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Miniguide is an extremely simple attitude control concept for spin-stabilized vehicles. It utilizes body-fixed horizon detectors as an absolute attitude reference, and a body-fixed reaction jet to provide control torque. The spin which stabilizes the vehicle in space also provides the scan for the attitude reference, and allows only a single control jet to provide all the control torque for two axes, and the required stability augmentation or damping torque.

The control logic and electronics requirements for the system are minimal. A flight-tested version to be described later in this paper used a total of only 13 transistors and two moving parts - a rate gyro and a solenoid valve. The total weight of this tested system was under 18 pounds including tankage and nitrogen, and with minor refinements of design could easily be reduced below 12 pounds.

This paper will describe the following:

a. Basic miniguide control theory for a pitch-attitude and a vertical-attitude control scheme

b. Flight test of the pitch control scheme

c. Studies of two applications of the Miniguide now under consideration

The vertical control, shown diagrammatically in figure 1 uses a horizon detecting telescope with a field of view several degrees wide. This telescope is set at an angle with respect to the spin axis which, when the spin axis is vertical and the vehicle at the correct altitude, allows the field of view to

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sweep around the edge of the apparent earth disk. In this condition, figure 2(a), the portion of the field of view occupied by the earth remains the same throughout the revolution of the vehicle. If the spin axis of the vehicle is not vertical, the portion of the field of view occupied by the earth will change as the vehicle spins, as shown in figure 2(b). The output of the telescope then will be a varying signal which is greatest when the telescope sees the most earth, and least when the telescope sees the least earth. This signal is then suitably amplified and fed to an electronic switch, figure 1. Whenever there is an a-c signal component present, the electronic switch will be activated during the portion of the signal when the telescope sees the most earth.

The output of the electronic switch operates a body-fixed reaction jet positioned on the body so that when actuated it torques the vehicle in such a way as to move the telescope off the earth. Since the spinning vehicle is acting as a gyroscope, this jet is positioned on the body at right angles to the telescope to precess the vehicle in the telescope plane. The system will thus null out the a-c component of the telescope signal, leaving the vehicle spin axis vertical.

Due to the lack of damping in the space environment, a coning motion will be built up during the control phase and the system may be dynamically unstable due to system time lags. To compensate for this, a stability augmentation signal is obtained from a rate gyro, and summed with the telescope signal. This gyro is placed with its input axis parallel to the axis about which the control jet produces torque. Since there is an energy interchange between the body axis each revolution, damping one axis is sufficient to damp the entire system, and only the single rate gyro and reaction jet are required.

The horizontal version of the pitch control, shown in figure 3, uses two body-fixed horizon detectors with narrow fields of view mounted on opposite

sides of the vehicle spin axis. With the spin axis horizontal, and at the operating altitude, the telescopes are set at an angle with the vehicle spin axis so that they produce a conical scan which just intersects the earth horizon as shown in figure 4(a). If the vehicle is in a nose-up position, the forward telescope scan will be completely off the earth, and the rear telescope scan will dip further into the earth, as shown in figure 4(b). In a nose-down attitude, the converse is true.

With the narrow field of view of the telescope, the output signals are essentially off-on signals - off when the telescope sees space, on when it sees the earth. These signals are suitably amplified and operate a switching amplifier which operates the control jet as in the vertical case. Again, the control jet is at right angles to the telescopes, and the spinning vehicle is precessed to the horizontal position. Stability augmentation is provided by a rate gyro signal as before. However, since the system described here is an on-off signal system, the rate is limited only to some preset threshold limit, determined by the amount of residual coning which may be tolerated. A lower rate threshold will cut down the coning amplitude and make the system more stable, but will require a longer time and use more reaction jet fuel to make a given correction.

Computer studies of the two types of Miniguide controls have shown both to be feasible. Analog programs have been run to determine the effects of such parameter variations as gain changes in the attitude and rate channels, time lags in the solenoid valve and reaction jet, rate gyro dynamic characteristics, changes in moments of inertia and spin rate, external torques such as aerodynamic and thrust misalinement, etc. These studies have demonstrated that the Miniguide systems are relatively insensitive to such parameter variations. Signal gain can be varied over a 4:1 range without serious deterioration of system

performance, time lags in the reaction jet and solenoid valve may be compensated to a large degree by shifting their position on the vehicle with respect to the telescopes and rate gyro, and external torques merely add a nutation mode to the nominal coning, but do not affect the pointing accuracy of the system. The biggest effect on system performance is caused by a spin rate other than that which was designed for. Too high a spin rate increases the gyroscopic stability of the body, and the system will exhaust the reaction jet supply before reaching the control attitude - too low a spin rate reduces the gyroscopic stability, and the system is left with an excessively large coning amplitude.

A Miniguide control system has been flight tested, which was a version of the pitch control described earlier in this paper. This system, shown in figure 5 was designed to correct the pitch axis tip-off errors expected in a reentry research vehicle flight, and as such had rather peculiar control requirements. The control was to begin at final-stage ignition, and was required to damp the vehicle coning rate to below 6° per second and return the vehicle to the correct pitch attitude within the initial 10 seconds of motor burning.

The vehicle picked for the flight test of the system was a four-stage configuration commonly called a Javelin. Six-degree-of-freedom trajectory studies were made of this vehicle to establish the initial conditions under which the control operation would begin. These studies predicted the initial conditions to be 235,000-foot altitude, with a pitch attitude of 55° for the launch conditions selected. In addition, studies of the flight records of other vehicles using the same type of blast separation system planned for the reentry vehicle under consideration indicated that vehicle coning rates as high as 36° per second might be imparted by the tip-off disturbance. Since the possibility also existed that no tip-off disturbance would be present, the command pitch attitude was

chosen as 65°, so that system operation could be checked in the absence of the tip-off disturbance.

The condition of system operation with the rocket motor burning imposed a limitation on the mounting angle of the rearward-looking telescope. If the telescope were directed to far aft, the line of sight would pass through the free jet, or plume of the rocket, and mask the earth horizon. At these altitudes - 200,000 to 300,000 feet - the free jet is rather extensive. However, previous flights had shown that the telescope could be aimed back as far as 20° aft of a normal to the body without passing through the free jet. This angle was not large enough to permit the telescope to see the rearward horizon, so an alternate control scheme was developed to compensate for this situation. It is a variation of the horizontal control case, in which the front telescope only is used as the attitude control reference.

Figure 5 shows the schematic of this control case. As can be seen, the rear telescope now looks essentially straight out the side of the vehicle, while the forward telescope is set to see the forward horizon when the vehicle is at the correct attitude and altitude. In operation, the rear, or side, telescope signal is inhibited for a period of time equal to one body spin revolution each time the forward telescope receives an earth signal. The action of this inhibit circuitry allows the system to operate in a manner identical to the horizontal system in this phase. If the forward telescope fails to receive an earth signal during one spin revolution, this inhibit signal is not applied, and the side telescope causes the vehicle to be torqued in a nose-down direction until the forward telescope again sees the earth.

After programing this system on an analog computer to demonstrate the feasibility of the scheme, the system hardware was developed and the breadboard

version included in the computer program to provide realistic effects of hysteresis, time lags, etc. The appropriate attitude signals from the computer were used to turn on and off small fast-response incandescent lamps mounted in the back of the telescopes, and the actual telescope signals were then processed by the electronics breadboard. The computer signals representing body rate were modified by the rate gyro dynamics, and this signal fed to the breadboard electronics. The output of the electronics operated a solenoid valve which had been modified to actuate a switch to provide the electrical analog of the control jet thrust, which was fed back as a control signal into the vehicle dynamics portion of the computer setup.

Computer runs were made to determine reaction jet thrust levels, gas requirements, steady-state conditions, effect of spin rate variation, and system response times for ranges of initial conditions of 15° attitude error and 40°-per-second coning rate to zero attitude error and coning rate. It was found that a gas volume of 150 cubic inches at 3,000 psi was sufficient to reduce the initial attitude and rate errors to steady-state conditions within the required 10 seconds. However, it was found necessary to provide a 1-second damping mode at control initiation, as some combinations of attitude and rate initial conditions caused the control jet to blow for almost a complete revolution, which added little to the attitude control or damping, but exhausted the gas supply quickly. If the vehicle motion was damped out before attitude control was initiated, this condition was alleviated.

Figure 6 is a reproduction of a sample computer run. The gas supply used was unregulated, so that the jet thrust varied with the amount of gas used. The thrust level of the control jet was 40 pounds at 3,000-psi supply, and the jet was located 34 inches from the vehicle center of gravity. The rate gyro signal

threshold was set at 4° per second, which corresponds to a vehicle coning amplitude of $\pm 2^{\circ}$ for the vehicle parameters used. These parameters were moment of inertia in roll, 6.5 slug-ft^2; in pitch and yaw, 60 slug-ft^2. The nominal spin rate was 180 rpm, with a possible variation from 150 rpm to 210 rpm.

Since the breadboard system proved adequate to perform the required control task, it was made up into flight hardware. Figure 7 shows the complete control system as it was used in the flight test. The weight of the entire system was under 18 pounds, and with minor mechanical redesign could be brought down to 12 pounds. The entire control electronics, shown in figure 8, used only 13 transistors. Since it contains some rather novel circuitry, a brief description will be given here.

The diode logic network allows signals from either of the two telescopes, or negative signals from the rate gyro to enter the control amplifier, which employs a field effect transistor for high input impedance. The amplified signals operate an electronic switch consisting of a Schmitt trigger and power amplifier, which operates the control jet valve directly.

This inhibitor consists of an amplifier similar to the control amplifier, and a special monostable multivibrator which may be retriggered at any time. The multivibrator is triggered by amplified signals from the forward telescope each time it sees the horizon. When triggered, it closes a diode gate in the input circuit of the side telescope for a period of about 0.3 second, or one spin revolution.

Power is supplied to the entire system except the solenoid valve before launching. Power is supplied to the solenoid valve by a pressure switch on the fourth-stage rocket motor at ignition to initiate control operation. A capacitive network energized by this switch closes diode gates in both telescope circuits

for about 1 second to allow rate signals from the gyro to subdue any large initial tip-off rates before attitude correction is started.

The emitter followers shown on each telescope input were actually mounted in the telescopes themselves to reduce noise pickup in the connecting cables. The telescopes had a 3/8-inch aperture and a $2\frac{1}{2}$-inch focal length, and used an unfiltered lead sulphide photodetector. The total field of view was approximately 1°, with a sunshade which was effective for sun angles of greater than 10° off the optical axis. The small incandescent lamps in the rear of the telescopes were provided to be used for system prelaunch checks.

The prelaunch test equipment consisted of motor-driven switches which actuated the incandescent lamps in the rear of the telescopes in the same time sequence which would occur during flight. It was possible to actuate both telescope lamps, or just that in the side telescope to simulate both modes of operation, and check the operation of the inhibitor circuit. System operation was monitored through the telemeter, which showed each telescope signal and the output of the electronic switch. Checks of the rate gyro were limited to determining that the gyro motor was running. This was checked through a small microphone which was strapped to the gyro case, and connected through the umbilical cable to a small amplifier and speaker in the blockhouse.

Figure 9 shows the results of the flight test of this system. The attitude measurements were obtained from a third telescope, set 8° to the rear of the forward telescope, which saw the earth once each revolution during all the coning excursions. The initial conditions at the time of control initiation were somewhat different than those which had been predicted by the computer studies - the attitude high at 61°, and the coning rate low at 4° per second, which resulted

in a lower usage rate of control gas than anticipated. However, the system controlled the pitch attitude to this set point in less than 5 seconds, and held the coning rate at 4° per second, as it should. During the initial 12 seconds, after which the vehicle was scheduled to be destructed for range safety reasons, the flight records agree very well with the preflight computer studies. However, the vehicle destruct failed to terminate the rocket-motor thrust, and the system continued to control the vehicle to the set point attitude. Since the vehicle had a large component of vertical velocity, the set point attitude decreased, due to the continual depression of the angle to the horizon. As can be seen in figure 9, at about 35 seconds the vehicle attitude begins to fall below the set point attitude. Since the vehicle attitude would, under these conditions, rise above the set point attitude if control were lost due to exhausting the control jet gas, the telemeter records were examined closely to determine the reason for the disparity.

It was found that the cause was an effect of earth limb darkening. Due to the angle to the horizon, and the relative position of the sun, less scattered sunlight was being returned to the telescopes at the forward horizon. When the amount of energy at the horizon fell below the threshold of the telescope channel, the vehicle was torqued in a nose-down direction until the threshold signal value was reached, creating an apparent horizon lower than the actual horizon. This effect is shown in figure 10, which is a reproduction of the telemeter records from 50 to 54 seconds after ignition. The signals from the forward telescope, indicated by the arrows, are from the portion of the horizon where the signal is below the threshold level of the telescope channel. The remaining signals are above the threshold level, and the apparent horizon lies between the two regions.

In the flight-tested version of the Miniguide system, there was a total yaw change in the vehicle flight path equivalent to about 15° at ignition of the final stage. This yaw deviation was introduced by the combination of aerodynamic torques produced by the varying angle of attack during the extended coast phase, tip-off disturbance, and the action of the control system. No attempt was made in this flight test to control this yaw dispersion. However, the results of the flight test in the controlled plane have stimulated interest in a two-axis control for the Miniguide. Some preliminary computer studies have been made which show that the output of a body-fixed azimuth sensor, operating on a suitable reference such as the sun, can be fed into the existing electronics and jet assembly to provide a two-axis control system. In this case, the only additional complexity would be the sensor, with no other additions to the electronic or mechanical complexity of the system.

A simple form of such an azimuth sensor operating from the sun is shown in figure 11. The detectors consist of shaded solar cells, mounted along the same body axis as the horizon telescopes. The cells and shades are set in the vehicle so that at the correct azimuth angle both cells are completely shadowed by their respective shades. If the vehicle is in a nose-left attitude, cell A will be exposed to the sun for a portion of the vehicle revolution, while cell B will be shaded throughout the vehicle revolution. For a nose-right attitude the reverse will be true, cell B exposed to the sun during a portion of the vehicle revolution, and cell A shaded. The output of these cells then actuates the control jet in the same manner as the horizon telescopes do in the pitch plane control, and torque the vehicle to a zero azimuth error attitude.

The preliminary computer studies indicate that the two-axis control will perform satisfactorily for sun angles as much as 45° off the optimum angle, where

the sun lies in the azimuth plane. An increase in control gas usage is incurred by the addition of the second controlled axis, though this increase has not yet been evaluated.

A study is now under way concerning the feasibility of using this two-axis control scheme for the injection of a payload into orbit, using a multistage vehicle with attitude control of the last stage only. Trajectory studies are being made to determine the accuracy with which the velocity vector prior to fourth-stage ignition can be controlled by appropriate selection of launcher azimuth and elevation and vehicle aerodynamic stability. This, in conjunction with the predicted altitude variation, control accuracy, and fourth-stage velocity increment determines the orbital parameters of the payload package. Preliminary results of this study indicate that it is feasible to inject a payload into a crude orbit using a relatively inexpensive four-stage vehicle and the Miniguide two-axis final-stage control. Such a vehicle can be launched from a simple rail-type launcher, and does not require the complex ground support equipment associated with inertial-type attitude control systems.

An application of the vertical Miniguide system is also currently under study at Langley. In this case the Miniguide system is used to aline the payload to a vertical attitude after burnout of the final stage. The payload is designed to measure horizon position and definition in various portions of the electromagnetic spectrum to a high degree of accuracy, and has a limited angle of scan over which it can acquire the horizon. A star scanner is included in the payload to determine the vehicle attitude to the extreme accuracy required - approximately 0.02°. The control system must erect the payload to a vertical attitude to within less than $1/2^{\circ}$, and reduce the coning rate to as low a value as is practicable. The control must then be shut off so that no external torques are applied to the

vehicle, and left off for a period of several minutes. During this time the vehicle travels down range, and due to the spherical earth drifts away from the local vertical. When the angle off vertical has reached approximately 3.5°, the control is operated again, and the vehicle erected to the new local vertical. This procedure is repeated as many times as is required to maintain the vehicle to within 3.5° of the local vertical.

Computer studies have been made of the above system to determine the design parameters such as jet thrust, gas bottle size, and rate and attitude signal gains. The vehicle parameters used correspond to the payload and empty rocket-motor case - I_{XX} between 2.5 and 3.5 slug-ft^2, $I_{YY} = I_{ZZ}$ between 20 and 40 slug-ft^2, spin rate 90 rpm nominal, possible variation between 60 and 120 rpm, and a 3-foot moment arm for the reaction jet. Initial conditions were 25° attitude error, and up to 40° per second coning rate.

Figure 12 shows the results of one of the computer runs, in which the initial condition in attitude was 25°, and the initial coning rate 40° per second in a direction to provide the most additional attitude error. As may be seen in figure 12, the system damps out vehicle rate, and corrects the attitude to vertical in just under 20 seconds, with enough gas remaining for at least two more attitude corrections. It will be noted on the body rate channel that during the major portion of the attitude correction phase of the system, the rate maintains a constant value. This is a result of limiting the amplitude of the attitude error signal which was found to add to the system performance and reduce the amount of control gas used.

From these records it was determined that the attitude was maintained to within $1/2^{\circ}$ with a coning rate of under $1/2^{\circ}$ per second. The system parameters were determined to be 0.2° switching threshold for the attitude signal, limited

at a signal equivalent to $2\frac{1}{2}^{\circ}$; 0.24°-per-second switching threshold for the rate signal, and an unregulated gas supply of 140 cubic inches at 3,000 psi, with a control jet which produces a thrust of 20 pounds at 3,000 psi. It was also determined that the rate and position gains could be varied from 1/2 the nominal value to four times the nominal value without seriously affecting the system dynamics, and still remaining within the design goals on steady-state position and rate, throughout the specified range of moments of inertia and spin rate.

On the basis of these studies, Langley plans to use this system on three vehicles to be fired within the next year and a half. It is felt that, due to the excellent correlation of the computer studies and flight results of the Miniguide which has been flight tested, these computer values are fairly realistic and can be duplicated in flight. One unknown area remains, however, which was pointed out in the previous flight test. The horizon will not emit equally around the entire earth's disk, and since the system described will try to erect to a vertical determined by the center of illumination of the horizon rim, it may not be at a true vertical. It is felt that it will be well within the field of view of the experiment radiometer, however, so the experiments would still be successful. The results of the horizon definition experiments, and the highly accurate attitude measurements provided by the star scanner should provide some idea as to the effect and magnitude of this horizon irregularity which can be included in the design of the future flights.

Due to the low cost, simplicity, reliability, and low weight of the Miniguide control system, it is felt that it is worthy of further study with respect to its future uses. Two of the planned uses have been discussed here, and we hope that more will be forthcoming in the future.

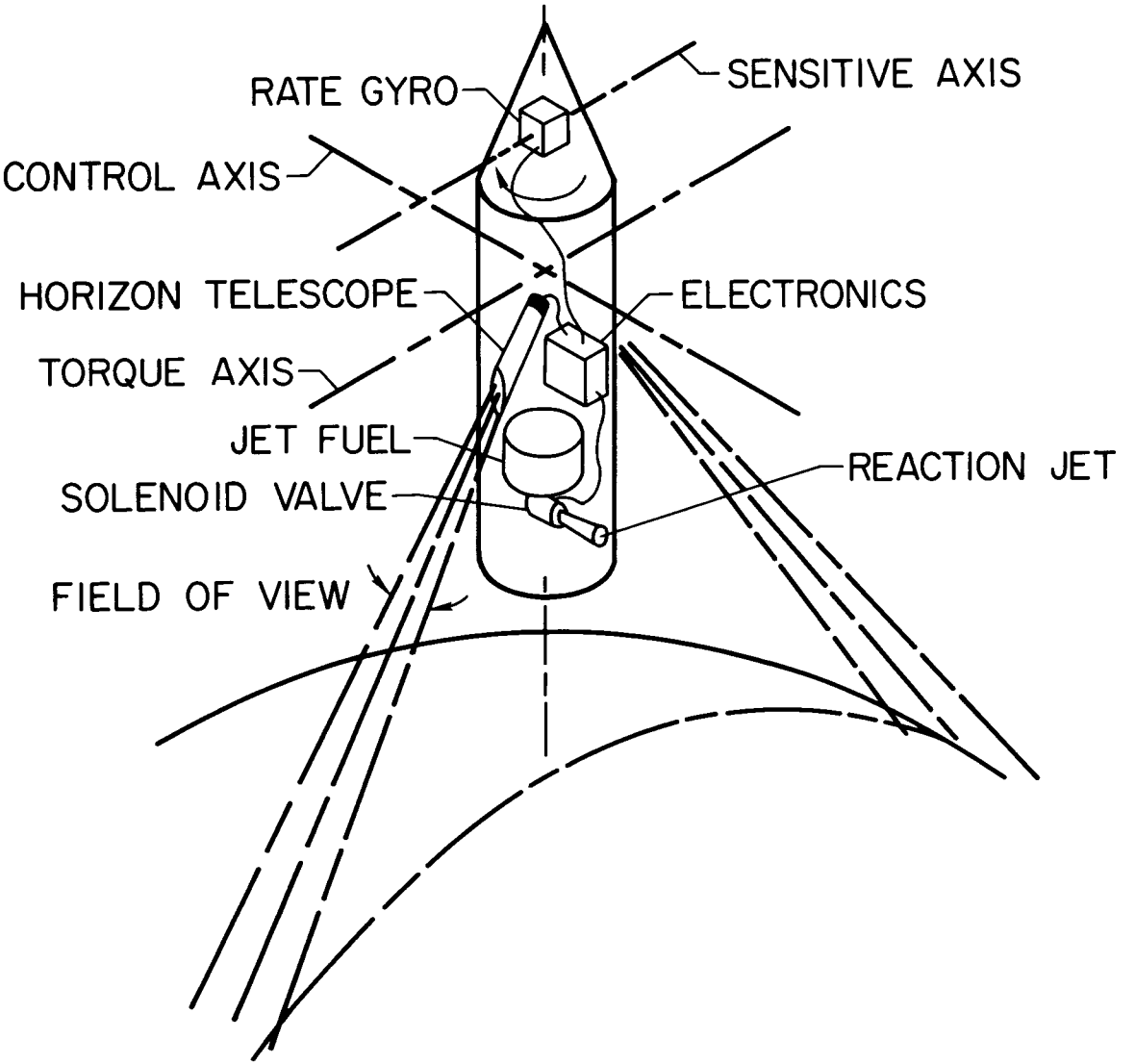

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Figure 1.- Vertical control.

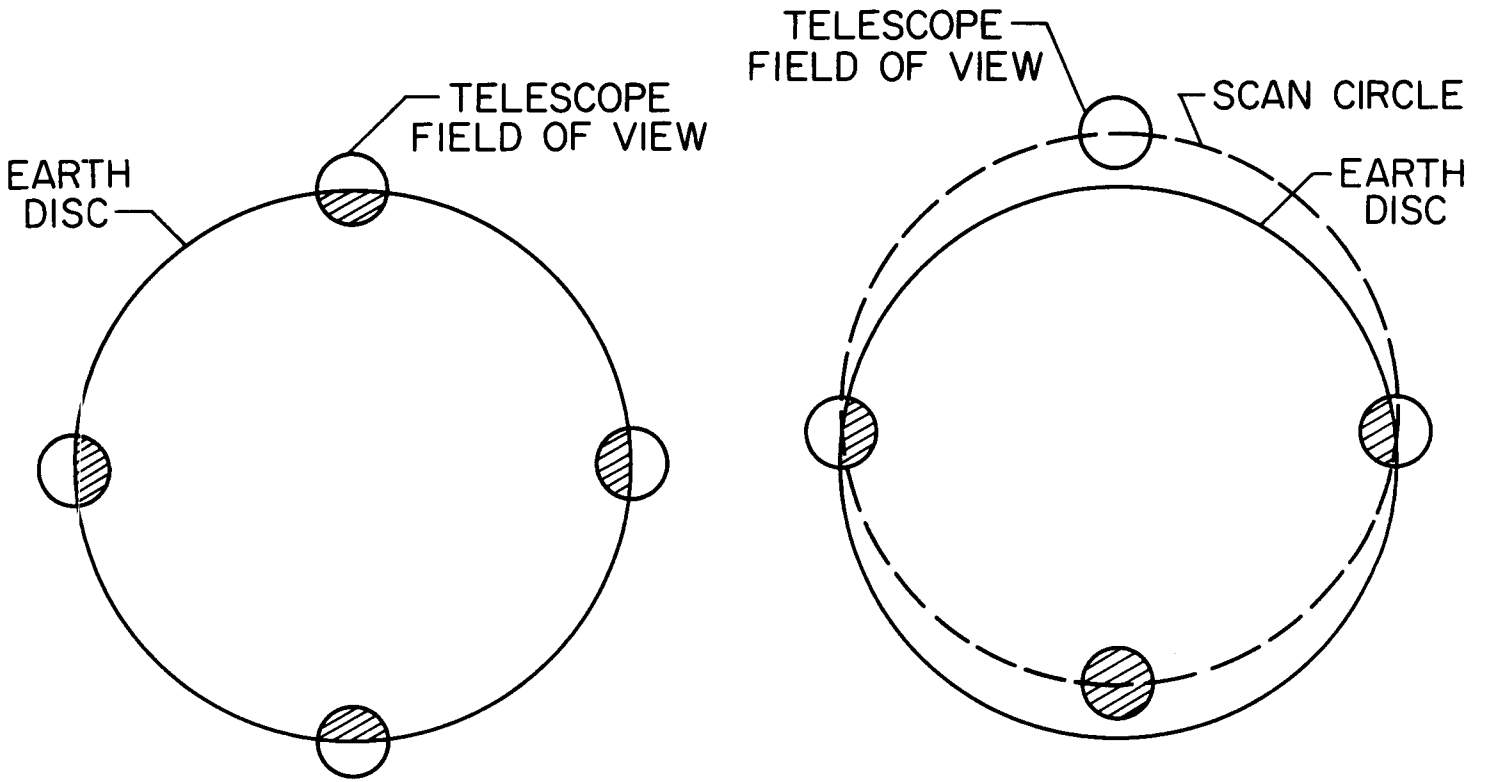

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Figure 2.- Vertical scan technique.

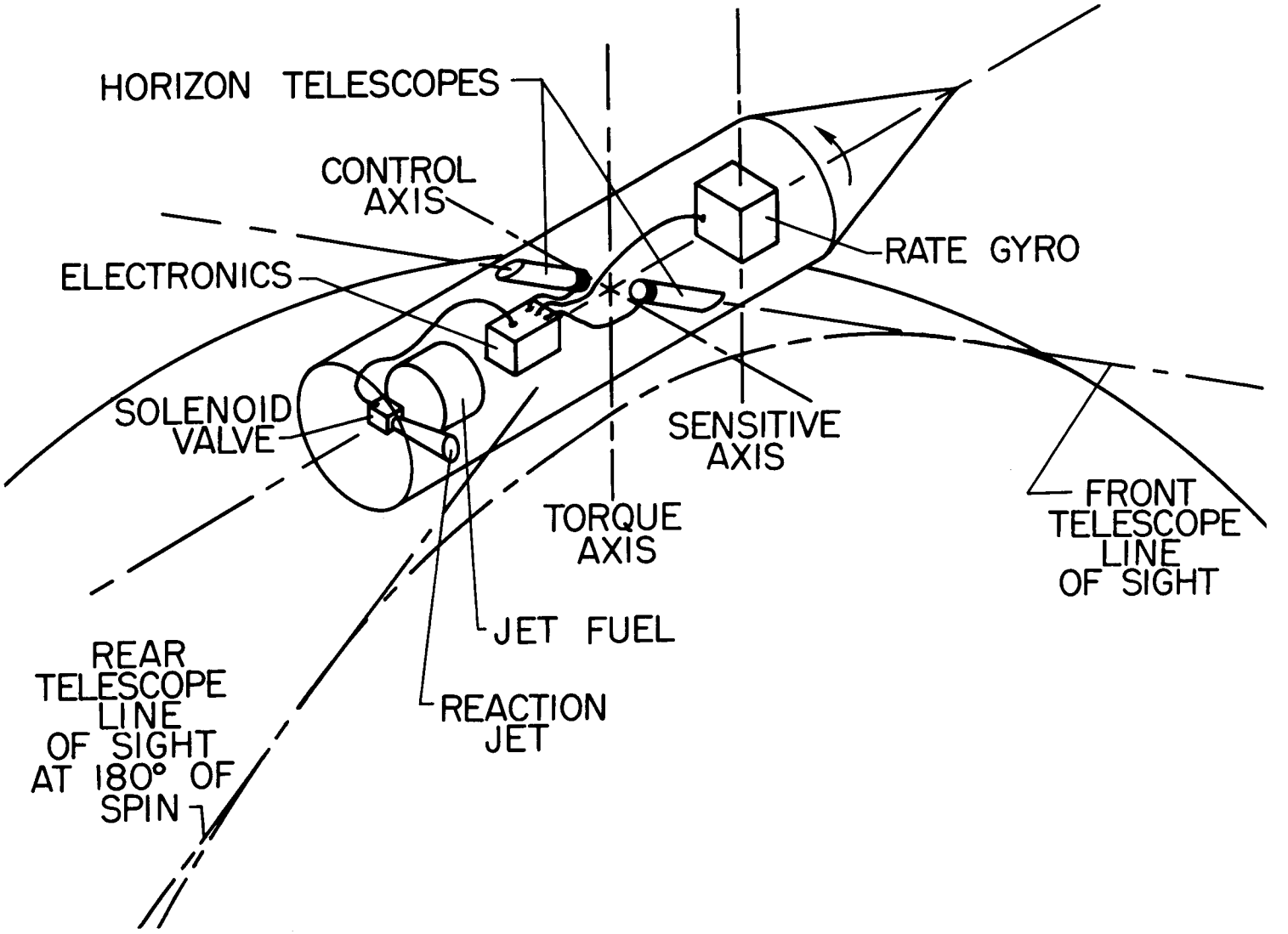

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Figure 3.- Horizontal control.

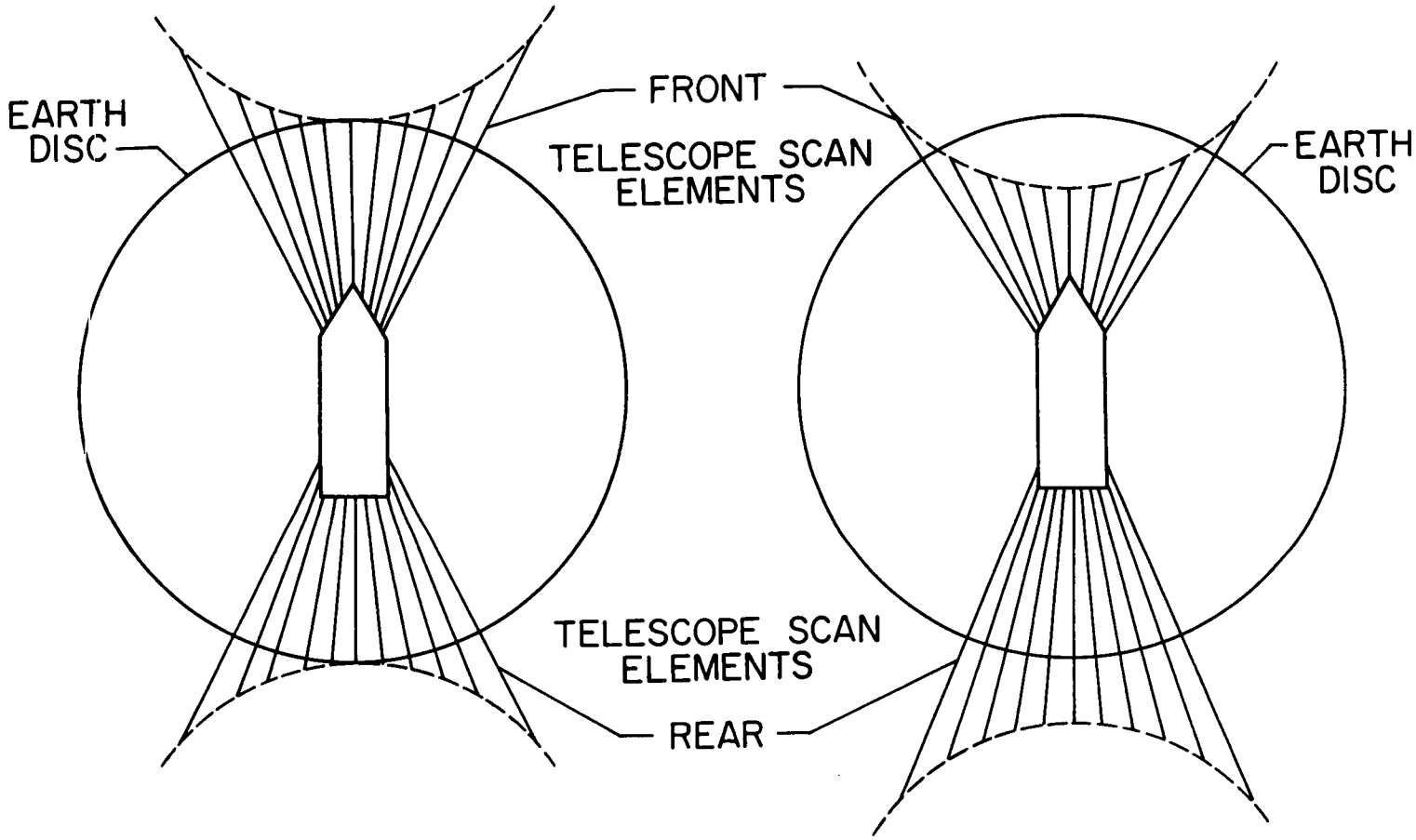

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Figure 4.- Horizontal scan technique.

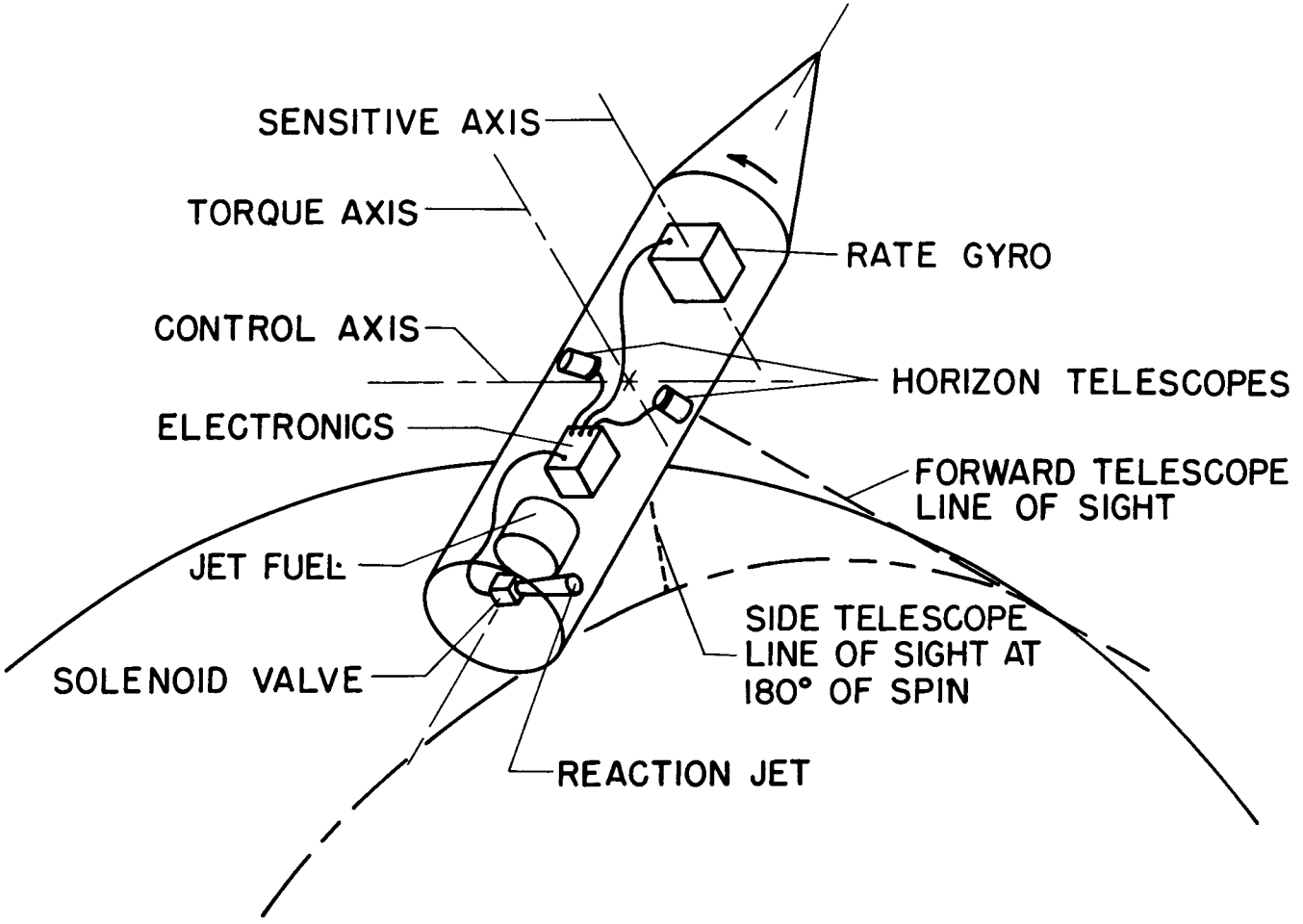

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Figure 5.- Flight-tested pitch control.

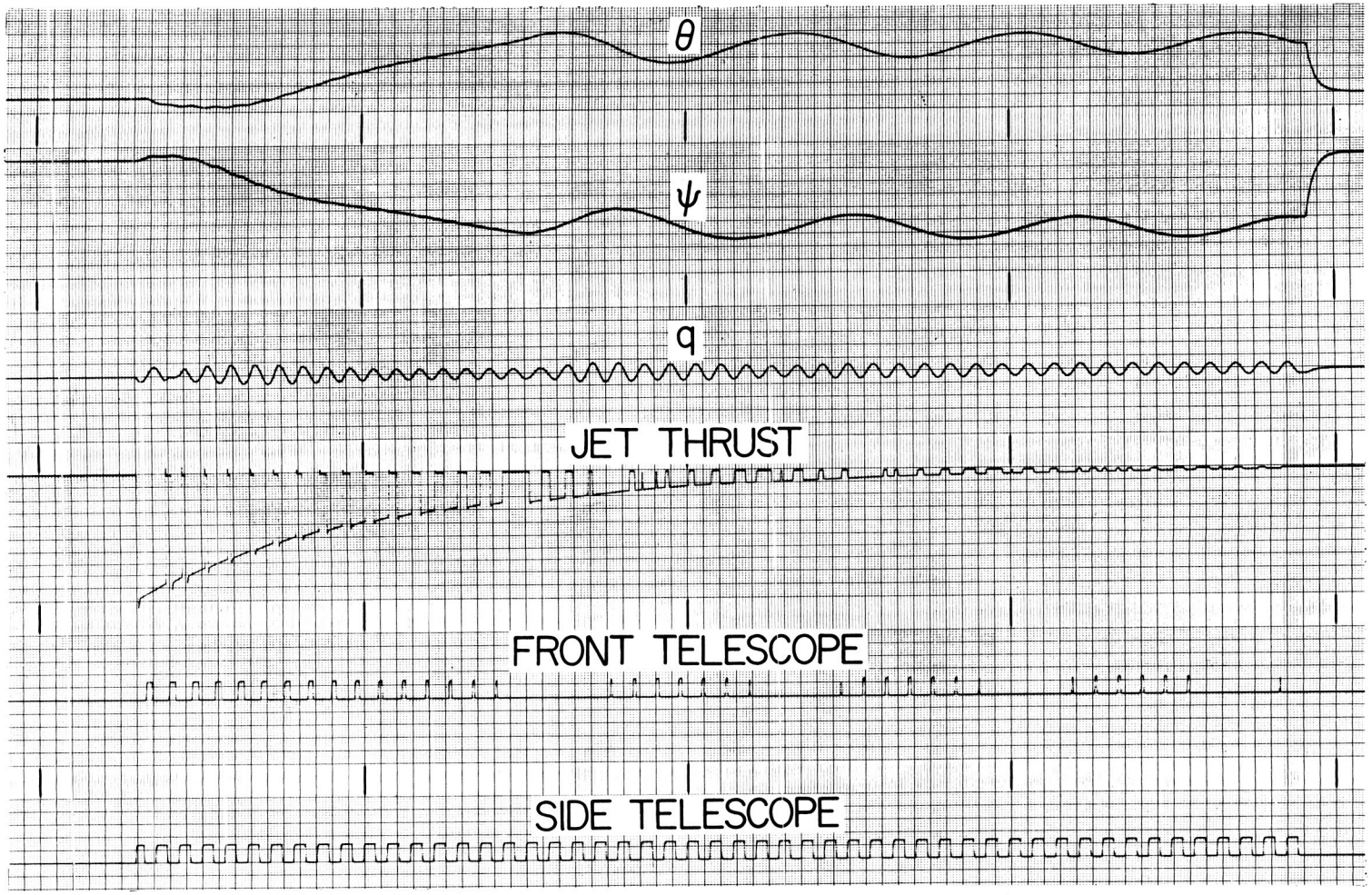

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Figure 6.- Flight system computer run.

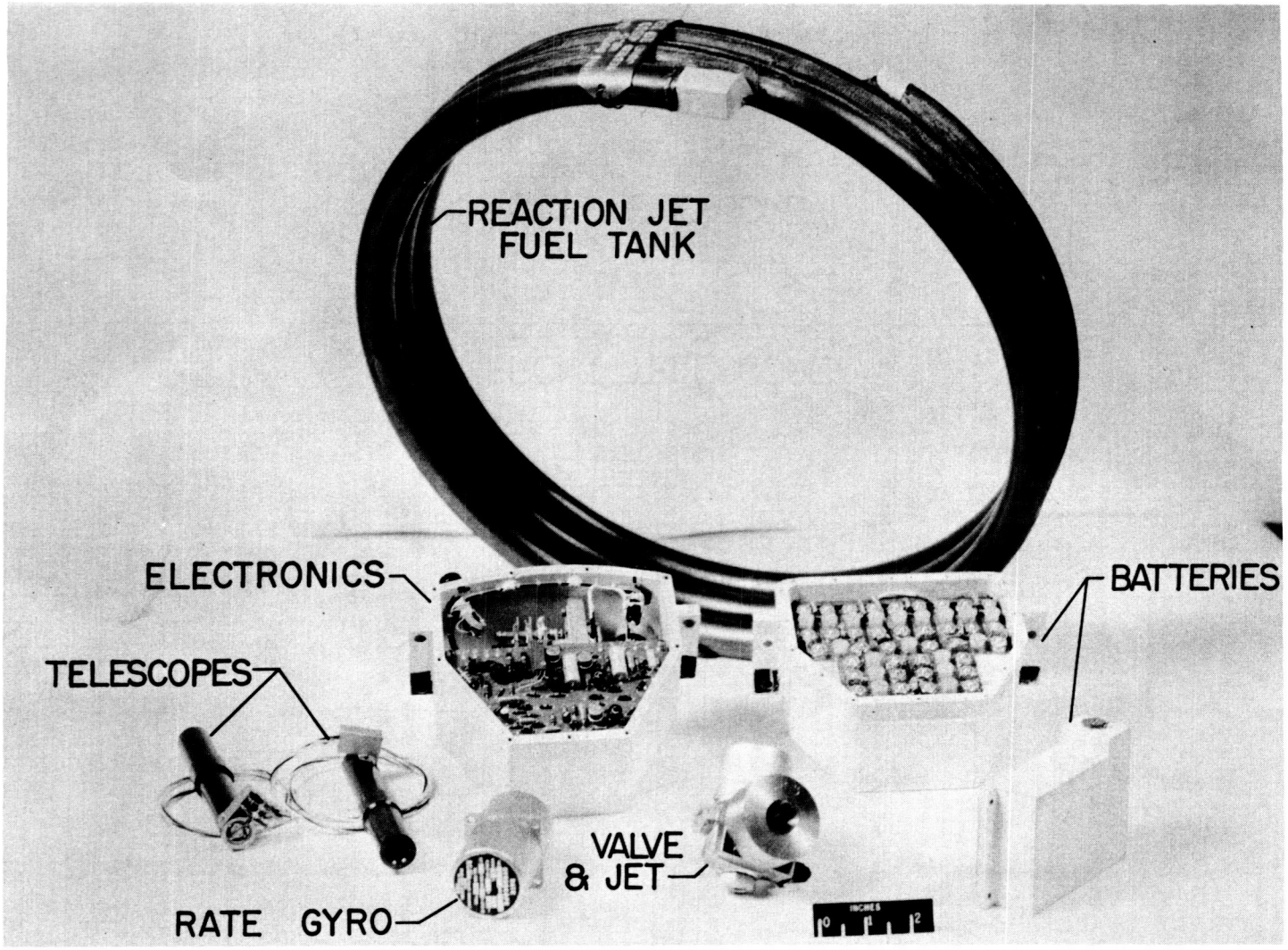

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Figure 7.- Complete control system.

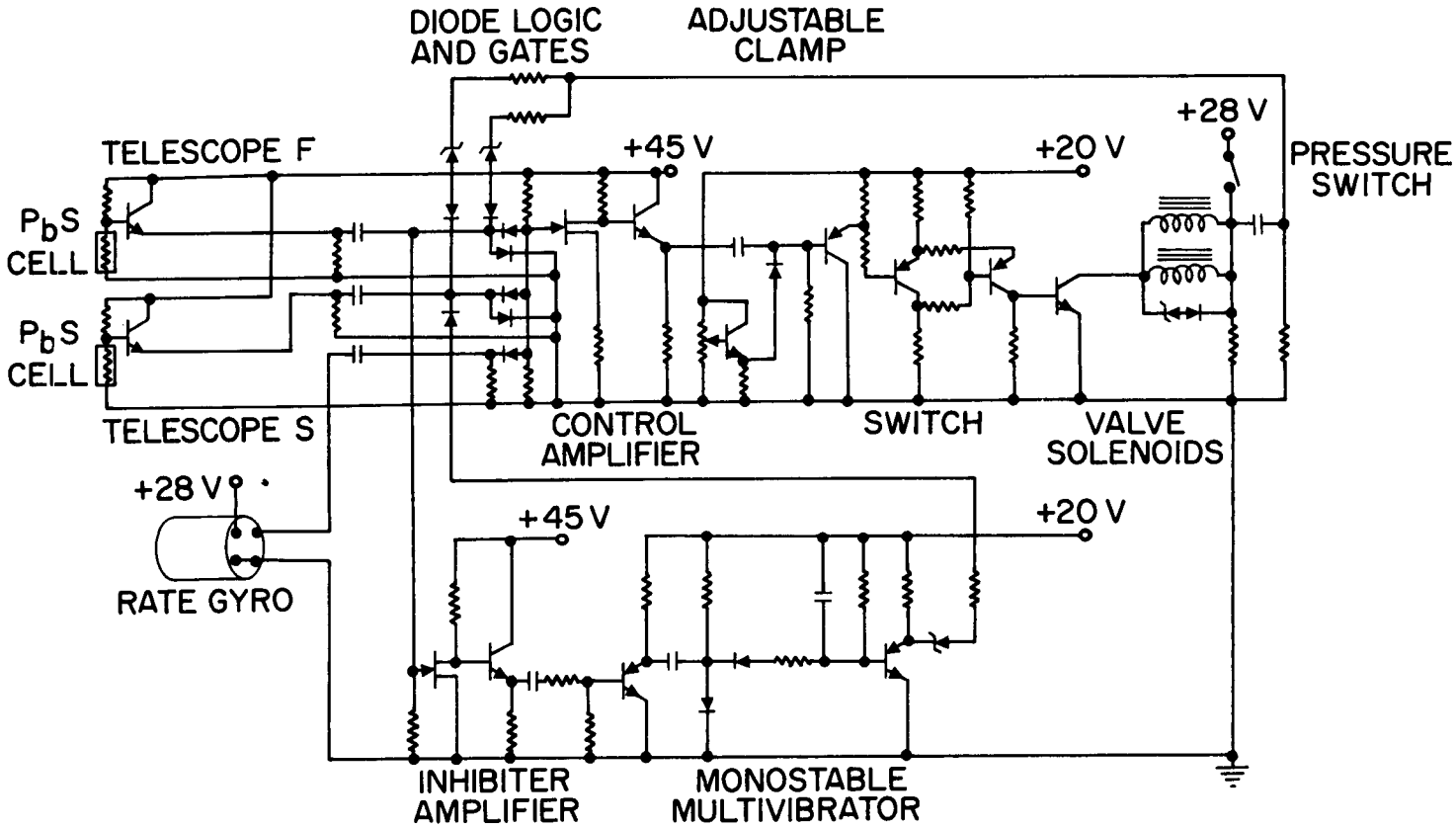

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Figure 8.- Control system electronics and wiring diagram.

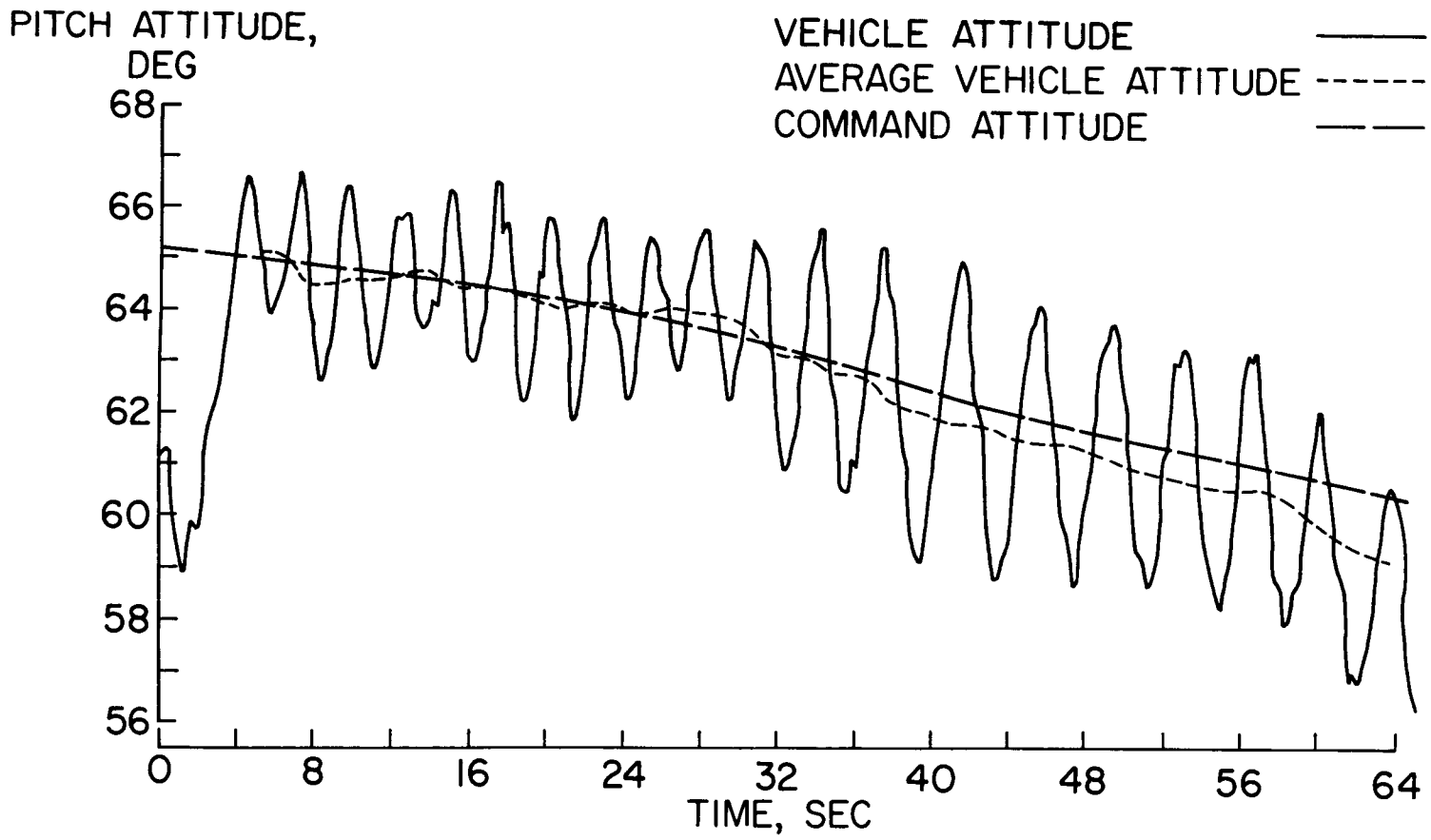

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Figure 9.- Flight-test results.

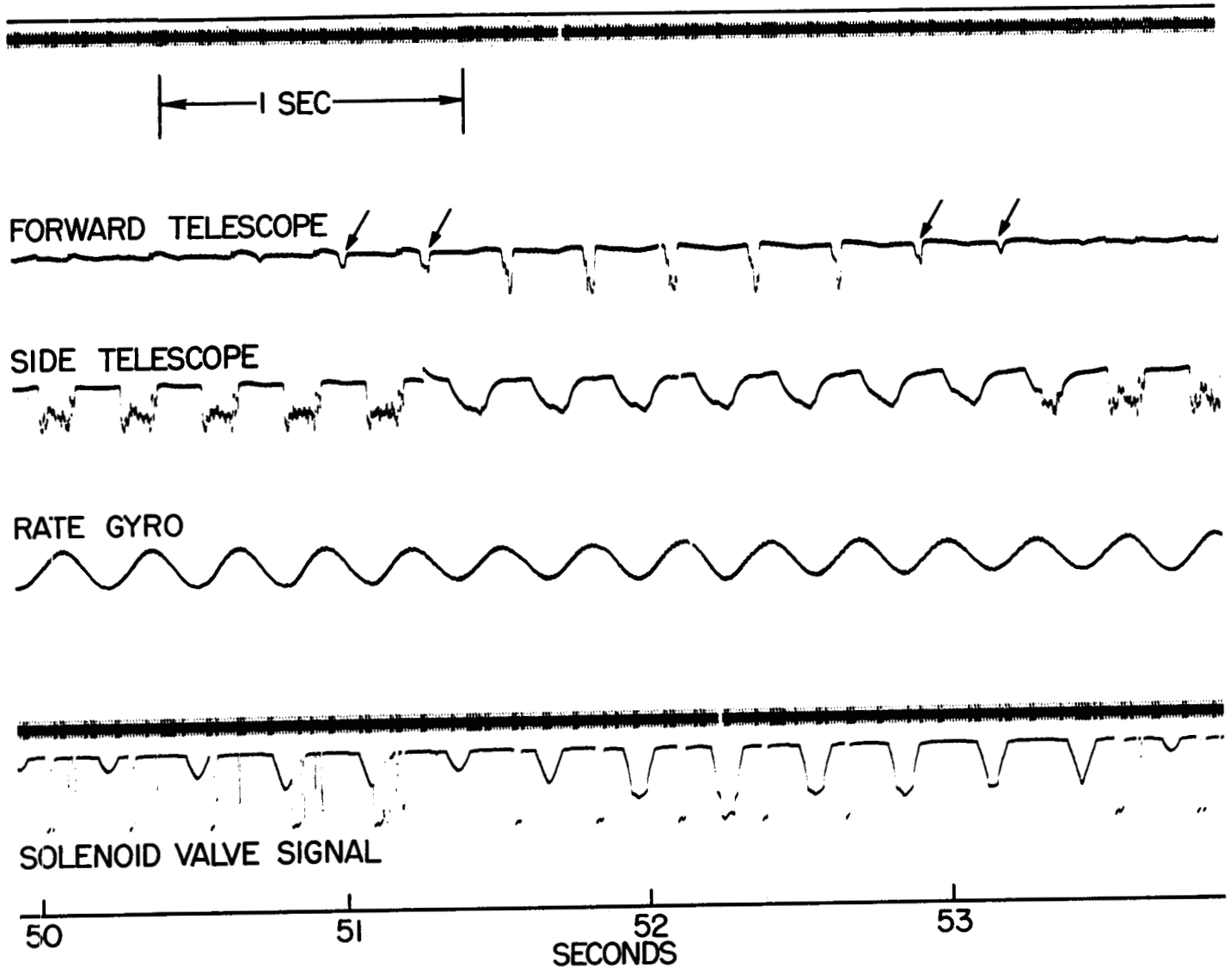

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Figure 10.- Flight telemeter records at 50 to 54 seconds.

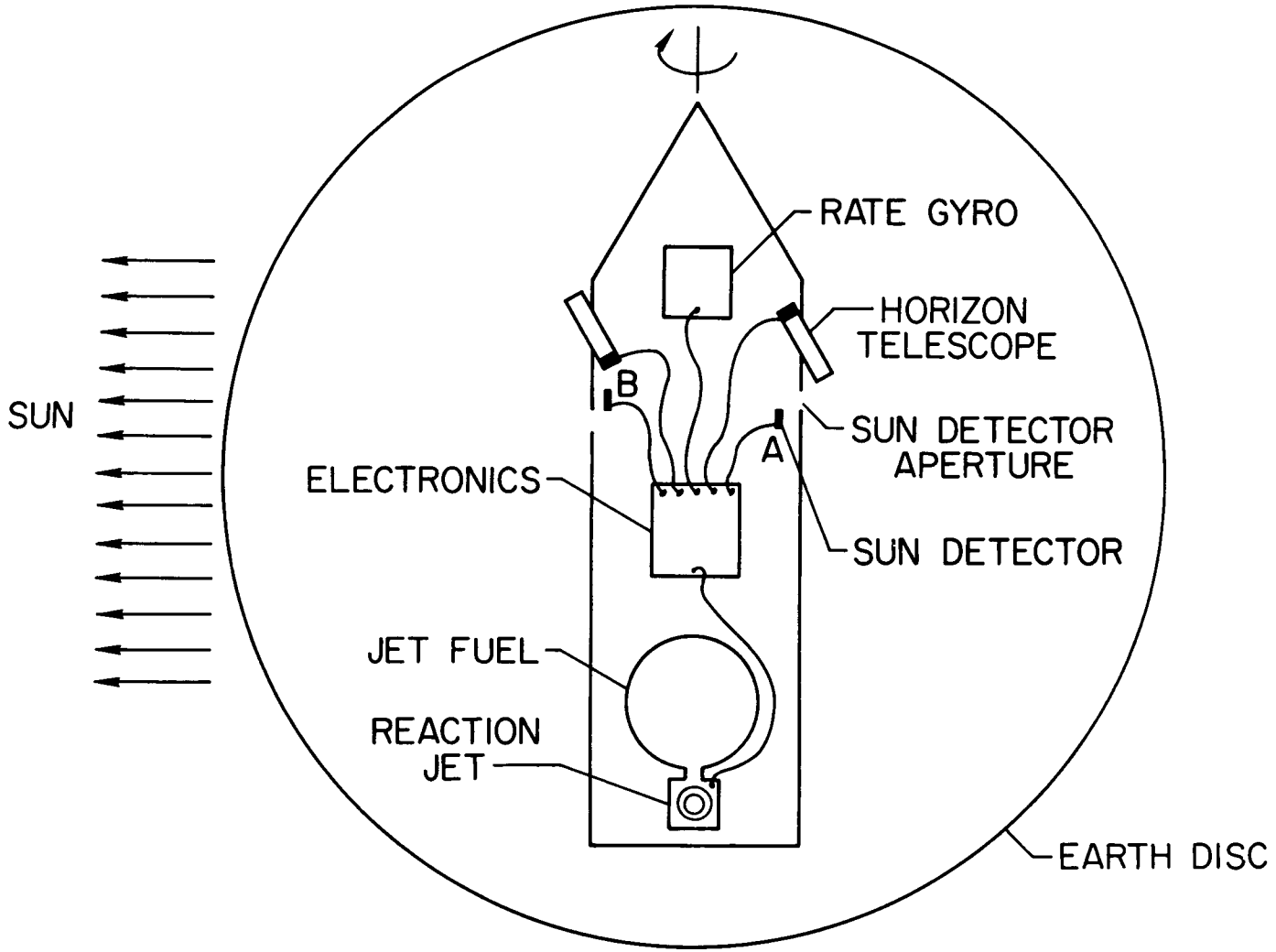

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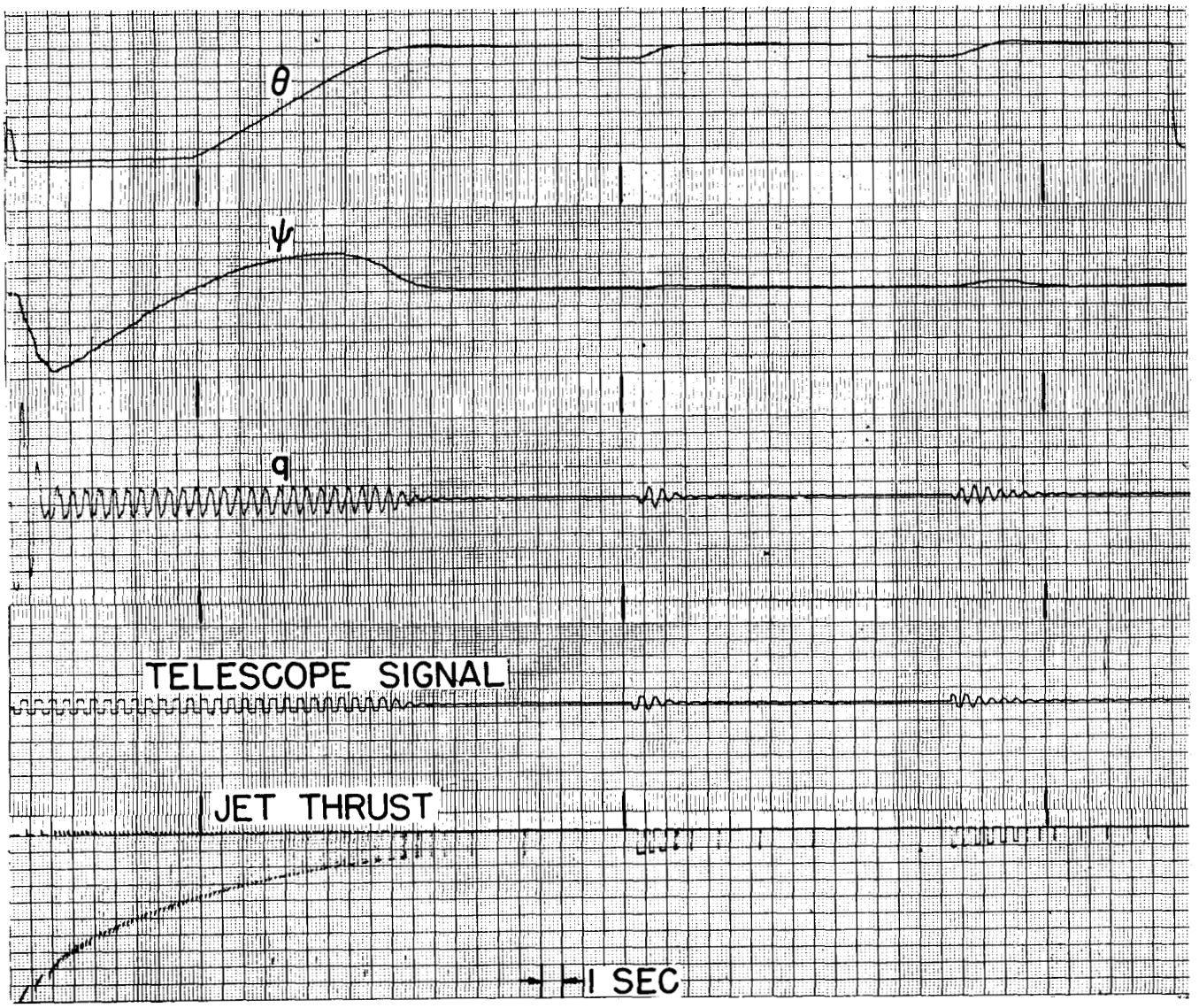
Figure 11.- Two-axis control.

NASA-Langley, 1963



NASA

Figure 12.- Vertical control computer run.