from about -125°F to +230°F. Effects of shadowing for $\psi > 90°$, of temperature variations, of solder tab disconnects, and of degradation due to natural particle radiation contribute to the decrease in output of the solar-cell arrays.

2. Battery. While the satellite is operated in its automatically programmed mode, the battery provides the necessary operating power for 10 minutes out of every hour. Battery data taken during this discharge period is telemetered and permanently recorded by the APL tracking station. During the 50-minute remaining time, the solar-cell arrays recharge the battery. No battery charge data is available for analysis because the transmitters are turned off during the charge period. However, voltage zener diodes electrically shunting the battery automatically drain off excess current from the solar cells, thus preventing the battery from overvoltaging while charging.

For most of the operating time, either the auxiliary 1 or the auxiliary 2 solar-cell array is used to provide heating energy to the battery. The nominal operating set temperature of the battery is between +60 and +65°F. Temperature excursions between +45 to 80°F for nominal charging rates will not damage or shorten the life of the battery. Typical data on the end-of-discharge values and

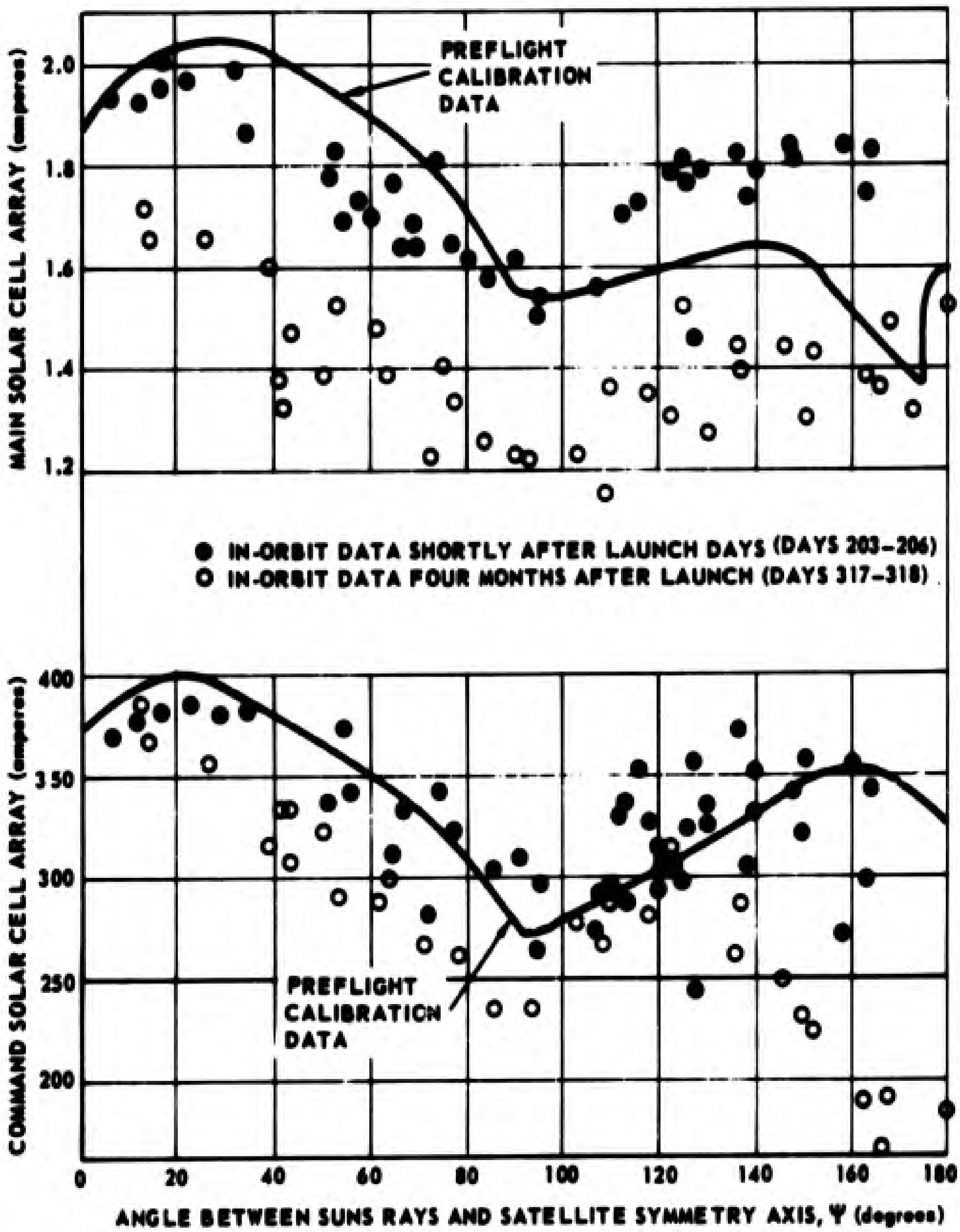


Fig. 7 IN-ORBIT DATA FOR SOLAR CELL ARRAYS

the temperature distribution of the battery are presented in Figs. 8, 9, and 10. These data indicate electrical constants that are within the stipulated design limit. The discharge voltage is within the voltage range of 8.8 to 12.5 volts while the battery is delivering currents from about 100 milliamperes to 4.5 amperes. These values correspond to the extraction of 1.0 to 45 ampere-minutes.

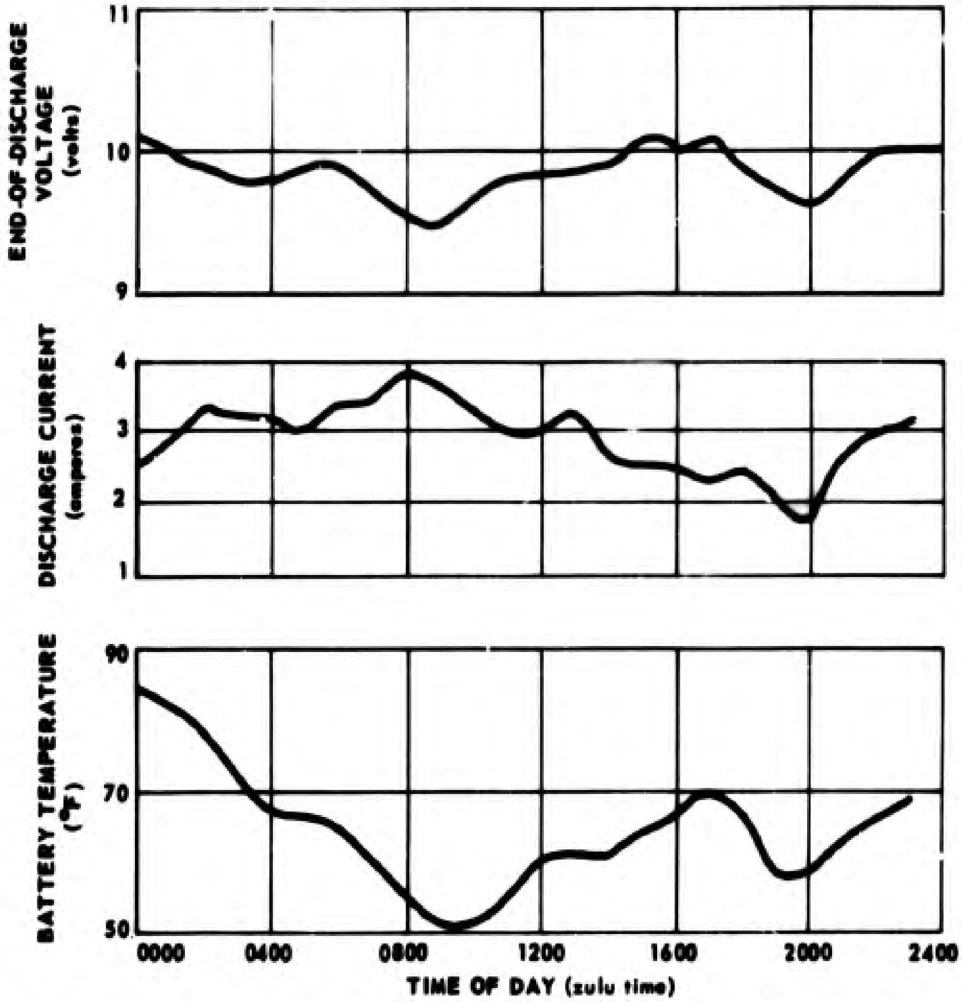


Fig. 8 IN-ORBIT DATA FOR BATTERY (DAY 204–1967)

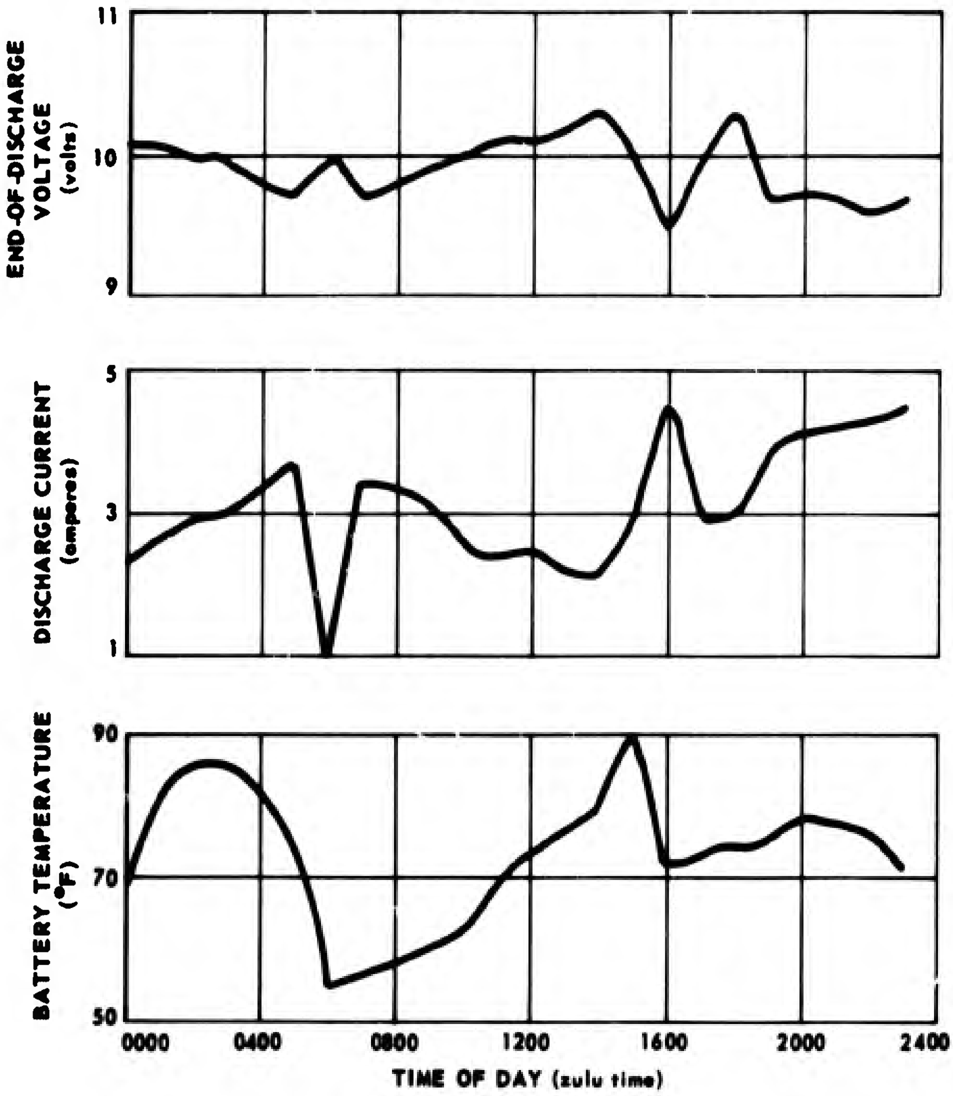


Fig. 9 IN-ORBIT DATA FOR BATTERY (DAY 238–1967)

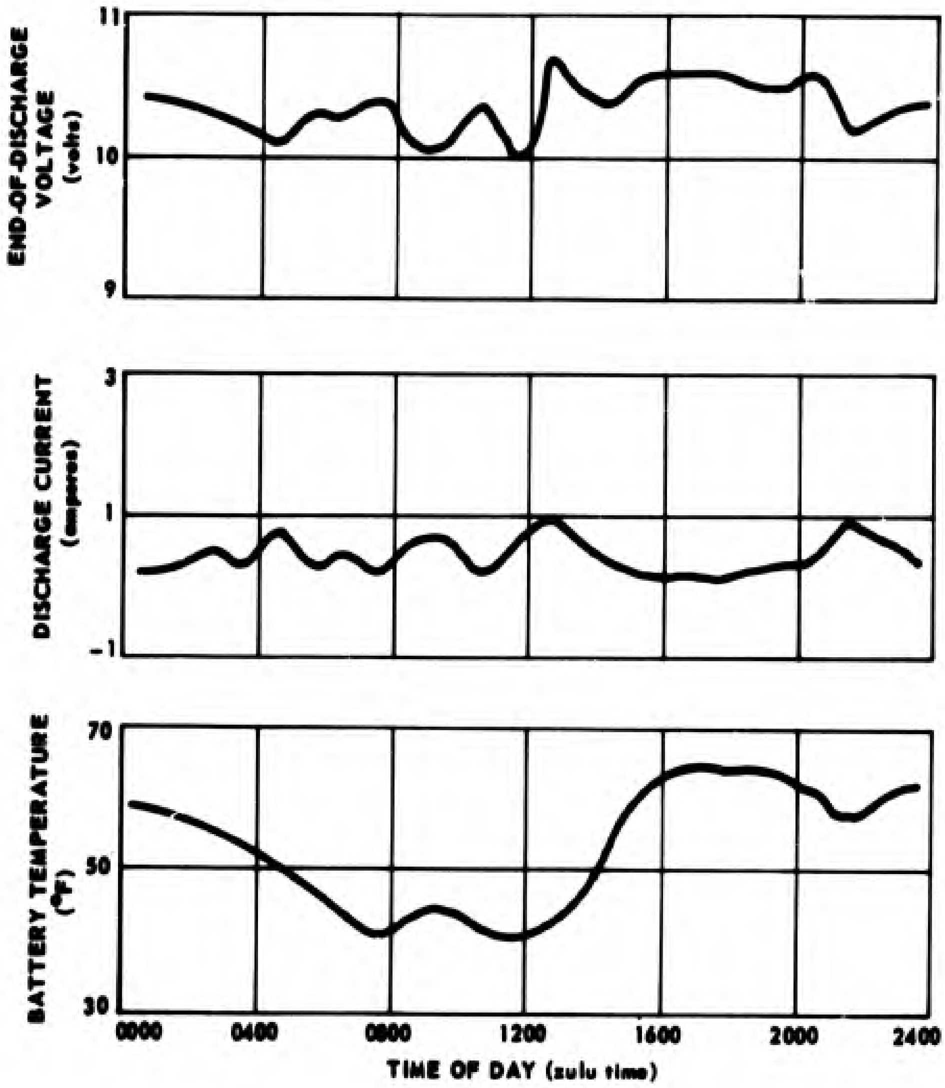


Fig. 10 IN-ORBIT DATA FOR BATTERY (DAY 318–1967)

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13. ABSTRACT

The DODGE (Department of Defense Gravity Experiment) satellite was launched on July 1, 1967 to an altitude of 18,000 statute miles into a near-synchronous orbit. The satellite was designed, developed, and built by the Applied Physics Laboratory as an experiment to demonstrate two- and three-axis gravity-gradient stabilization using magnetic and hysteresis damping techniques. A solar-cell/battery power system furnishes all of the electrical energy required to operate the satellite subsystems. The solar power generating capability is 45 watts (average). A 6-ampere-hour 10-volt nickel-cadmium battery operates peak electrical loads during light and dark orbits, as required. DC-to-AC inverters and DC-to-DC converters transform the solar/battery voltage to the proper levels required to operate the on-board electrical loads. The power system contains protective devices for safeguarding the battery against excessive discharge currents and charging voltages.

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14. KEY WORDS
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