CONTRACT NO. NAS5-3173

FINAL ENGINEERING REPORT

TIROS X

METEOROLOGICAL SATELLITE SYSTEM

VOLUME I

Prepared For The

GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS
AND SPACE ADMINISTRATION

By The

ASTRO-ELECTRONICS DIVISION
DEFENSE ELECTRONIC PRODUCTS
RADIO CORPORATION OF AMERICA
PRINCETON, NEW JERSEY

AED R-2801

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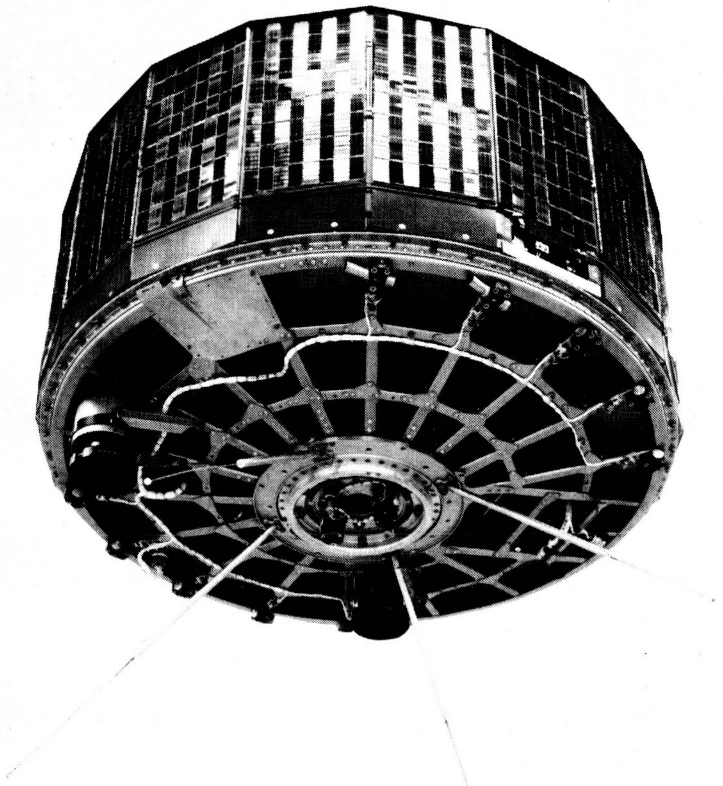

The TIROS X Satellite

PREFACE

TIROS X was developed and built by the Astro-Electronics Division (AED) of the Radio Corporation of America for the Goddard Space Flight Center of the National Aeronautics and Space Administration, under NASA Contract NAS5-3173.

TIROS X was the second of the Series III-type TIROS satellites to be launched, and while under development was designated as spacecraft "OT-1". As such, it was first assembled in March 1964, in a configuration similar to TIROS VII. However, based on the results of a study directed by NASA and performed by AED, spacecraft OT-1 was reconfigured in February 1965 to suit the requirements of a near-polar, sun-synchronous orbit. This reconfiguration was based on the utilization of two axial-mounted standard TIROS cameras, the addition of a Quarter Orbit Magnetic Attitude Control (QOMAC) system, and the minimization of any additional modifications. The reconfigured spacecraft was successfully launched on July 1, 1965, as TIROS X, the first standard TIROS satellite to utilize a near-polar, sun-synchronous orbit.

This report, which comprises two volumes, is the Final Engineering Report for the TIROS X Meteorological Satellite System. The functions of this report are as follows:

(1) To describe the efforts involved in reconfiguring spacecraft OT-1 to suit the requirements of a near-polar, sun-synchronous orbit.

(2) To provide technical descriptions of the design improvements and the equipment added to the standard TIROS spacecraft configuration and to the basic TIROS Command and Data Acquisition (CDA) ground stations for the TIROS X system.

(3) To describe the effects of choice of launch date and launch time on the operation of the TIROS X system.

(4) To provide technical descriptions of the basic design improvements and additions made to the standard TIROS system for TIROS X.

(5) To describe the various system and subsystem tests performed during the TIROS X program.

(6) To describe the various environmental tests and calibration tests performed on the integrated spacecraft during the TIROS X program.

(7) To describe prelaunch and launch activities at the launch site, at the TIROS Technical Control Center, and at the CDA ground stations.

As noted previously, the TIROS X final report is divided into two volumes. Volume I comprises Parts I through IV. PART I serves as an introduction to the TIROS X program, and describes the systems concept upon which the program was based, the launch and orbit considerations, and the photocoverage available with TIROS X. PART II presents technical descriptions of the satellite subsystems and details the various phases of testing performed on individual units and systems and on the integrated spacecraft. PART III covers ground station equipment; while PART IV covers the preparation and launch-phase operations at the launch site, the TIROS Technical Control Center, and the TIROS CDA ground stations.

Volume II comprises PART V, which contains a description of both the satellite-borne and ground-station portions of the command and control subsystem and that material which is of a classified nature, e.g., data pertaining to command frequencies and programming sequences.

In general, the TIROS X final report presents only brief functional descriptions of those spacecraft subsystems and subsystem elements which are essentially the same as their counterparts on previous TIROS satellites. However, greater detail is given to the design modifications made specifically for TIROS X and to the reconfiguration of the spacecraft for the TIROS X mission. Complete and detailed discussions of those components which were changed in previous TIROS programs are described in the respective final reports for those programs.

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PART I. INTRODUCTION

SECTION I. PROGRAM SUMMARY

TIROS X was successfully launched from Cape Kennedy, Florida, by means of a Delta Rocket, on July 1, 1965. It was the first standard TIROS satellite to be planned for a near-polar, sun-synchronous orbit, and its launch date was scheduled to provide photo-coverage for the 1965 hurricane season.

The TIROS X program was performed by AED, under NASA contract NAS5-3173, utilizing spacecraft OT-1. The spacecraft was originally assembled in the standard TIROS configuration (almost identical to TIROS VII). At that point the spacecraft was scheduled to be launched in the second quarter of 1964, into a 400-nautical-mile circular orbit at a 58-degree inclination. Assembly and debugging were completed by December 1963; however, in March, 1964, spacecraft OT-1 was placed in a "Hold" status by NASA directive.

In January of 1965, under the same contract and in response to a NASA request, AED conducted a study of the feasibility of modifying spacecraft OT-1, with minimal re-configuration, to suit the requirements of a near-polar, sun-synchronous orbit, including an added attitude-control capability (QOMAC system) to increase daily photo-coverage two to three times over that available with standard TIROS satellites. The planned orbital altitude and inclination were specified at 400 nautical miles and 98.6 degrees, respectively, to establish the sun-synchronous nature of the orbit and extend the available photocoverage.

In February, 1965, AED issued a technical plan outlining the necessary changes and modifications and pointing out the advantages to be gained in extended coverage, constancy of illumination, and favorable sun angles with the proposed system. Based on the results of this study, NASA issued Modification 17 to contract NAS5-3173, rescinding the "Hold" on spacecraft OT-1 and directing the proposed reconfiguration.

The AED effort on the TIROS X program included the following activities:

(1) Implementation of the TIROS X spacecraft design proposed by AED. The re-configuration included the following:

(a) the addition of the dual-channel attitude horizon scanner, the solar-aspect indicator, and the QOMAC system,

(b) various modifications to existing equipment, including modification of the camera clocks to increase the remote-picture interval, and

(c) the disconnection of the north-indicator subsystem.

(2) Qualification of the new subsystem elements required for the reconfiguration, and provision of qualified flight spare units as back-up equipment. (The subsystems and spare units which were either unchanged or only slightly modified for the reconfigured spacecraft were requalified.)

(3) Modification of the spacecraft structure and harness, as required for the new configuration.

(4) Integration of the qualified subsystem elements on the modified spacecraft structure and debugging of the complete assembly, ensuring that the integrated spacecraft met the established specifications.

(5) Environmental testing of the integrated spacecraft to flight-level requirements.

(6) Calibration of the spacecraft's sensing elements to ensure the capability of accurate attitude-control and photocoverage predictions.

(7) Prelaunch analysis of the orbital elements.

(8) Preparation of instructions for performing the "turn-around" maneuver.

(9) Preparation of instructions for checkout of the spacecraft's functions and evaluation of the spacecraft's performance during the initial orbits.

(10) Delivery of the spacecraft, upon satisfactory completion of environmental testing and final calibration, to the ETR for launch.

(11) Provision of engineering support for the launch effort.

(12) Modification of the TIROS CDA stations and Go/No-Go equipment to ensure compatibility with previous TIROS satellites.

(13) Preparation of instructions for commanding the satellite to execute specific attitude-control and data-gathering functions.

(14) Preparation of instructions for receiving and processing data from the spacecraft, and maintaining the ground equipment in operating condition.

(15) Provision of direct engineering support to the TIROS ground stations and TTCC during the launch and immediate post-launch periods.

The AED prelaunch and launch phase responsibilities culminated with the successful launch and orbit of TIROS X at 2307 EST on July 1, 1965. This document, the final engineering report on the TIROS X Program, covers the AED effort up until the time of the launch. Further data on the launch-phase activities and subsequent operation of the satellite will be presented in the separate evaluation report to be issued upon completion of TIROS X operations.

SECTION II . SYSTEM OPERATION

A. SPACECRAFT DESCRIPTION

The primary operational instruments aboard the TIROS X satellite are two wide-angle television (TV) cameras, each of which utilizes a slow-scan, 0.5-inch vidicon, supported by associated control, communication, and recording equipment, and a 2-watt TV transmitter. The satellite also contains the attitude-control and attitude-measurement devices necessary to achieve the initial satellite orientation and maintain this attitude during the course of the mission. Telemetry circuits for measuring the satellite's operating parameters and transmitting these parameters to the TIROS CDA ground complex are also included. Two beacon transmitters are included in the satellite to facilitate tracking and to provide two subcarrier channels for the transmission of attitude data, "housekeeping" data, and other telemetry data.

The satellite's operation at remote locations is controlled by means of a command and control subsystem, which receives, decodes, and stores ground-initiated commands and activates the satellite components required to execute the commands. Power for the satellite's electrical components is supplied by means of a solar-energy converter, comprising 9120 P-on-N silicon solar cells, and 63 nickel-cadmium storage batteries. The spacecraft structure which is very similar to that of the standard TIROS spacecraft, is 42 inches in diameter and 22 inches in height.

B. DYNAMICS CONTROL AND SPIN-AXIS ORIENTATION

The dynamics-control subsystem provides for (1) controlling the spacecraft's nutation and spin rate when it is first injected into orbit, (2) establishing the initial orientation of the spacecraft as soon as possible after injection, and (3) controlling the satellite attitude and spin rate throughout the mission.

A spin rate of approximately 126 rpm is imparted to the satellite/third-stage rocket assembly upon separation from the second stage of the launch vehicle. Approximately 1.5 minutes after the combined assembly is injected into orbit and the third-stage rocket has been turned off, the satellite automatically separates from the third-stage rocket, maintaining the 126 rpm spin rate. When separation occurs, the lift-off and separation switches close, an indication of separation is telemetered to the ground, the precession dampers are automatically activated, and the timing for the automatic activation of the despin mechanism is initiated. The precession dampers are tuned-energy-absorption-masses which rapidly damp any components of force occurring at separation which might tend to nutate the satellite. Approximately 7 minutes after separation of the satellite from the third-stage rocket, the despin timer causes the firing of a pair of squibs that release the despin mechanisms. The despin cables, with the attached weights, are wrapped around the satellite, about the periphery of the baseplate. When released, the cables unwind, and, with the attached weights, are cast off from the satellite while absorbing sufficient energy to cause the spin rate to be decreased to the 8 to 12 rpm range (within approximately 0.5 second).

Though both precession dampers and the despin mechanism are designed for automatic operation, a back-up capability for ground-controlled activation is also available.*

To achieve the desired orbit, the launch sequence and injection conditions were planned to cause the satellite spin axis, at the point of injection into orbit, to be approximately in the orbital plane, perpendicular to the line of nodes, and pointing toward the southern hemisphere. Since the launch time was near the summer solstice, the initial sun angle with respect to the spacecraft would be unfavorable (i.e., $\gamma \approx 115°$). To achieve mission mode, there was a requirement for a change of spin-axis orientation after launch, before picture-taking operations could be initiated; to accomplished such a change, TIROS X was equipped with a QOMAC system.

The turn-around maneuver is effected by the establishment of known magnetic fields about the satellite which interact with the earth's magnetic field to produce a torque on the satellite. The satellite's magnetic field is generated when current is caused to flow in one half of the center-tapped QOMAC coil. Quarter-orbit reversals in the selected half of the QOMAC coil are programmed to reverse the direction of the satellite's magnetic field according to the satellite's location in the magnetic field of the earth. The torque produced by the current flow in the QOMAC coil causes the spin axis to precess toward the desired attitude in a known direction and at a rate proportional to the dipole field strength. The TIROS X turn-around maneuver is shown in Figure I-1.

On orbit 0001, after nutation damping and despin have occured and the orbital elements have been determined, the turn-around maneuver is initiated. It is planned to be accomplished within the first 20 orbits, so that the desired gamma angle will be achieved and suitable power and thermal profiles for satellite components will be established as soon as possible after launch.

The torquing program is initiated at a prescribed time after the ascending node crossing. With the design of the QOMAC system, the selection of the appropriate starting time permits positioning the precession vector anywhere in the orbital plane. The precession axis is, in fact, located essentially midway between the injection position of the spin axis and the desired position of the spin axis.

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*As described in Volume II, the classified supplement to this report.

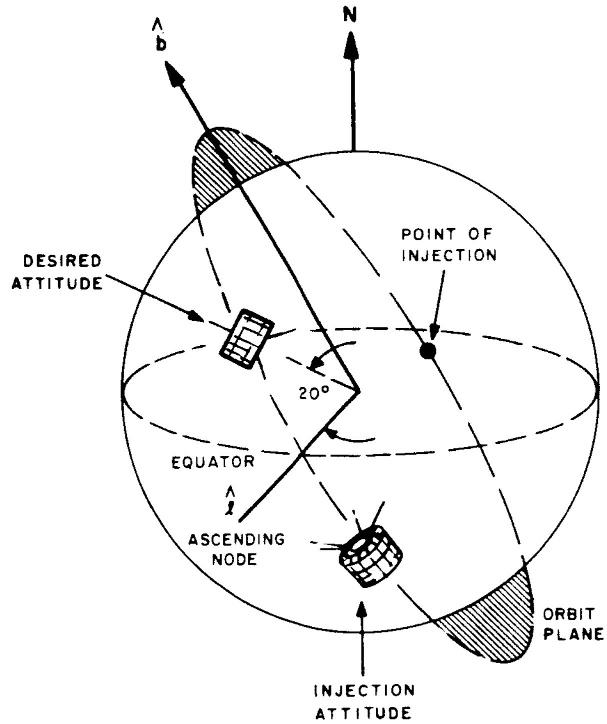


Figure I-1. TIROS X Turn-around Maneuver

A QOMAC clock is utilized to turn on current to the QOMAC coil. The camera clock first counts out the time delay, and alarms (as programmed via ground command), energizing the QOMAC clock which counts out quarter-orbit intervals for switching the dipole polarity of the QOMAC coil. For the 400-nautical-mile orbit planned for TIROS X, the QOMAC clock is preset to "count" 24.9-minute intervals, to an accuracy within 4 percent.

During the turn-around maneuver, the satellite's attitude is monitored by the TIROS ground system by means of telemetry data from the solar-aspect indicator and the attitude horizon scanner. (During this time, the exact orbital parameters are also determined, along with the status of the tracking and telemetry subsystem, the command and control subsystem, and other components within the spacecraft.)

The solar-aspect indicator provides a digital-coded measurement of the gamma angle, the angle between the satellite-sun line and the satellite spin axis. The attitude horizon scanner supplies data from which the spin-axis nadir angle is determined. The spacecraft attitude, then, is determined from simultaneous measurements of the gamma angle and the nadir angle, or from a continous analysis of the nadir-angle function with time.

As the TIROS X mission continues, nadir-angle control by means of the QOMAC coil is also used to obtain the desired picture coverage. For example, during the summer months, an orientation of the spin axis in the orbital plane directed at a 20° N declination would yield near optimum coverage of the hurricane belt.

After the desired attitude has been achieved with the turn-around maneuver, the spin axis is permitted to drift for several days so that the satellite's residual magnetic dipole moment can be measured. Based on this measurement, a constant torque is programmed in the MBC coil to negate the residual dipole moment. An additional torque is then programmed in the MBC coil to compensate for the 1-degree per day precession rate of the orbital plane in a sun-synchronous orbit.

When long-term observations indicate an attitude drift in a constant direction, the MBC system is used to offset the tendency to drift, thereby reducing the frequency with which QOMAC torquing need be applied.

C. CAMERA OPERATION

1. Modes of Operation

Three modes of operation are available with the TIROS X TV-picture subsystem: direct, remote, and playback.

Direct-picture requests are transmitted to the satellite while it is in contact with the CDA station. The transmitted command specifies the camera system to be used. The resultant video signals are read out in real-time, processed, and applied to the associated frequency-modulated TV transmitter for direct transmission to the ground.

The majority of pictures to be taken during the TIROS X mission, however, will be remote pictures, since the satellite is within communications range of the CDA stations for only a relatively short portion of each orbit. While the satellite is in contact with a ground station, the desired start time for a remote-picture sequence is programmed into the clock associated with the selected camera system, permitting TV coverage of an area of interest remote from the ground stations.

Pictures taken while the TV-picture subsystem is in the remote mode of operation are stored on magnetic tape. These pictures are played back and transmitted to ground when the satellite receives a playback command from a ground station.

The TIROS X camera clocks were modified to increase the remote picture interval from one picture every 30 seconds to one every 60. Thirty-two remote pictures are still taken by a camera in one remote sequence, but the change extends the coverage of each sequence by reducing the excess picture overlap that was obtained on TIROS I through VII and doubling the length of the swath covered in each sequence.

2. Direct Mode

To obtain direct pictures, the ground station transmits the designated spacecraft address followed by the direct camera command for either camera system. Upon receipt of this command, warm-up power is applied to the selected camera system for a specified period. At the end of the warm-up period, full operating power is applied to the selected system enabling the start of picture-taking operations by means of momentary interruptions in the direct-camera command. (When full operating power is applied, logic within the spacecraft causes automatic transmission of "housekeeping telemetry" over both beacon transmitters.) The interruptions in the direct-camera command are normally programmed to occur at 30-second intervals, and result in the actuation of the camera shutter, exposing the vidicon face-plate. During the 2-second interval following vidicon exposure, the image on the vidicon is read out, and the resulting video signal is applied to the recorder electronics where it frequency-modulates a video subcarrier oscillator. The resulting video subcarrier is applied directly to the activated TV transmitter, by-passing the tape transport. The resulting F-M/F-M modulated RF carrier is transmitted directly to the interrogating CDA ground station.

A direct camera sequence can be terminated either by simply ending the direct-camera command for a minimum specified period, or by replacing the direct-camera command with a playback command. In normal operation, the direct-camera sequence is terminated by the start of a playback sequence.

The operation of both camera systems is identical, as is the manner of commanding each for direct operation. However, camera system No. 1 is usually commanded for a direct sequence before a playback sequence is commanded, while camera system No. 2 is commanded after a playback sequence.

3. Playback Mode

The playback mode of operation is employed to permit playback of remotely taken pictures and to permit setting the satellite clocks for the next remote sequence (or for a QOMAC sequence). A playback sequence usually follows a direct sequence of camera system No. 1. However, it can be programmed to occur immediately after the satellite is addressed. In the former case, the start of playback will not result in the transmission of housekeeping telemetry; in the latter case, a telemetry readout will be initiated when full operating power is applied.

As in the case of direct camera operation, a warm-up period is required before the playback of data will begin. However, this warm-up period can be eliminated when playback of a camera system follows within a specified period after direct-camera operation of that same system. The elimination of such warm-up time is important because of the relatively short satellite-to-ground contact periods.

Playback operation is programmed by the transmission of a playback command to either of the two camera systems. After a playback command has been transmitted for the proper interval, power is applied to the selected tape recorder and recorder electronics and to the selected TV transmitter.

When power is applied to the recorder tape transport, playback commences and is synchronized by the 500-cps output from the associated camera clock. The video subcarrier is read-out and applied through the tape recorder electronics unit to frequency-modulate the associated TV transmitter.

When playback of the data is completed, the recorder stops automatically and, since the tape is erased during playback, the unit is ready for the next remote-picture-taking sequence; the second camera system can then be commanded for a playback sequence.

4. Remote Mode

The remote-picture-taking mode of operation is used for obtaining photocoverage while the satellite is orbiting over geographic areas out of communications range of a CDA station. This mode of operation is used to fulfill the majority of the requests for photocoverage, since the periods during which the satellite can be contacted are relatively short.

Either or both of the satellite's camera systems can be programmed for remote operation by setting and starting the associated camera clocks. The setting of the camera clock associated with a particular camera system is achieved during playback of that system by interruption of the playback tone with a series of clock-set pulses. Depending upon the number of pulses transmitted to the clock, the time between the starting of the clock and the start of the remote sequence can be varied from 0 to 5 hours in 2-second increments. After the clocks (or clock) have been set, they can be started by a command from either a CDA station or the clock-start station at Santiago, Chile.

When a satellite clock alarms, the associated TV camera and tape recorder are enabled and synchronizing signals are applied to both of these units. After the 1-minute period provided for warm-up of the camera and recorder, the camera clock supplies a 1-ppm shutter pulse to expose the camera vidicon and a signal to turn on the tape recorder during each 2-second vidicon-readout period. Remote pictures are taken at intervals of 60 seconds. At the conclusion of the 32-minute, 32-picture remote sequence, the clock turns off, removing power from the TV equipment and returning that equipment to a "standby" status. If the second camera system has also been programmed for remote operation, the sequence described above will be repeated at the programmed time and that system will also be returned to "standby". Both camera systems will remain in the "standby" mode of operation until the satellite is again interrogated by a CDA station and commanded to play back the stored data.

D. ATTITUDE MEASUREMENT

As the satellite orbits the earth, the attitude horizon scanner yields an indication of the spin-axis attitude with respect to the orbital plane (nadir angle), and the solar-aspect indicator yields an indication of the spin-axis orientation with respect to the satellite-sun line (the gamma angle).

The two infrared (IR) sensors in the attitude horizon scanner are independent units arranged in a "V" configuration; and, as the satellite spins, the optical axis of at least one sensor traces a conic section taking in both space and earth. The output of each head is supplied to a beacon transmitter for modulation on the beacon carrier. As noted, the attitude data is recorded at the CDA stations; and from a comparison of the earth-scan time to the spin period from one of the two heads, the satellite's attitude can be determined. *

Figure I-2 is a graphic representation of the orbit-attitude relationship of TIROS X at a number of different orbital positions. The cyclic variation of the satellite's

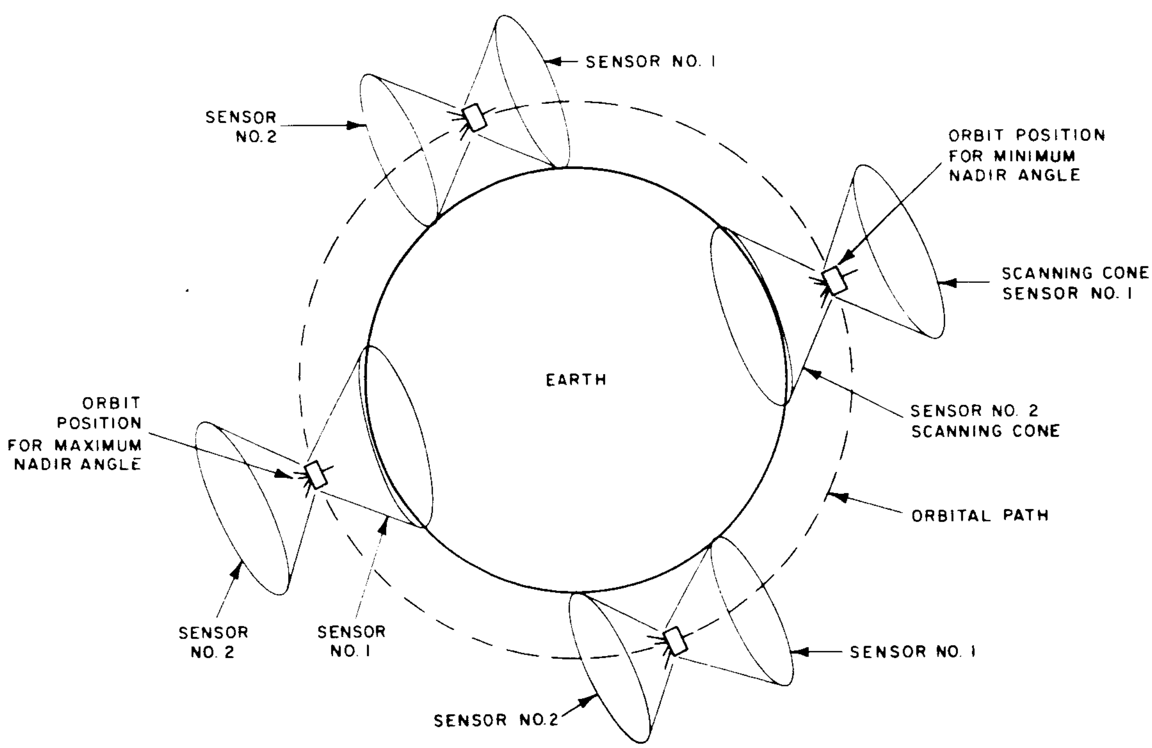


Figure I-2. Horizon-Sensor Scanning Cones with Respect to Earth for Various Positions in Orbit

*The technique used on the TIROS X system for the determination of spin-axis attitude is the intersecting cone technique, which has been set-up for use with a computer. Detailed information on attitude determination is available in the TIROS X and TIROS IX attitude manuals.

roll angle as it orbits the earth results in an earth-contact "history" as shown in Figure I-3 for both sensors comprising the attitude horizon scanner.

The solar-aspect indicator yields a digital-coded measurement of the gamma angle, i.e., the angle between the spin vector and the satellite-sun vector when the sun is in the plane defined by the two vectors. Gamma-angle measurements are used in attitude studies and in power availability studies.

During initial direct or playback sequences, solar-aspect data is automatically obtained during the 28-second warm-up period of the command camera system, and is applied to the associated beacon transmitter on the commanded side for transmission to the CDA station. During this 28-second period, three or four measurements of the gamma angle will be obtained.

The time period between the leading edges on consecutive measurements of the gamma angle is an accurate measure of the satellite's spin period. The time period between consecutive sky-earth transitions in either channel of attitude data can also be used as the nominal spin period.

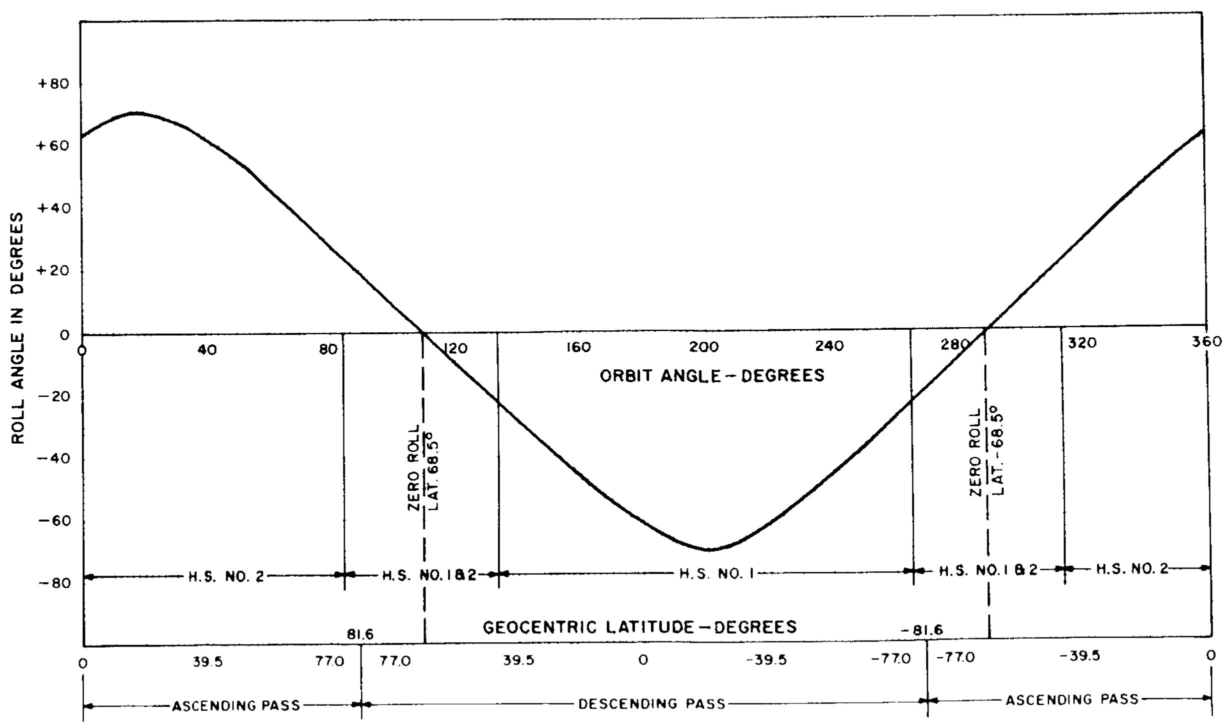


Figure I-3. Roll-Angle History and Attitude-Horizon-Scanner Coverage

E. TRACKING AND TELEMETRY

The tracking and telemetry operations of TIROS X are performed by means of a subsystem including two 50-mw beacon transmitters, two 1300-cps subcarrier oscillators (SCO's), and two 40-point telemetry switches.

As the satellite proceeds in orbit, beacon signals are transmitted to the ground to aid the CDA stations in tracking and acquiring the satellite as it enters communications range.

Beacon transmitter No. 1 operates at 136.23 Mc, No. 2 at 136.92 Mc. Each beacon carrier is amplitude-modulated by means of a frequency-modulated subcarrier oscillator to provide a means of transmitting attitude and telemetry data to the CDA stations. Attitude data from the "down-looking" head of the attitude horizon scanner is applied to beacon No. 1, while data from the "up-looking" head is applied to beacon No. 2.

When the satellite receives a command for a direct-picture sequence or a playback sequence, a time-shared series of telemetry data is automatically applied to the beacon transmitters replacing the attitude data. This data includes "housekeeping" telemetry, solar-aspect, MBC-switch-stepping, and QOMAC polarity. "Housekeeping" data can also be programmed by means of a specific command. A dual-channel paper-chart recorder is located at each CDA station for the recording of the data transmitted on the beacon signals.

"Housekeeping" telemetry data provides an indication of the status of various satellite operating parameters. The telemetry signals are obtained in the form of voltage levels from sampling points within the satellite and are applied to the telemetry switches, and thence to the associated beacon transmitters.

F. POWER REQUIREMENTS

During the course of the operational life of the satellite, the required electrical power is supplied by means of a solar-energy converter consisting of 9120 solar cells, and a battery pack consisting of 63 storage cells. During orbital day, the solar cells are used as the primary power source for the satellite's electrical system, while the battery pack is used during the orbital night to supply all power required by the satellite. In cases where peak-power requirements during orbital day exceed the power output of the solar cells, the battery pack automatically supplies the power difference. A patch of 60 solar cells on the top of the spacecraft hat provides telemetry indications of the overall condition of the solar-cell array, permitting CDA station monitoring of any solar-cell degradation. Each camera system has its own regulated power supplies which operate at -24.5 volts and -13.0 volts.

G. ANTENNAS

A single dipole antenna is used by the satellite for the reception of commands from the CDA stations. For transmissions to the ground, two dual-frequency, crossed-dipole antennas and an RF matching and coupling network are utilized. A notch filter is also included in the antenna subsystem to ensure isolation of the command receivers from the beacon transmitters.

During a CDA-station interrogation, the RF matching and coupling network couples the three operating transmitters (one TV transmitter and both beacon transmitters) to the radiating elements, provides an impedance match for the coupling, minimizes interaction and feedback between the transmitters, and effects circular polarization by exciting the elements in phase quadrature.

H. GROUND COMPLEX

The TIROS X system utilizes a ground complex comprising the following facilities:

(1) Two primary CDA (command and data acquisition) ground stations located at Fairbanks, Alaska, and Wallops Island, Virginia.

(2) A back-up station, used for engineering evaluation studies, located at the AED Space Center, near Princeton, N.J.

(3) The TIROS Technical Control Center (TTCC), located at NASA's Goddard Space Flight Center in Greenbelt, Maryland, at which the satellite programming for each orbit is formulated and transmitted to the primary CDA ground stations.

The command programs are transmitted to the satellite from the CDA stations. The satellite, in turn, transmits TV, attitude, and telemetry data to the CDA stations, where the data is processed and recorded and transmitted to the TTCC and to the other facilities associated with the TIROS ground complex.

SECTION III. ORBITAL CONSIDERATIONS

A. GENERAL

A circular, near polar, sun-synchronous orbit was planned for TIROS X because such an orbit provides greater picture coverage, consistent target illumination, and more favorable sun angles. The mechanics of the sun-synchronous orbit are the same as those of any circular earth orbit, with the unique exception that the orbital plane of the sun-synchronous orbit revolves around the earth's polar axis in the same direction and at the same rate as the earth-sun line. This nodal precession results in a nearly constant angle between the orbital plane and the direct rays of the sun, so that the satellite's subtrack on the surface of the earth receives nearly the same solar illumination on every orbit throughout the year.

A sun-synchronous orbit provides a nearly constant gamma angle (the angle at which the sun's direct rays fall on the satellite) with the advantages of nearly constant, annual orbital variations in satellite temperature, eclipse time, and energy supply from the satellite solar-cell array. Furthermore, because the variation in earth illumination remains nearly the same, the dynamic range of the satellite's camera systems can be adjusted for optimum performance over a limited illumination range.

B. ESTABLISHMENT OF A SUN-SYNCHRONOUS ORBIT

The geometry of a sun-synchronous orbit is shown in Figure I-4.

The lack of symmetry of the earth's gravitational field causes a gradual precession of the orbital nodes, westward if the orbital inclination is less than 90 degrees, and eastward if the inclination is greater than 90 degrees. Since the earth's movement around the sun produces an eastward movement of the earth-sun line, the inclination for the TIROS X orbit was set greater than 90 degrees.

The orbital inclination and altitude, together, determine the rate of precession of the orbital plane. The rate of precession of the earth-sun line is 0.986 degree per day, and the planned altitude and inclination for the TIROS X mission were set at 400 nautical miles and 98.36 degrees, respectively, to achieve a similar precession rate for the orbital plane. The TIROS X orbital plane, then, rotates eastward at a rate

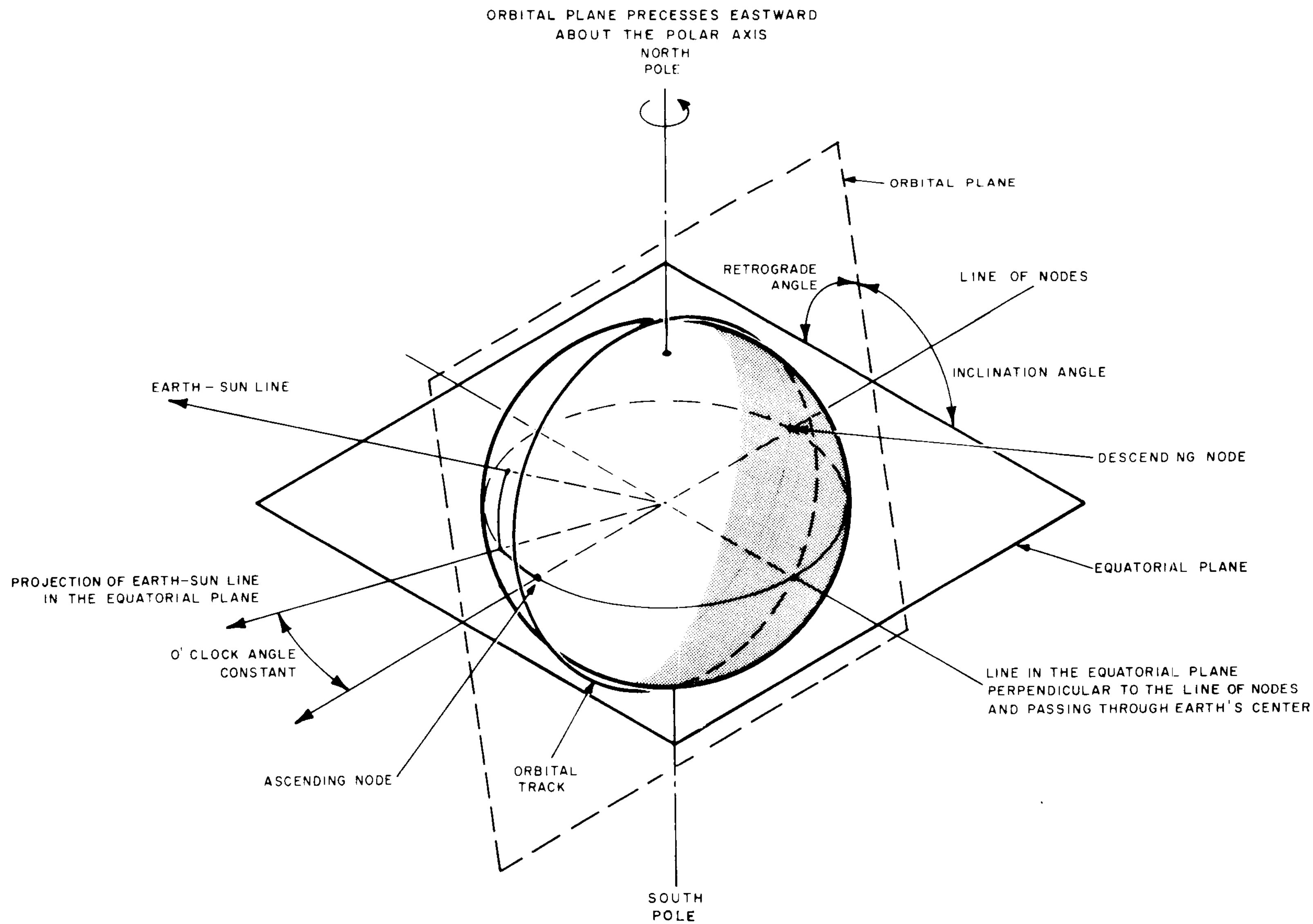


Figure I-4. Geometry of a Sun-Synchronous Orbit of approximately 1 degree per day, maintaining a constant "o'clock angle", i.e., the angle between the line of nodes and the line formed by the normal projection of the earth-sun line in the equatorial plane.

C. ORBITAL PARAMETERS

1. Selection Criteria

The fundamental parameters in the selection of the specific orbital parameters for TIROS X were the altitude and the time of the ascending node. Most of the other parameters were, within small variations, the results of the choice of these two fundamental parameters.

The planned orbit for TIROS X called for the following nominal characteristics:

• shape:	circular
• altitude:	400 nautical miles
• inclination:	98.36 degrees
• nodal period:	99.63 minutes

The considerations affecting the choice of an orbital altitude included (1) the use of the Delta DSV-3C launch vehicle and (2) the fact that a spacecraft launched southward into a retrograde orbit from the Eastern Missile Range must be guided in a "dogleg" trajectory around the tip of Florida.

2. Altitude Effects on Spacecraft Operation

Since the spacecraft design for TIROS X is predicated on the 400-nautical-mile orbit, the choice of altitude affects the area coverage of the TV cameras and picture resolution, in addition to the operation of the attitude horizon scanner, the solar-aspect indicator, and the path losses with the TV and beacon transmitters.

At the planned altitude, the number of orbits per day is not an integer; and the satellite, therefore, does not trace the same lines of longitude on each day. Consequently, the areas covered by the satellite at specific times on successive days are not identical, and the pictures taken at those times cover different areas.

3. Drift Rate of Orbital Plane Precession

At the 400-nautical-mile orbital altitude planned for TIROS X, an inclination of 98 degrees is required to yield a precession rate which will make the orbit sun-synchronous, i.e., an orbital precession rate of 1 degree per day. Any error in altitude or inclination at the point of satellite injection would cause an error in the precession rate. (Inclination errors would have the greater effect.)

Data obtained from six previous TIROS launches indicated that an average drift in the precession rate of 0.05 degree per day could be expected and that orbit time would drift approximately 20 minutes after 100 days.

4. Time of Ascending Node

a. General

The time of the ascending node determines both the angle of the sun's rays relative to the orbital plane, and the direction in which the satellite travels when it crosses the equator in daylight. The main influences of the time of the daylight crossing are on the illumination of the orbital path and on the illumination of the satellite. Thus, the time of the ascending node affects not only the scene illumination for picture taking but also two satellite operational characteristics: namely; the temperature variation of the satellite and the power output of the satellite's solar-cell array.

The most important consideration in choosing a daylight ascending node crossing for TIROS X was the speed with which remotely taken pictures would reach a ground station for processing and use. The location of the TIROS X CDA stations in the upper part of the northern hemisphere was the determining factor in this consideration. The time the satellite passes over a ground station is related to the time of acquiring remote picture sequences from any area of the earth. As the earth rotates on its axis and the various areas of the world are brought into the sunlight under the satellite's orbital path, the remote pictures of both hemispheres are most quickly acquired by the CDA stations if the daylight node is an ascending node. A daylight ascending node is considered particularly advantageous for immediate playback of pictures of the tropical hurricane regions and of the Near and Far East, Europe, and the Atlantic Ocean; and, since the TIROS X launch was scheduled to provide coverage during the 1965 Atlantic hurricane season, this was an important consideration.

b. Time of Daylight Crossing

(1) General

The choice of the time of daylight crossing will fix, for a given time of the year, the gamma angle, which in turn is limited by restraints such as thermal considerations and power availability. The following details the factors limiting the choice of gamma angle and the effect of the time of daylight crossing on the gamma angle.

(2) Factors Affecting Choice of Gamma Angle

(a) Thermal Considerations

The temperature limits for TIROS X were considered to be as follows:

- on the spacecraft top and sides: 85°C
- on the spacecraft components: -10°C to + 35°C.

The 35°C level was considered the limit for spacecraft components because rapid chemical degradation of the batteries can occur at temperatures above this level.

Figure I-5 is a polar plot showing the relationship between gamma angle and temperature level for both the spacecraft top and sides and the spacecraft components.* It can be seen from this graph that the 35°C thermal limit specifies the 20 to 70 degree range for acceptable gamma angles for long-term mission operation.

(b) Power Availability

Figure I-6 is a polar plot showing the relationship between gamma angle and power output of the TIROS X solar-cell array. The shape of this curve reflects the change in total power output of the array as the projected solar-cell area charges with each gamma-angle value. The reference curve for the typical load assumes 70 ampere minutes of power are required from the array on each orbit to maintain the capability to obtain and playback a complete 32-picture remote sequence while avoiding excessive battery discharge. (A value of 80 percent for the charge/discharge efficiency of the batteries is also assumed.)

As can be noted in Figure I-6, from power-supply considerations alone, the gamma angle is limited to a value in the range 0 to 90 degrees (which excludes the marginal values from 90 to 105 degrees).

(c) Camera Operation

Figure I-7 is a polar plot showing the maximum field-of-view of the TIROS X TV cameras. As can be noted in this Figure, this field-of-view is 104.8 degrees; and the rotating satellite "sweeps out" a cone of a 52.4-degree half angle,

*For this analysis, component power dissipation, Q_C, has been assumed to be 25 watts.

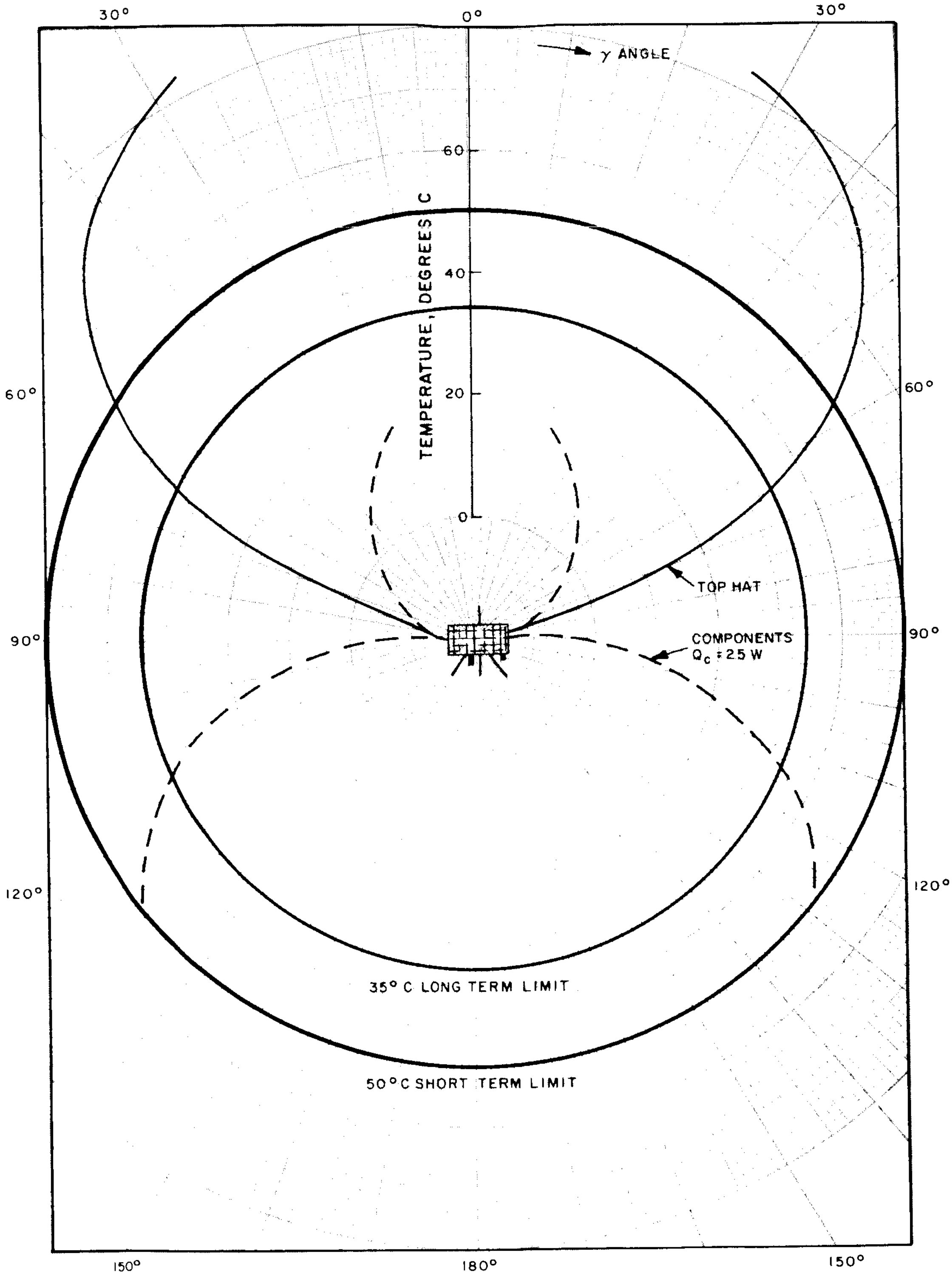


Figure I-5. Gamma Angle Versus Temperature for the TIROS X Mission

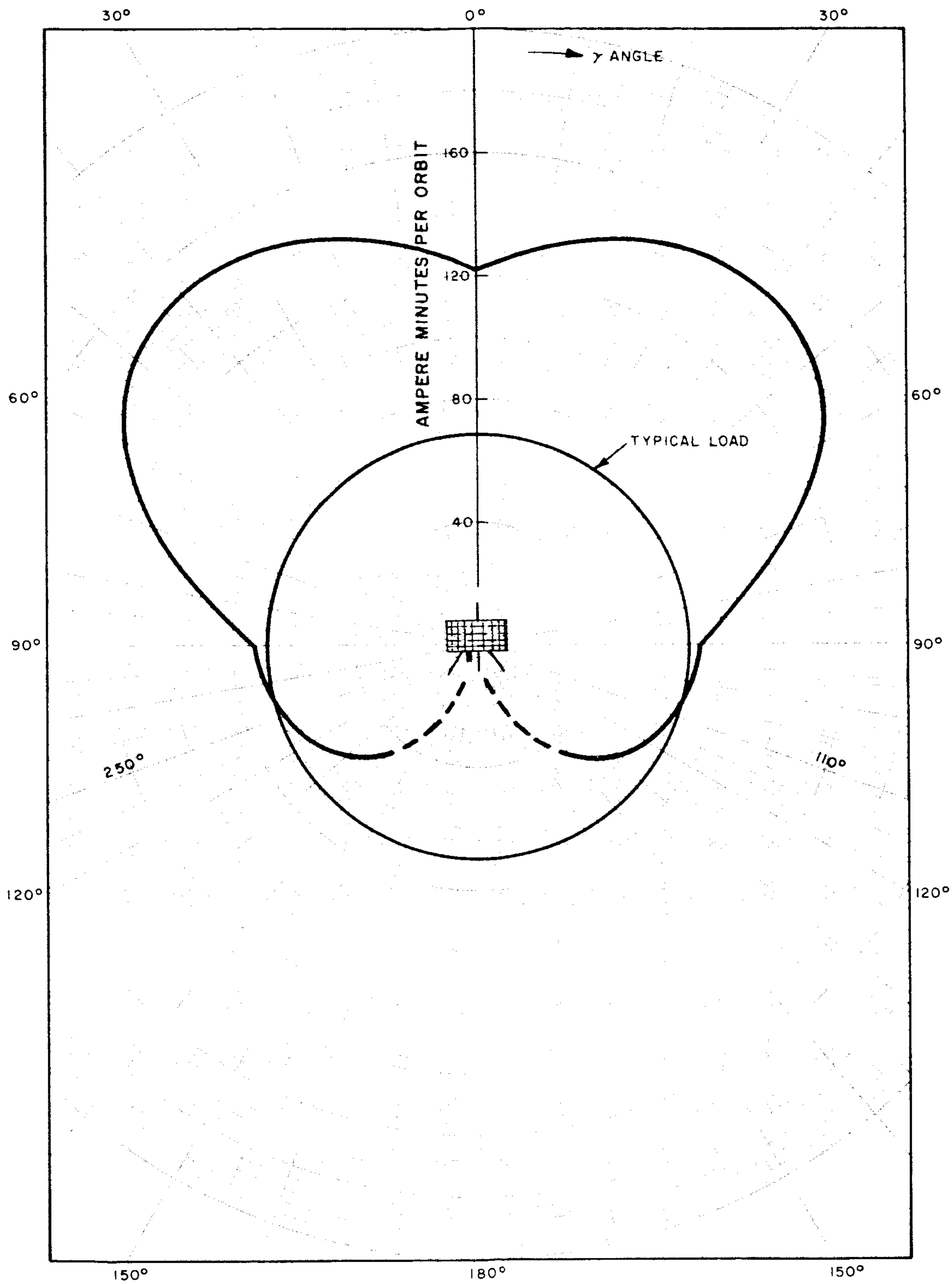


Figure I-6. Gamma Angle Versus Solar-Cell Array Power Output for the TIROS X Mission

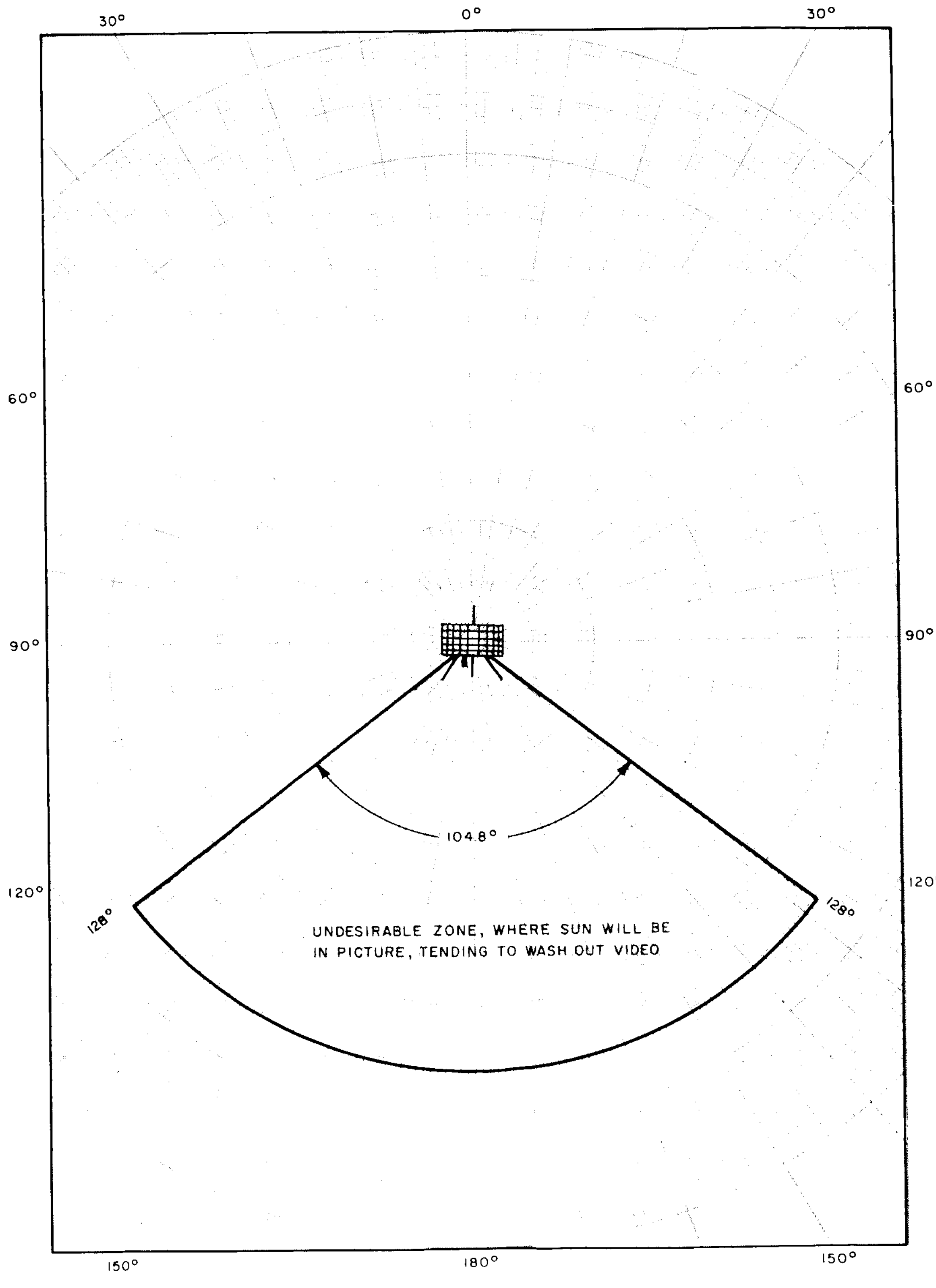


Figure I-7. Maximum Field-of-View of TIROS X TV Cameras as Related to Choice of Gamma Angle centered about the 180-degree point. If the gamma angle is above 128 degrees, the sun will be within the field-of-view of the cameras; the sun's rays will then be continuously impinging on the camera lenses and, if picture-taking is commanded, the resultant video will tend to "wash out." In addition, if the sun is within the field-of-view of the cameras, thermal problems in the lens and shutter must be considered.

(d) Attitude Measurement

The TIROS X attitude horizon scanner comprises two independent IR sensors mounted on the baseplate at an included angle of 80 degrees. Sensor No. 1, the "up-looking" sensor, is mounted with its optical axis at a 50-degree angle to the satellite spin axis; sensor No. 2, the "down-looking" sensor, at a 130-degree angle.

When scanning space, both sensors will experience sun interference which could be severe enough to distort or completely obscure the horizon-crossing indications in the attitude data. Sensor No. 1 is equipped with a sun filter because, as can be noted in Figure I-2, this sensor will be "looking" into space more often than sensor No. 2. As shown in Figure I-8, for sensor No. 1, there is a 10-degree range, centered about the optical axis, within which the sensor can experience sun interference equivalent to the minimum earth signal; for sensor No. 2, this range is 32 degrees. The use of the sun filter on sensor No. 1 cause the difference in the possible-interference ranges between the two sensors. The sun filter attenuates the response of the sensor to the earth signal, but attenuates the response to the sun to a much greater degree.

The interference ranges about the optical axes of the sensors, i.e., the range from 45 to 55 degrees for sensor No. 1 and that from 114 to 146 degrees for sensor No. 2, must be avoided to prevent sun interference on the TIROS X attitude data.

(e) Summary

From the foregoing it can be observed that the mission-mode gamma angle on TIROS X should fall between 20 and 70 degrees, avoiding the range between 45 and 55 degrees, if possible, to avoid sun interference on attitude data from attitude sensor No. 1. Gamma angles above 70 degrees are undesirable because of thermal constraints, sun interference with attitude sensor No. 2 data, power-supply restraints, and camera-lens considerations. Similarly, gamma angles from 0 to 20 degrees are undesirable because of thermal constraints.

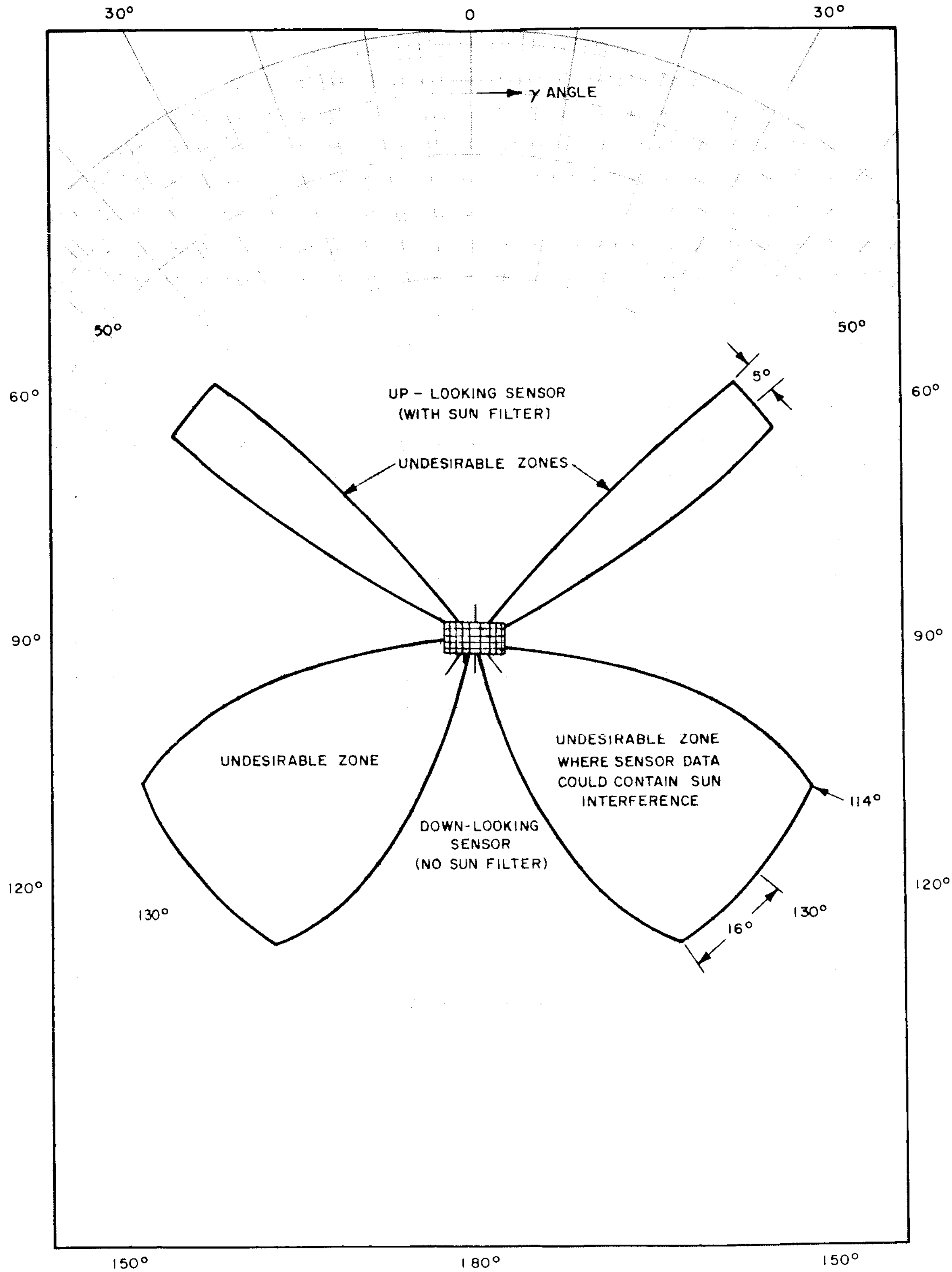


Figure I-8. Operational Zones of TIROS X Attitude Horizon Scanner as Related to Choice of Gamma Angle

SECTION IV. PHOTOCOVERAGE FOR THE TIROS X MISSION

A. DYNAMIC COVERAGE

The Northern Hemisphere was of primary interest to the TIROS X mission because of the land masses located in this hemisphere and because of the locations of the TIROS CDA stations. The two primary CDA stations in the TIROS X program are located at Fairbanks, Alaska, and Wallops Island, Virginia.

To enable convenient visualizing of the orbital track of the satellite on the earth's surface (both Northern and Southern Hemispheres) with respect to the camera coverage and illumination boundaries, a sub-satellite point ephemeris for three successive orbits at an altitude of approximately 400 nautical miles has been superimposed on a Mercator projection of the earth and presented in Figure I-9. In this figure, the sub-satellite-point locations in 2-minute intervals are indicated by a series of marks, and for reference, the two primary CDA stations and their 5-degree contact circles are also indicated.

B. SCENE ILLUMINATION

Due to the spherical shape of the earth, the solar elevation angle varies with latitude for a given time of year, and because of the tilt of the earth's axis with respect to the ecliptic, this variation changes as the earth moves around the sun. The time of the nodal crossing determines the change in solar-elevation angle along the orbital track, as well as the annual variation in solar-elevation angle at any specific point along the orbital track.

The elevation angles along the satellite subtrack at the solstices and at the equinoxes are shown for a 1400-hours ascending nodal crossing in Figure I-10 and for a 1500-hours ascending nodal crossing in Figure I-11. The plots are the same as for a 1000-hours and a 0900-hours descending node, respectively. From these curves it is apparent that higher illuminations of the sub-satellite point result from the 1400 (1000) hours nodal crossing. Even higher values would result as a "high-noon" nodal crossing is approached. However, these higher values would be obtained at the cost of relatively "warm" satellite temperatures and lowered energy output of

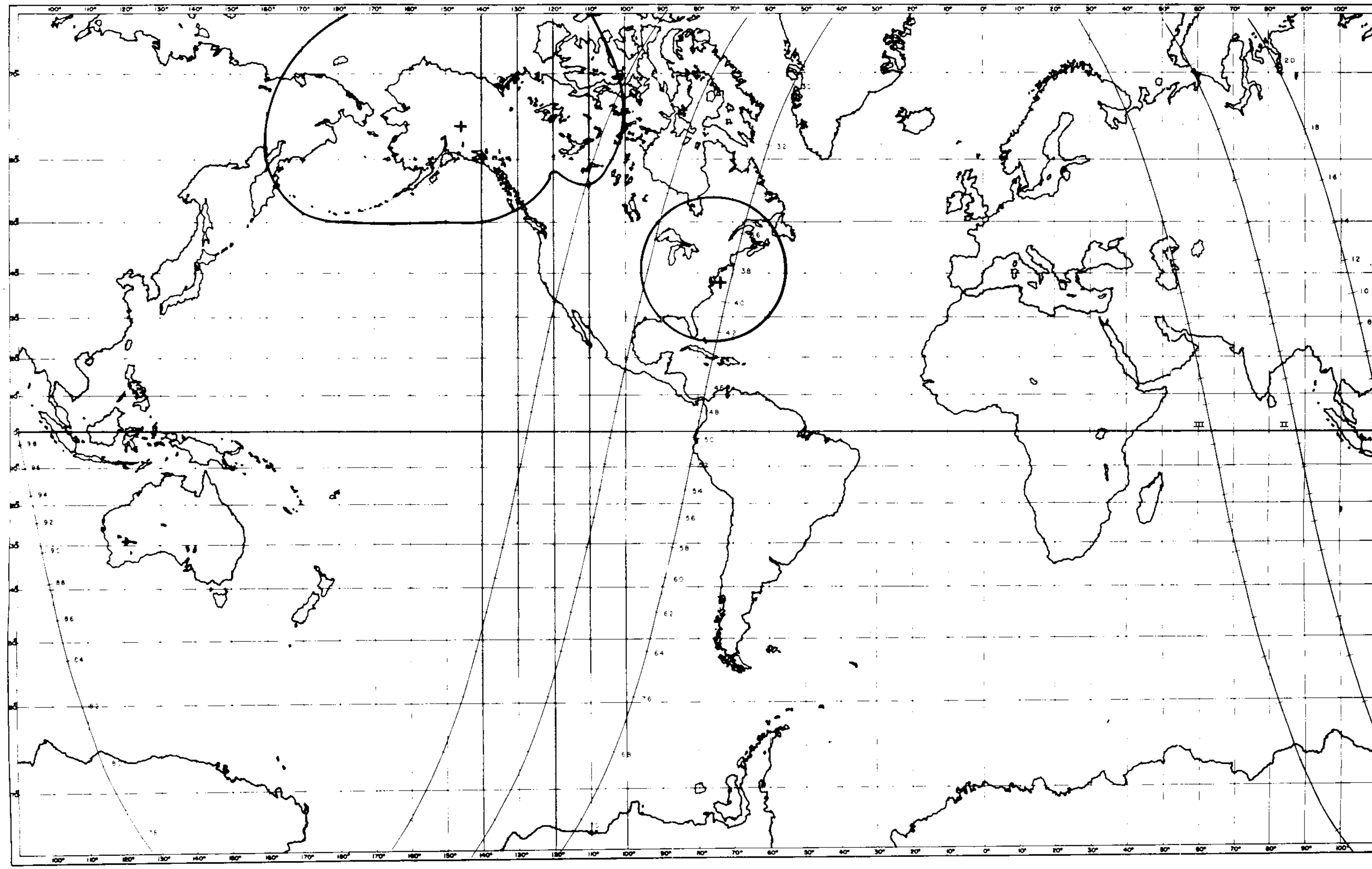


Figure I-9. Mercator Projection of World, Showing Three-Orbit Ephemeris for TIROS X

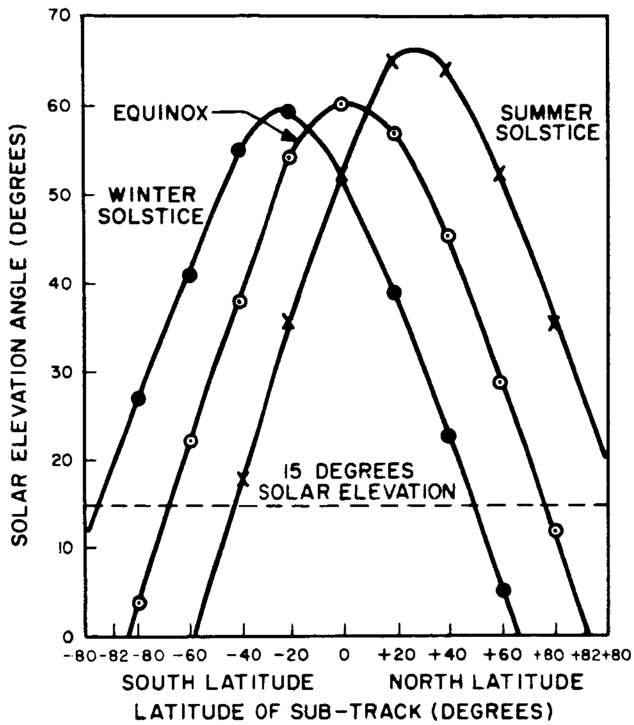


Figure I-10. Latitude Versus Solar-Elevation Angle for a 400 N. Mi., Circular Orbit, at a 98-Degree Inclination, for a 1400-Hours Ascending Node

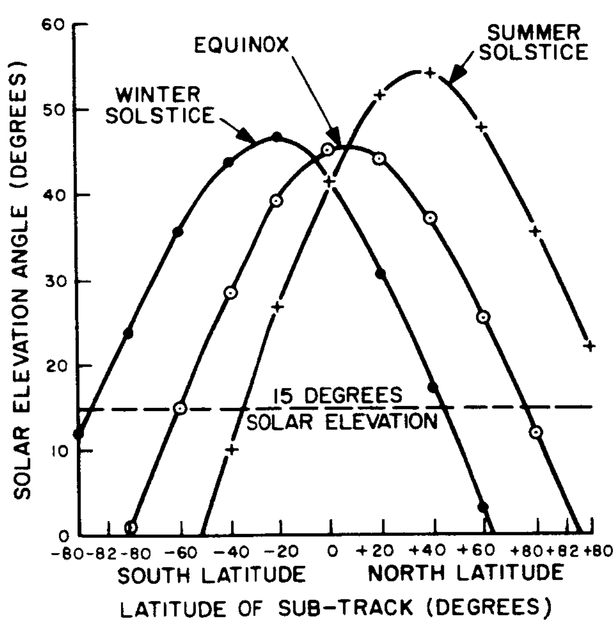


Figure I-11. Latitude Versus Solar-Elevation Angle for a 400 N. Mi., Circular Orbit, at a 98-Degree Inclination, for a 1500-Hours Ascending Node the satellite solar array. Furthermore, it would be necessary to "adjust" the dynamic range of the camera system to compensate for a wider variation of light input. Since the dynamic range for the camera system is a constant, the only "adjustment" that could be made would be to accentuate the higher brightness levels at the expense of the lower brightness levels.

Since storm coverage is an important consideration, the solar-elevation angles between latitudes 5° N and 45° N were considered. During the months of (approximately) June through November, the hurricane activity is primarily in the northern hemisphere and lengthening the viewing time over this area is desirable.

Another important consideration with regard to illumination is the total distance along the orbital subtrack over which the solar elevation angle is greater than 15 degrees. The value of 15 degrees is used as a minimum solar-elevation angle for obtaining useful pictures. As can be seen in Figures I-10 and I-11, this distance is not significantly different at the equinoxes or at the solstices for the 1400 and 1500 hours nodal crossings.

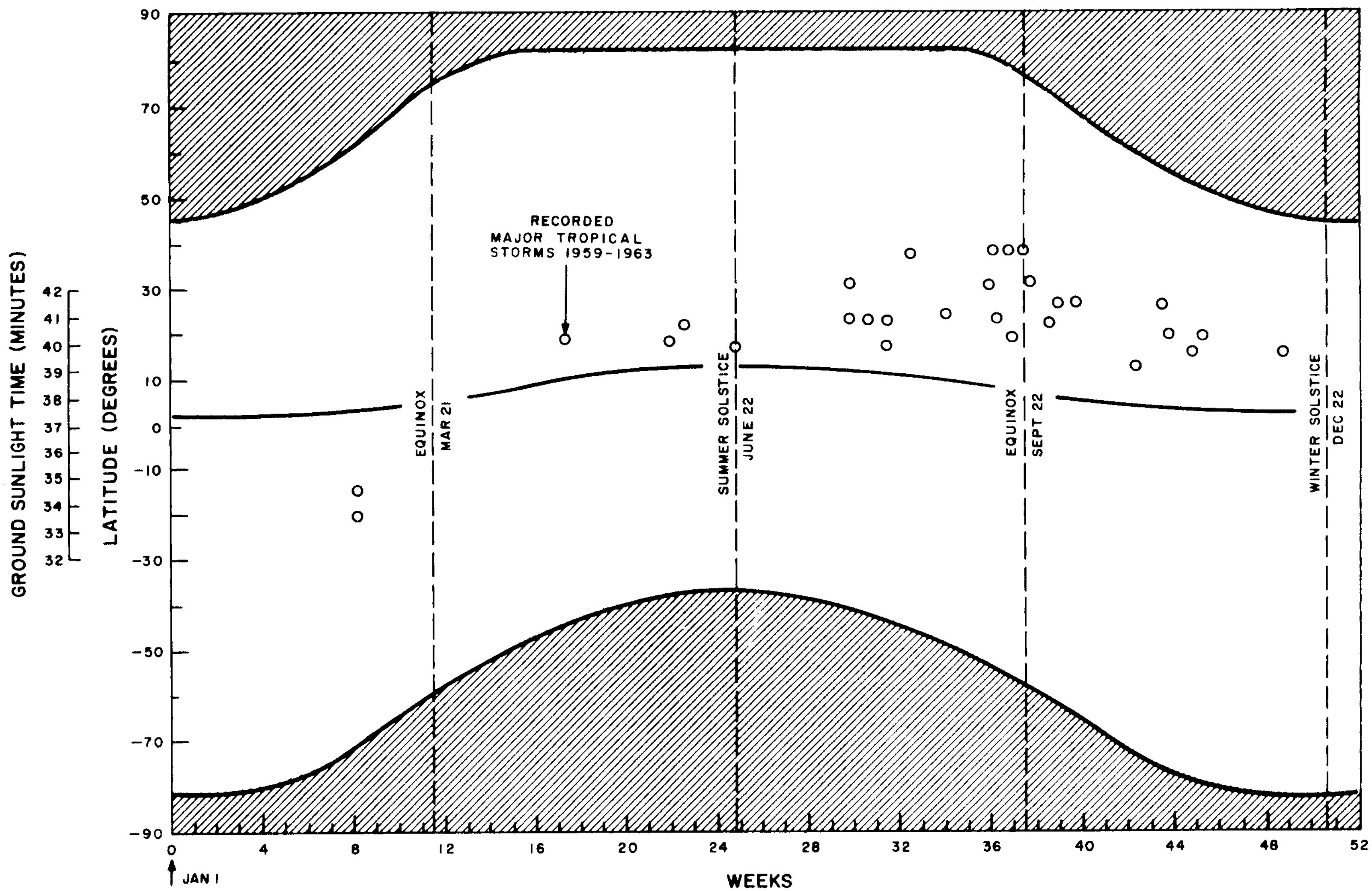
The 15-degree solar-elevation-angle boundaries for an orbit with a 1500-hours ascending node are plotted in Figure I-12 for an entire year. Such an orbit offers desirable overall illumination characteristics and good duration of illumination of the northern hemisphere. The curve in the center of the illustration indicates that, for this orbit, the variation in ground sunlight time over the course of the year is only slight.

C. SCENE COVERAGE

Projections of the TIROS X field-of-view upon the earth's surface are shown in Figure I-13. The particular conditions illustrated are for an orbit with a 1430-hours ascending node, and a NON location of 20°S (Winter). As an arbitrary choice for purposes of the illustration, the midpoint of the 32-minute camera sequence has been centered at 20°S. The particular cases illustrated are for camera-center zenith angles of 42°, 27°, and 0°, which are represented at 70°S (and 40°N), 47°S (and 6°N), and 20°S, respectively. Due to the spinning of the satellite and the axial configuration of the cameras, the direction of view tends to vary from picture to picture in the same manner as on all standard TIROS satellites.

D. LATITUDE COVERAGE

From a brief survey of geographic locations of storms, hurricanes, typhoons, etc., occurring from June through September, it can be seen that the major area of interest



NOTE: BOUNDARIES ARE AT 15 DEGREES SOLAR ELEVATION

Figure I-12. Annual Illumination Boundaries for a 1500-Hours Ascending Node, 98-Degree Inclination, 400-N. Mi., Sun-Synchronous Orbit

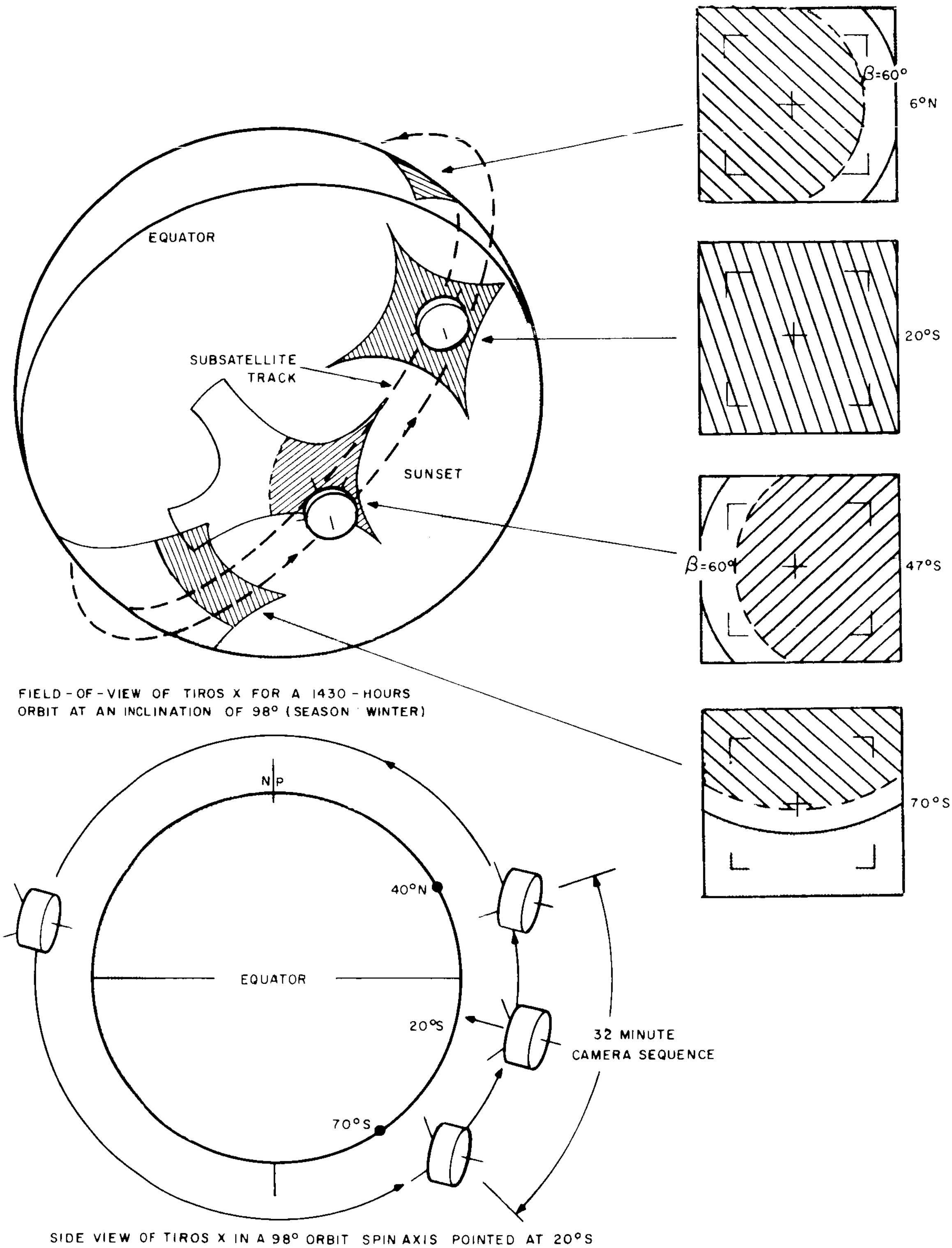


Figure I-13. TIROS X Field-of-View Upon Surface of the Earth for picture coverage is in the northern hemisphere between the 5° and 45° latitudes. In the case of a 1500-hours ascending node, the optimum camera coverage would be obtained with a minimum nadir angle occurring at 20° N latitude.

Based on the time dependence of the locations of storm centers, the spin axis could be tilted down, toward 20° S latitude in November, to cover the southern hemisphere, and in the following May, reoriented for covering the northern hemisphere. This procedure would offer the advantage of following the solar-illumination patterns.

E. GROUND-STATION CONTACT

The command and data acquisition range of each of the TIROS CDA ground stations was shown in Figure I-9 by the 10-degree elevation contact circles of the antennas at the respective stations.

The minimum command time required by both the Wallops Island and the Fairbanks stations is approximately 4.5 minutes. This allows sufficient time for satellite address and warm-up (approximately 46 seconds) and readout of a complete remote sequence plus clock-set for the following sequence (approximately 2 minutes). Under best conditions, the maximum command time available to the interrogating CDA station will be approximately 10 minutes.

On a typical day during the operational life of TIROS X, the Wallops Island station, alone, will be able to command the satellite on three orbits, and the Fairbanks station, alone, will be able to command the satellite on eight orbits. In addition, there will be one orbit on which either CDA station will be able to acquire the satellite, but because of the elevation angle neither will be able to command it; there will also be a three consecutive orbits on which no contact with the satellite will be available.

PART II. SATELLITE DESIGN AND TESTING

PART II. SATELLITE DESIGN AND TESTING

SECTION I. DESIGN OF SATELLITE COMPONENTS

A. INTRODUCTION

The decision to reconfigure spacecraft OT-1 to meet the requirements of the TIROS X mission involved adapting a spacecraft which was originally planned for an orbit with a 58° inclination to a configuration which could be used in a near-polar, 98° inclination orbit.

As originally assembled, spacecraft OT-1 was a standard TIROS spacecraft, and as such was very similar to TIROS VII. The modifications made to spacecraft OT-1 were initiated after assembly and debugging had been completed for the original configuration, and the modifications were planned so as to involve a minimal effort.

In the original configuration, dummy weights had been substituted for the IR equipment such as had been used on TIROS VII. The final configuration also included dummy weights for this equipment, but because of the new layout it was necessary to relocate the weights on the spacecraft baseplate.

In addition, the north-indicator subsystem was electrically disconnected in the reconfiguration, though, to facilitate balancing of the spacecraft, the subsystem equipment was not physically removed.

The equipment added to the spacecraft in the reconfiguration included (1) a second, independent IR sensor, which together with the sensor already on the spacecraft, constituted the TIROS X attitude horizon scanner, (2) the QOMAC system, i.e., QOMAC coil, clock, and control unit, and (3) a digital solar-aspect indicator. These new equipments were added to increase the attitude-control capabilities on TIROS X, and were based on units proven in use on TIROS IX.

Figure II-1* presents a logic diagram of the final configuration of spacecraft OT-1. The basic information-gathering equipment on the spacecraft comprises a standard TV-picture subsystem, consisting of two wide-angle camera systems. In addition, the spacecraft includes the following subsystems:

*Because of its size, this illustration is placed at the end of this volume.

- the command and control subsystem,
- the tracking and telemetry subsystem,
- the reference-indicator subsystem,
- the dynamics-control subsystem,
- the power-supply subsystem, and
- the antenna subsystem.

Various modifications to existing equipment on spacecraft OT-1 were also made as part of the reconfiguration. The most significant of these was on the camera clocks, changing the interval between successive pictures in a remote sequence from 30 seconds to 60 seconds. With this change, the length of a 32-picture remote sequence was doubled, i.e., from 16 to 32 minutes, extending the coverage of each sequence and eliminating excessive overlap between successive pictures.

Other changes and modifications made to spacecraft OT-1 for the TIROS X program are detailed in the following descriptions of individual subsystems.

The physical dimensions of all TIROS satellites, including TIROS X, are the same, i.e., a TIROS satellite is an 18-sided polyhedron with a height of 22 inches and a diameter of 42 inches.

B. TV-PICTURE SUBSYSTEM

1. General

The TV-picture subsystem comprises two identical camera systems, each including a 1/2-inch vidicon camera and associated electronics, two tape recorders, and two 2-watt TV transmitters.

A block diagram of the TIROS X TV-picture subsystem is contained in Figure II-2.

Two picture-taking modes are available with the camera systems: direct and remote. In the direct mode of operation, TV pictures are taken in direct response to ground command and are transmitted directly to the ground station initiating the direct-picture command. Remote operation is employed for gathering cloud-cover information over areas where the satellite cannot be contacted by a ground station. In this mode of operation, pictures are taken in response to commands previously programmed into the camera clock associated with the selected camera system. Pictures taken while

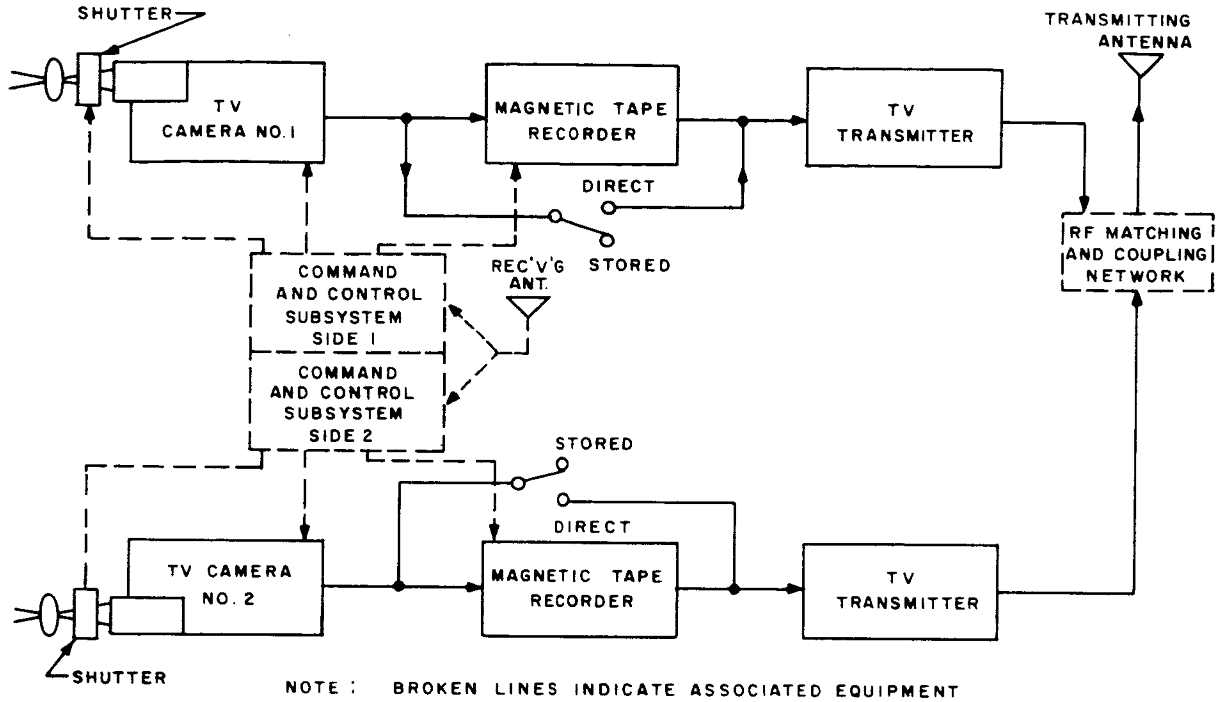


Figure II-2. TIROS X TV-Picture Subsystem, Block Diagram

the TV-picture subsystem is in the remote mode of operation are stored on magnetic tape. These pictures are played back and transmitted to ground when the satellite receives a Playback command from a TIROS ground station.

The TV cameras utilize a ruggedized 1/2-inch vidicon designed especially for satellites The useful frame area of the vidicon faceplate is 1/4 by 1/4 inch. A crossed-line reticle marks the center of the vidicon, and corner lines are inscribed as additional references. The vidicon is exposed for each picture by means of a solenoid-operated focal-plane shutter. The electrical output is coupled through a preamplifier to a video amplifier. The output of the video amplifier is frequency-modulated on an 85-kc subcarrier, then either recorded on magnetic tape or transmitted to the ground, depending on the mode of operation. Some of the more significant characteristics of the TV cameras are presented in Table II-1.

The tape transport mechanism in the TIROS tape recorders was developed especially for satellite operation, from the standpoint of weight and power consumption as well as electronic performance. While recording, the mechanism operates only when individual photographs are being recorded; during playback, each recorder runs continuously in the playback direction until all the tape is transferred to the storage reel and an automatic end-of-tape switch is actuated, preparing the unit for the next record cycle. The tape is erased immediately after playback and again passes the erase head (in the other direction) just before recording.

TABLE II-1. CHARACTERISTICS OF TIROS X TV CAMERAS

Lens Manufacturer	Elgeet
Lens Angle	104°
Lens Relative Aperture	f/1.5
Shutter Speed	1.5 msec
Lines per Frame	500
Vertical Sweep Duration	2 sec
Video Bandwidth	62.5 kc
Power Consumption (Average)	9 watts
Weight	6.5 pounds

2. Functional Description

a. TV Cameras and Camera Electronics

Figure II-3* presents a logic diagram of a TIROS X TV camera and camera-electronics unit. The standard TIROS TV camera, such as those in the TIROS X TV-picture subsystem, utilizes a focal-plane, travelling-slit shutter, driven by a moving coil in a magnetic field. The shutter blade is accelerated quickly, travels across the face of the vidicon at nearly constant speed, and is stopped at the end of travel by a silicone-rubber bumper. The exposure time is nominally 1.5 milliseconds.

A vertical-deflection sawtooth waveform is generated at a rate of 0.5 pulse-per-second. The amplified vertical-deflection voltage is applied to the vertical winding of the yoke. Centering and size controls are included to permit the position of the scanned area to be changed to accommodate the individual vidicon being used.

The vidicon power supply contains a regulated dc-to-dc converter to provide high voltages and a current regulator for the focus-coil and filament currents. The supplied voltages are minus 20 and minus 30 volts for the control grid, 20 to 30 volts for the target, 250 volts for electrostatic focus, and 300 volts for grid No. 2. A current regulator supplies 100 milliamperes, ±1 percent, for the magnetic-focus coil and the

*Because of its size, this illustration is placed at the end of this volume.

vidicon heater, which are connected in series. The remaining bias voltages and preamplifier heater and plate voltages are obtained directly from the satellite's solar-conversion power supply.

A beam-current regulator is included to ensure that the intensity of the scanning beam remains relatively constant and is unaffected by factors such as vidicon aging, or ion contamination. The regulator performs its function by changing the voltage applied to the control grid of the vidicon whenever there is a tendency for the beam current to change.

b. Tape Recorders

(1) General

Each of the TIROS X magnetic tape recorders consists of a tape transport (R1), a signal-electronics unit (R-3), and a power-electronics unit (R2). The power-electronics unit contains the end-of-tape logic circuits. The purpose of this unit is to provide operating power to the tape transport during record and playback sequences. The tape transport and the signal-electronics unit function as an integral component during record and playback.

The tape transport consists of two coaxial reels, a capstan, and an idler mounted in a triangular configuration. The capstan and idler are angled and guide the tape as it moves from the plane of one reel to that of the other. A constant-tension spring assembly between the reels maintains uniform tape tension regardless of the amount of tape on each reel. The capstan is belt-driven by a synchronous hysteresis motor. Recording is performed by means of RCA-VR502 heads, and playback by means of RCA-VR402 heads. A permanent magnet provides the erase capability.

The tape transport contains a polyester tape 1.5-mil thick, and 0.375 inch wide, and operates at a tape speed of 50 inches per second. The power requirements for the signal-electronics are (1) in direct operation: 24.5 vdc, 45 ma; (2) in playback operation: 24.5 vdc, 10 ma; and (3) in record operation: 24.5 vdc, 40 ma. The power requirements for the power-electronics are (1) in playback operation: 26 vdc, 0.9 ampere; and (2) in record operation: 26 vdc, 0.9 ampere. (The power-electronics unit is not used when the camera system is operating in the direct mode.)

(2) Direct Mode

Figure II-4* is a logic diagram of the TIROS X tape recorder. During direct operation, the signal-electronics unit is enabled by operating power from the TV camera and command-control unit for the selected camera system. Power from

*Because of its size, this illustration is placed at the end of this volume.

the TV camera is applied to the 10-kc gated oscillator, emitter follower, and fast rise-time circuit. The fast rise-time circuit is enabled during the 2-second readout period and applies operating power to the modulator and head drivers. Power from the command-control unit is applied to the mixer and amplifier-limiter.

DC video signals from the camera are applied through an emitter follower to the video modulator circuit, where the amplitude variations of the video signal produce frequency variations of an 85-kc subcarrier oscillator. The composite video signal, a frequency-modulated signal, is applied through the amplifier-limiter to frequency-modulate the TV transmitter. *

(3) Record Mode

During the record cycle, i.e., remote operation, picture regulated power from the TV camera is applied to the 10-kc oscillator, emitter follower, and, during each readout period, through the fast rise-time circuit to the modulator and head drivers. Also during remote operation, unregulated power is applied to the power-control circuit of the power-electronics unit, by means of the associated camera control unit.

Nominally one second before the vidicon shutter is actuated, unregulated power is applied through the power-control circuit of the power-electronics unit to the 500-cps converter. This converter is synchronized by the 500-pps output of the time-base generator (in the camera clock) and supplies a 500-cps, 440-volt squarewave through the motor-reversing relay to the synchronous motor of the tape transport. Because the motor-reversing relay is de-energized during record, the squarewave applied to the motor windings causes the tape transport to move in the record direction.

The composite video resulting from vidicon readout is applied through the emitter follower to the modulator and produces frequency variations in the 85-kc subcarrier oscillator. This FM signal is then applied through the head driver to the video record head of the tape recorder.

At the end of the record period, the fast rise-time circuit is disabled and, in turn, disables the modulators and head drivers. Also, unregulated power is removed from the power-electronics unit, causing the tape transport to halt operation until 1 second before the next shutter trigger. When the overall remote sequence ends, regulated power is removed from the signal-electronics unit, causing the tape transport to stop. When the overall remote sequence ends, regulated power is removed

*The 10-kc oscillator for channel 2 of the recorder was used on TIROS VII with the north-indicator subsystem, but is not used on TIROS X; the composite video signal is, therefore, free of the 10-kc carrier.

from the signal-electronics unit, the recorder power control is switched to playback and all three components of the tape recorder remain disabled until the start of playback.

(4) Playback Mode

During the playback mode of operation, operating power is applied to the tape recorder from the command-control unit. Playback unregulated power is applied to the power-electronics unit and enables the 500-cps converter and energizes the motor-reversing relay. With the relay energized, the 500-cps signal from the converter is coupled to the motor winding and causes the motor to run in the playback direction.

The data recorded is read out by the video-playback head and amplified in the playback amplifier before being applied to frequency-modulate the TV transmitter signal. The video signal is amplified by the amplifier-limiter and is applied to frequency-modulate the TV transmitter.

When tape playback is complete, an end-of-tape signal is sent from the tape transport to the power-electronics unit. This switches the recorder to the record state and interrupts playback power to the motor; consequently, the tape transport stops. Because the tape is erased as it is played back, the tape recorder is ready for the next remote sequence.

c. TV Transmitters

The TV transmitters used on TIROS X are 2-watt units identical to those used on TIROS VII. Figure II-5 is a block diagram of the 2-watt TV transmitter. The signal from the TV camera is an amplitude-varying signal obtained from the camera vidicon with frequency components from 0 to 62.5 kc. This video signal frequency-modulates an 85-kc voltage-controlled oscillator (VCO) up to ±15 kc. This constant-amplitude, frequency-modulated video subcarrier then frequency-modulates the video transmitter 2.25-Mc VCO ±130 kc. The result, then, is an FM/FM video signal for transmission to the ground stations.

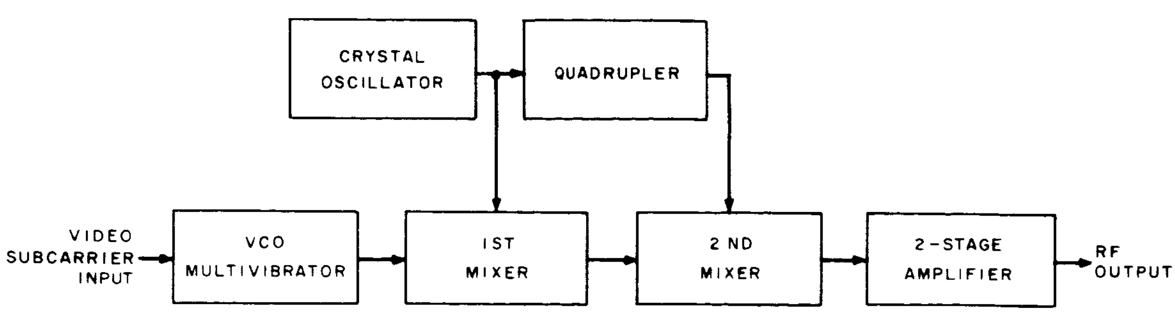


Figure II-5. TV Transmitter, Block Diagram

3. Testing

a. TV Cameras

Camera No. 1, Serial No. 3-201-01, and its associated electronics unit, Serial No. 3-203-01, were successfully subjected to a vibration test on September 10, 1963. During a post-vibration check on September 14, 1963, it was determined that vidicon F-1 had developed an open heater. A new vidicon was installed and the camera was successfully re-vibrated on September 19, 1963. Camera No. 1 and its electronics were subjected to thermal-vacuum testing from September 19 through September 22, 1963, with satisfactory results.

Camera No. 2, Serial No. 3-202-02, and its associated electronics unit, Serial No. 3-203-02, were successfully subjected to a vibration test on September 22 and 23, 1963. The post-vibration check revealed that vidicon F-2 had developed tube spots. After replacement of the vidicon, the camera and electronics successfully passed a second vibration test on September 27, 1963. A thermal-vacuum test on the camera and electronics was performed from October 3 through 6, 1963, with satisfactory results.

b. Tape Recorders

(1) Camera System No. 1

The recorder transport, Serial No. 3-301-02, underwent vibration testing on August 29, 1963, and thermal-vacuum testing from September 12 through 15, 1963, with satisfactory results. The units were modified with the new end-of-tape leaders in April 1965, after the unit was initially delivered for integration on spacecraft OT-1. (The new leaders were first added to the TIROS IX recorders.)

The tape recorder power converter, Serial No. 3-302-03, underwent vibration testing on September 5, 1963, with satisfactory results, and thermal-vacuum testing was successfully completed on September 28.

The tape recorder signal electronics, Serial No. 3-303-03, successfully completed vibration testing on August 20, 1963, and thermal-vacuum testing on September 2, 1963.

(2) Camera System No. 2

The recorder transport, Serial No. 3-301-01, successfully completed vibration testing on August 22, 1963, and thermal-vacuum testing on August 29, 1963. The modified end-of-tape sensors were incorporated in April 1965, after the unit was initially released for integration on spacecraft OT-1.

The tape recorder power converter, Serial No. 3-302-01, successfully completed vibration testing on August 21, 1963, and thermal-vacuum testing on August 31.

The tape recorder signal electronics, Serial No. 3-303-01, successfully completed vibration testing on August 19, 1963, and thermal-vacuum testing on September 2.

c. TV Transmitters

TV transmitter No. 1, Serial No. 3-401-07, was successfully subjected to vibration testing on January 30, 1964. On February 5, the unit completed thermal-vacuum tests with satisfactory results.

TV transmitter No. 2, Serial No. 3-401-04, successfully completed vibration tests on November 14, 1963, and thermal-vacuum tests on November 20.

d. TV Transmitter Filters

The TV transmitter filters, Serial Nos. 3-402-02 and 3-402-03 successfully completed vibration testing on August 30, 1963, and thermal-vacuum testing on September 17.

e. DC/DC Converters

The DC/DC converters, Serial Nos. 3-702-01 and 3-702-05, successfully completed vibration tests on July 18, 1963, and thermal-vacuum tests on July 26.

C. COMMAND AND CONTROL SUBSYSTEM

The material on this subsystem is contained in Volume II, the classified supplement to this report.

D. TRACKING AND TELEMETRY SUBSYSTEM

1. General

The TIROS X tracking and telemetry subsystem includes two beacon transmitters, which operate at 136.23 Mc (beacon No. 1) and 136.92 Mc (beacon No. 2). A constantly powered 1300-cps subcarrier oscillator (SCO) is associated with each transmitter. The subsystem also includes two telemetry switches which sample 39 data points each, at a nominal sampling rate of 0.8 seconds per point. The switches sample the same data points, but in differing sequence, and the resulting telemetry is designated "housekeeping" telemetry. Figure II-6 presents a block diagram of the TIROS X tracking and telemetry subsystem.

The following time-shared series of operational data is transmitted to the CDA ground stations by means of the beacon transmitters: attitude data, solar-aspect data, "housekeeping" data, QOMAC-dipole-polarity data, and MBC (Magnetic Bias Control)-switch position data. A summary of TIROS X telemetry data is presented in Table II-2.

During the initial launch period, the beacon No. 2 (136.92 Mc) SCO is biased to approximately 1400 cps as an indication that the lift-off switches are closed. As soon as the satellite separates from the third-stage rocket, the lift-off switches open and the SCO returns to a center frequency of 1300 cps.

As shown in Figure II-6, the TIROS X attitude horizon scanner is composed of an "up-looking" sensor (No. 1) and a "down-looking" sensor (No. 2). Attitude data from each sensor is continuously applied to the associated SCO and beacon transmitter, except during the intervals noted below.

TABLE II-2. SUMMARY OF TIROS X TELEMETRY DATA

Telemetry	Telemetry Voltage Range (volts)	SCO Frequency (cps)	Time of Occurence	Duration
Solar Aspect Angle	Binary "1" : 0 Center Frequency: -1.25* Binary "0": -2.50	1310* 1350* 1400*	During warm-up of commanded "side" when satellite is illuminated by sun and $0 \leq \gamma < 128^\circ$	30 sec*
"Housekeeping"	±2.5* (Both Channels)	1200 to 1400*	After warm-up of (either) commanded "side"	32 sec*
MBC Switch Position	0 to -2.5	1300 to 1400*	Upon command for stepping of MBC switch	Duration of Attitude Tone
QOMAC Coil Dipole Polarity	———	1300 1310* 1317*	No QOMAC Cycle Positive Dipole in QOMAC Coil Negative Dipole in QOMAC Coil	Approx 25 min each
Attitude Horizon Scanner	±2* (Both Channels)	1200 to 1400* (Both Channels)	Interrupted on both channels for "housekeeping" telemetry and on either channel for solar-aspect or MBC switch position data	Continuous
Lift-Off Switch	-2.5	1393*	Until satellite has separated from third-stage rocket	Orbit 0000** only

* Approximate value

** Orbit 0000 is defined as occurring between the time the satellite is ejected into orbit and the time of the first ascending node

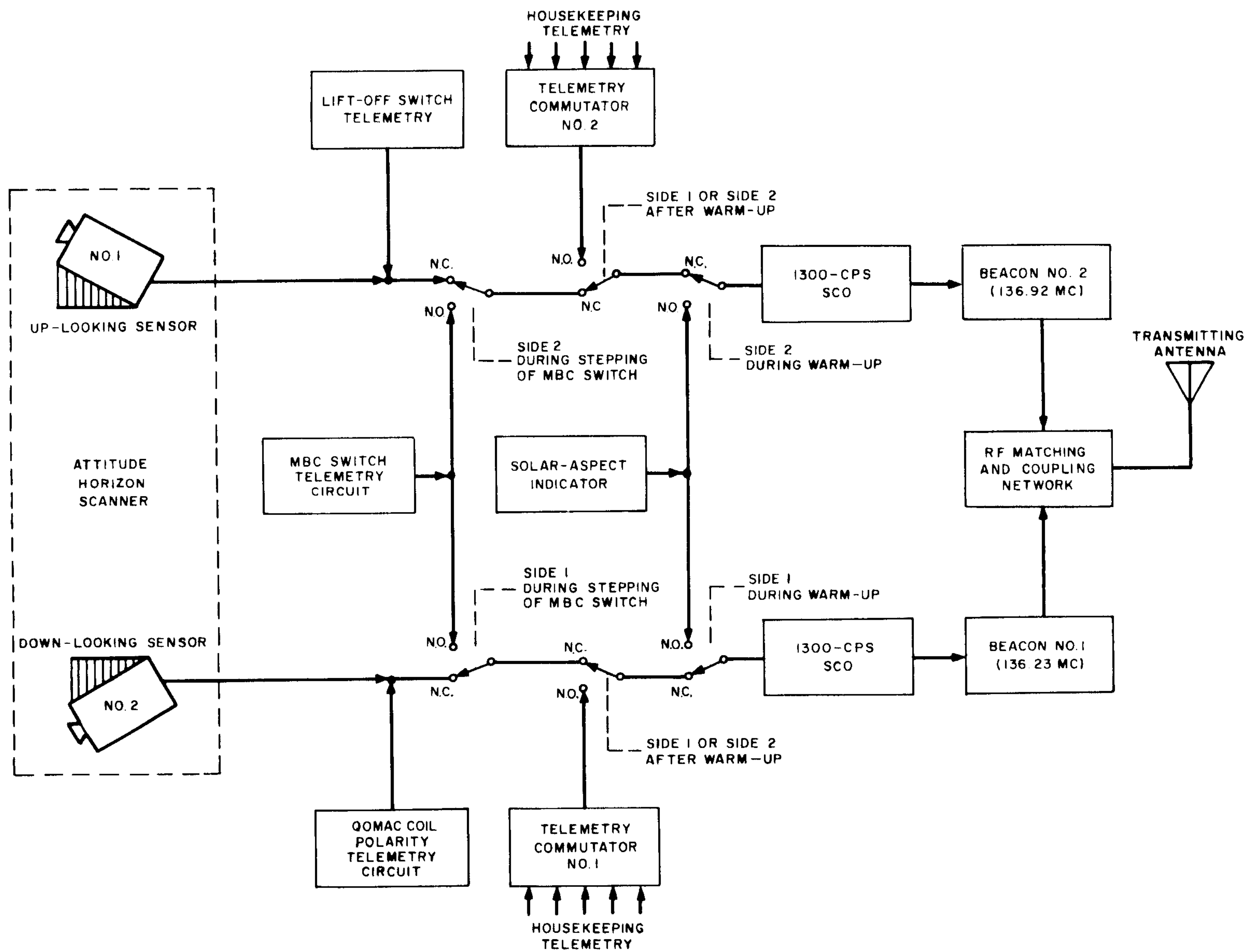


Figure II-6. Tracking and Telemetry Subsystem, Block Diagram

While a command for stepping of the MBC switch is being sent to the satellite, attitude-horizon-scanner data is removed from the beacon on the commanded "side" and replaced with MBC-switch telemetry. For each position of the MBC switch there is a specific associated magnetic dipole produced in the MBC coil; and, consequently, there is a specific telemetry voltage associated with each switch position. The solar-aspect indicator is energized and the resultant data coupled to either beacon transmitter at the initiation of the warm-up period of the associated command "side". During the warm-up period, three or four separate measurements of the gamma angle can be made by the solar-aspect indicator.

The two telemetry switches are automatically energized immediately after warm-up of either command "side," or upon a telemetry command from a CDA station; and "housekeeping" telemetry from each commutator replaces the attitude data on the associated beacon while the 39 data points are being sampled. Table II-3 presents the channel assignments for each of the TIROS X telemetry commutators.

When a QOMAC-coil operating cycle is programmed, the attitude-horizon-scanner data on beacon No. 1 is biased, according to the polarity of the QOMAC dipole, for the duration of the cycle. As indicated in Table II-2, a positive dipole in the QOMAC coil will cause the attitude data to be shifted from a normal center frequency of 1300 cps to approximately 1310 cps; a negative dipole will cause the data to be shifted to approximately 1317 cps. When the QOMAC cycle is completed, the center frequency returns to 1300 cps.

2. Equipment Description

a. Beacon Transmitters

The TIROS X beacon transmitters are solid-state units which operate at 136.23 Mc (beacon No. 1) and 136.92 Mc (beacon No. 2), each with a power output of 50 milliwatts. The transmitters are crystal-controlled, with a carrier frequency stability of 0.005 percent.

The power input to the beacon transmitters is supplied at -24.5 volts dc, and the output signal from each unit is an amplitude modulated RF signal. The use of amplitude-modulation of the beacon carriers ensures the presence of the RF carrier for tracking operations and the absence of sideband components within 1 kc of the carrier (as necessitated by the requirements of the NASA Minitrack network).

b. SCO's

The two SCO's utilized in the TIROS X telemetry and tracking subsystem are solid-state units operating on IRIG Channel No. 5 (1300 cps). The specifications for these units are as follows:

Deviation	±7.5%
Deviation bandwidth, DBW	15% of F_c

Input impedance	750 kilohms, minimum
Input sensitivity	0 to -5.0 VDC with -2.5V producing F_c
Output impedance	50 kilohms, maximum
Linearity	±0.25%
Distortion	Less than 1%
Amplitude modulation	1.0 db
Output voltage	Adjustable from 0 to 1.5 volts peak-to-peak into an 8 kilohm load
Supply voltage	-24.5 VDC (20 ma)
Supply voltage stability	Internal regulator limits F_c deviation to less than 0.15% DBW-per-volt change in input voltage
F_c drift, long term	Less than ±1% of DBW at 22°C
Temperature stability	F_c stable within ±1.5% DBW over temperature range of -20°C to +85°C.

c. Telemetry Switches

The TIROS X telemetry switches are identical to the units used on TIROS VII. The 40 positions covered with each switch include 39 data points and a "home" position, and the points are sampled at a nominal sampling rate of 0.8 second per point. Each unit operates at an average input power of 100 milliwatts, and a supply voltage of -24.5 volts dc (2.5 amperes, peak). As can be noted in Table II-3, the channel assignments for each switch include four sampling points which provide calibration voltages as a means of facilitating the interpretation of "housekeeping" telemetry data.

3. Testing

a. Beacon/SCO Assemblies

The beacon transmitters and the SCO's were purchased from commercial manufacturers by AED. These units were subjected individually to qualification testing by their manufacturers. After being subjected to bench testing at AED, the units were integrated into the beacon/SCO assemblies and subjected to qualification testing before being released for integration with the spacecraft.

TABLE II-3. TIROS X TELEMETRY COMMUTATOR CHANNEL ASSIGNMENTS

Telemetry Switch Position No.	Commutator No. I (136.23-Mc Beacon)	Commutator No. II (136.92-Mc Beacon)
	Parameter	Parameter
1	Calibration: -2.5 Volts	Calibration: -2.5 Volts
2	QOMAC Clock Power	500-cps Converters for Tape Recorders
3	Calibration: -1.0 Volt	Calibration: -1.0 Volt
4	QOMAC Sync Pulses	Temperature, QOMAC Coil ‡
5	Calibration: -2.0 Volts	Calibration: -2.0 Volts
6	Calibration: 0 Volt	Calibration: 0 Volt
7	"X" String Battery Output: -28 Volts	Regulated -24.5 Volts, System No. 1
8	"Y" String Battery Output: -28 Volts	Regulated -24.5 Volts, System No. 2
9	"Z" String Battery Output: -28 Volts	Regulated -13 Volts, System No. 1
10	Load Bus: -28 Volts	Regulated -13 Volts, System No. 2
11	Regulated -24.5 Volts, System No. 1	"X" String Battery Output: -28 Volts
12	Regulated -24.5 Volts, System No. 2	"Y" String Battery Output: -28 Volts
13	Regulated -13.0 Volts, System No. 1	"Z" String Battery Output: -28 Volts
14	Regulated -13.0 Volts, System No. 2	Load Bus: -28 Volts
15	Vertical Sync Pulse, System No. 2	Solar-Cell Array Output Voltage
16	Horizontal Sync Pulse, System No. 2	Temperature, Base ‡
17	Vertical Sync Pulse, System No. 1	Solar-Cell Array Output Current
18	Horizontal Sync Pulse, System No. 1	Temperature, Base †
19	Vidicon High Voltage, Systems 1 and 2	Filament and Focus Current, Vidicon No. 1
20	Temperature, Base ‡	Temperature, Base *
21	Filament and Focus Current, Vidicon No. 1	Filament and Focus Current, Vidicon No. 2
22	Filament and Focus Current, Vidicon No. 2	Vidicon High Voltage, Systems 1 and 2
23	Rocket Switch "Home" Position	Temperature, Solar-Cell Patch *
24	Temperature, Hat, 3-in Radial *	Solar-Cell Patch Voltage
25	Temperature, Hat, 12-in Radial *	Temperature, TV Camera No. 2 ‡
26	Solar-Cell Patch Voltage	Temperature, TV Xmtr No. 1 ‡
27	Temperature, Base †	Temperature, Beacon Xmtr No. 2 ‡
28	500-cps Converters for Tape Recorders	QOMAC Sync Pulses
29	Temperature, QOMAC Coil *	QOMAC Clock Power
30	Temperature, Side Panel *	Temperature, Camera Clock No. 2 †
31	Temperature, Solar-Cell Patch *	Temperature, Battery Pack †
32	Solar-Cell Array Output Voltage	Vertical Sync Pulse, System No. 2
33	Temperature, Base *	Horizontal Sync Pulse, System No. 2
34	Solar-Cell Array Output Current	Vertical Sync Pulse, System No. 1
35	Temperature, TV Camera No. 2 †	Horizontal Sync Pulse, System No. 1
36	Temperature, TV Xmtr No. 1 †	Temperature, Hat, 3-in Radial *
37	Temperature, Beacon Xmtr No. 2 ‡	Temperature, Hat, 12-in Radial *
38	Temperature, Camera Clock No. 2 ‡	Temperature, Solar-Cell Patch *
39	Temperature, Battery Pack ‡	Rocket Switch "Home" Position
40	"Home" Position	"Home" Position

* Temperature Sensor Range: -30 to +100°C
† Temperature Sensor Range: -20 to +10°C
‡ Temperature Sensor Range: +10 to +40°C

Vibration testing on beacon No.1, Serial No. 3-406-01, was successfully completed on July 17, 1963, and thermal-vacuum testing on July 21.

Vibration testing on beacon No. 2, Serial No. 3-405-02, was successfully completed on July 30, 1963, and thermal-vacuum testing on August 16.

b. Telemetry Switches

The two telemetry switches, Serial Nos. 3-902-05 and 3-902-06, were subjected to vibration testing on September 3, 1963, with satisfactory results.

E. REFERENCE-INDICATOR SUBSYSTEM

1. General

The TIROS X reference-indicator subsystem comprises the solar-aspect indicator and the attitude horizon scanner. (A north-indicator subsystem, such as that included on TIROS VII, was also installed on spacecraft OT-1; but this subsystem was disconnected for the TIROS X mission.)

Attitude data from the attitude horizon scanner (nadir angle data) and the solar-aspect indicator (gamma-angle data) is transmitted from the satellite via the beacon transmitters to the ground stations, where it is displayed on a Sanborn recorder, and reduced and interpreted.

2. Solar-Aspect Indicator

a. General

The solar-aspect indicator utilizes a Gray-coded light mask which produces direct digital readings of the γ angle, i.e., the angle between the satellite-sun line and the satellite spin axis. The readings are presented as serial, Gray-coded words. The solar-aspect indicator has a 1-degree resolution over a range of 128 degrees. It consists of two separate subassemblies: the sensing element (the aspect indicator) and an electronics package. The sensing element consists of a 7-bit Gray-coded reticle equipped with a small solar cell under each bit, and a double-slitted "command" reticle which also includes a solar cell. The electronics package contains an amplifier for each bit, bi-stable multivibrators to establish thresholds and to convert the parallel input to a series output, and control circuits. The command reticle causes the angle determined by the Gray-coded reticle to be read out when the sun is in a plane defined by the spin axis and the satellite-sun line passing through the aspect eye.

b. Aspect Indicator

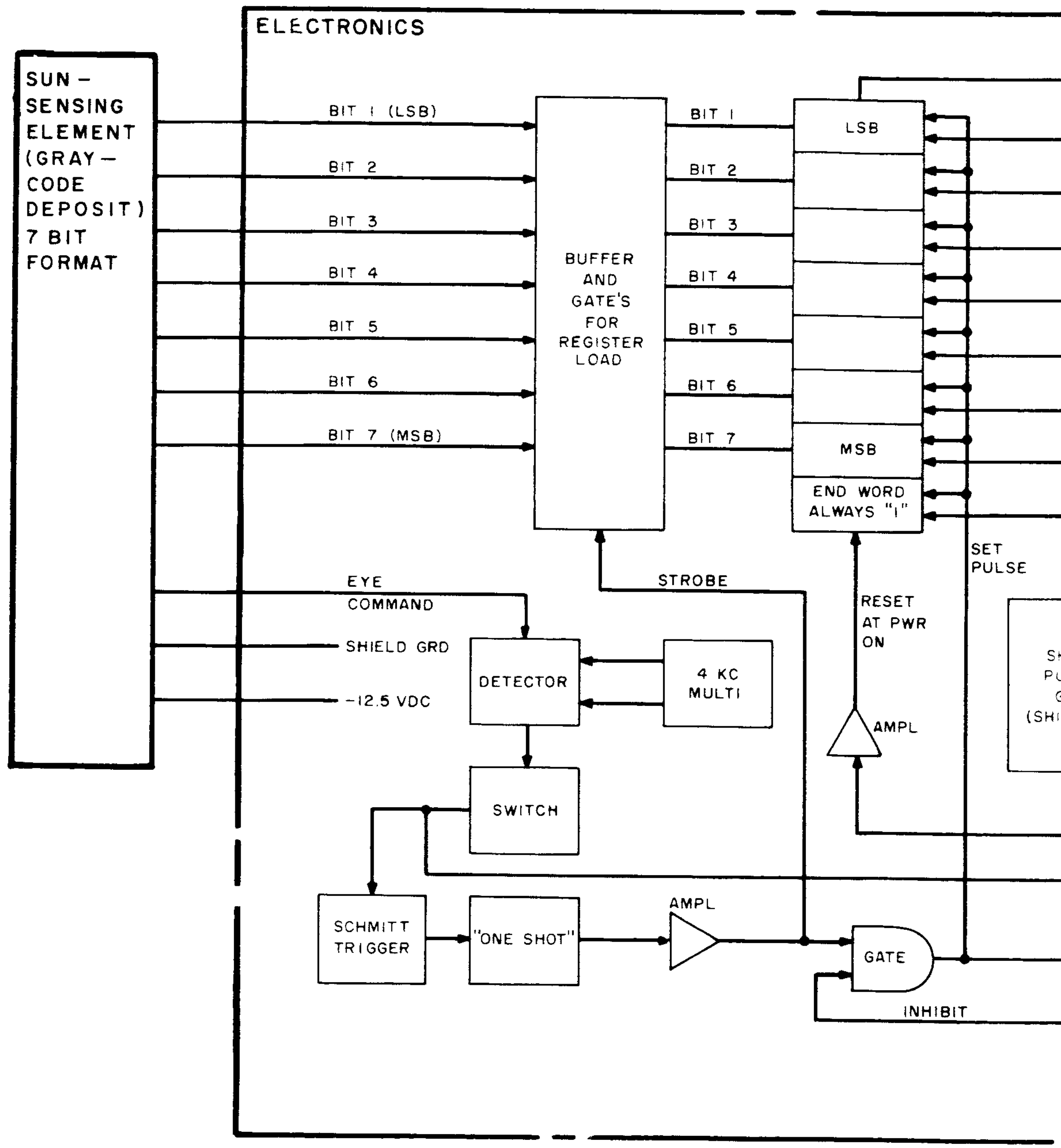
The reticle is a small oblong block of fused quartz with a slit centered along the top surface, and a light mask, arranged in a Gray-coded pattern, photographically applied to the bottom surface. Gray code was chosen for the light-mask pattern to eliminate any errors in the measurement of the gamma angle caused by errors in the synchronization of the data bits. Such errors could be easily noted in the Gray code since only one data bit would change when the measured angle changes one degree. Sunlight enters the indicator through the slit in the top surface of the reticle, casting a narrow band of illumination across the light mask. Each of the seven bits on the Gray-coded reticle is superimposed on a photocell which detects the light (or notes the absence of light) in the aperture and produces a corresponding electrical output signal. P-on-N silicon solar cells are used as the photosensitive units. Although the output signal from this type of cell is relatively small (on the order of 40 to 50 microamperes) and therefore requires amplification, the cells can readily withstand the space environment and possess uniform outputs having a linear relationship to the incidence of light. The effects of ultraviolet and Van Allen radiation are minimized since the cells are substantially shielded by the fused quartz block.

The command reticle and its associated photocell is used as a trigger for the aspect indicator. The unit permits the sun angle to be read out only when the sun is contained in a plane at a right angle to the slit over the Gray-coded reticle. (This occurs once per satellite revolution.) The device consists of a reticle slitted on the top and bottom surfaces and mounted over a photocell. The slits are oriented 90 degrees to the slit in the Gray-coded reticle.

c. Electronics Package

A block diagram of the solar-aspect indicator is shown in Figure II-7. The output of each of the solar cells in the aspect indicator provides a maximum signal of approximately 50 microamperes into a low-impedance load. The signals from each bit are amplified by one-stage, chopper-stabilized d-c amplifiers and are applied to the first seven bits of an eight-bit shift register. Chopper transistors in the bit amplifiers are used to a-c couple the solar cell output to the amplifier, thereby preventing leakage in the solar cell due to the reversed-biased diode characteristics during high ambient-temperature conditions.

The command-eye produces a constant output at various gamma angles, and is connected to a d-c chopper which is driven by a 4-kc multivibrator. The chopped command-eye signal is amplified and coupled to a capacitive integrator through an emitter follower, which establishes the charge path for the integration of the chopped command-eye signal. The second output of the multivibrator is used to saturate a switch in the discharge path of the integrator circuit. The charge level of the integrator is dc-coupled through an emitter follower to the input of the Schmitt trigger.


ELECTRONICS
SUN – SENSING ELEMENT (GRAY – CODE DEPOSIT) 7 BIT FORMAT
BIT 1 (LSB)
BIT 2
BIT 3
BIT 4
BIT 5
BIT 6
BIT 7 (MSB)
BUFFER AND GATE'S FOR REGISTER LOAD
BIT 1
BIT 2
BIT 3
BIT 4
BIT 5
BIT 6
BIT 7
LSB
MSB
END WORD ALWAYS "1"
SET PULSE
STROBE
RESET AT PWR ON
AMPL
EYE COMMAND
SHIELD GRD
–12.5 VDC
DETECTOR
4 KC MULTI
SWITCH
SCHMITT TRIGGER
"ONE SHOT"
AMPL
GATE
INHIBIT

II-17/II-18

①

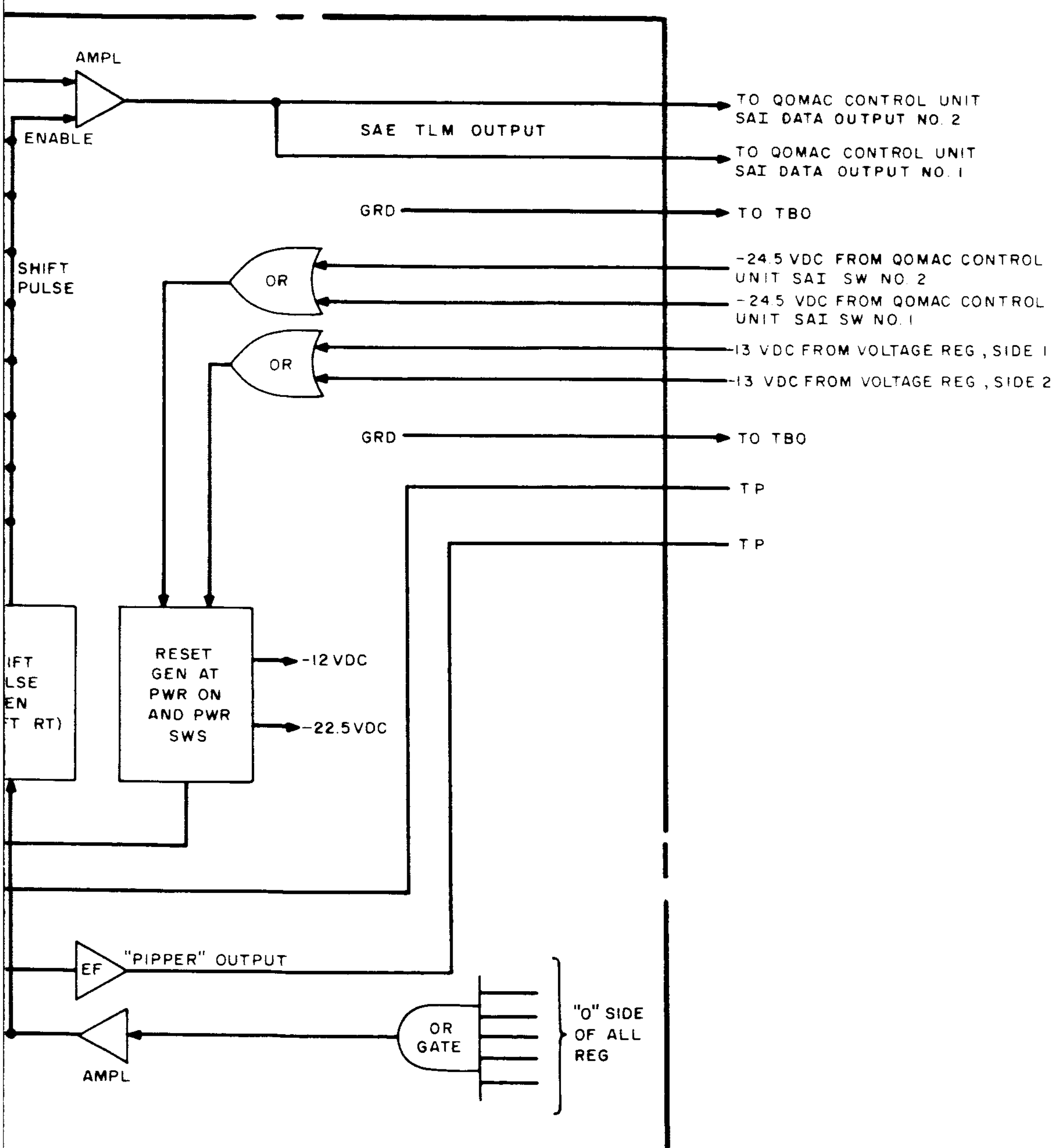


Figure II-7. Block Diagram of
Solar-Aspect Indicator

2

The combination of the relatively short RC time constant of the charge path, and the long RC time constant of the discharge path (switched in at a 4-kc rate) combine to produce the step pattern of the output waveform. This output permits the Schmitt circuit to trigger at a discrete level, rather than on an undefined point on the command eye wavefront.

When a command pulse is received from the command eye amplifier, the negative going slope of the Schmitt-trigger output starts a 500-microsecond monostable multivibrator. The output of the multivibrator is inverted and applied to the chopper transistors in the bit amplifiers. The inverter output is also differentiated in the bit amplifiers and the negative half of the pulse is "AND'ed" with the collector of the bit amplifier in a manner that causes a "1" to be set into the register whenever the collector is above a predetermined level.

The compensated output of the bit amplifiers approaches a truncated triangular wave. Ideally, the transition from a "0" to a "1" should occur at the point where the light coming through the slit falls half on the opaque, and half on the transparent area of the Gray-coded reticle. The flip-flops in the register are designed to trigger at the midpoint, setting at "1" for a signal above the midpoint and at "0" for below. A serial output is obtained by connecting the threshold flip flops together in a shift register configuration. A shift oscillator is used to generate the shift pulses. The "0" side of all shift register flip flops are applied to an OR circuit, whose output is used to enable the shift oscillator. A "1" is set into the "end-of-word" bit when the register is reset; therefore the shift oscillator will be enabled until this bit has been shifted out, after which the oscillator is disabled, ending the readout sequence.

On initial power turn-on, the integrator provides an output which is connected through diodes to the "0" side of the shift-register flip-flops and holds them off for 50 to 100 milliseconds. At the end of this period, a transistor switch is closed, disabling the "0" diodes until power is removed. The system will operate off a redundant set of d-c power supplies (-13 volts, and -24.5 volts) and can be switched to either set in the event of supply failure. (The -24.5 volt supply is turned on by a ground command; and the -13 volt supply is turned on by the solar-aspect indicator, itself.)

3. Attitude Horizon Scanner

The attitude horizon scanner comprises two infrared horizon sensors and associated electronics. The two horizon sensors are installed such that sensor No. 1 looks upward from the baseplate at an angle of 40 degrees and sensor No. 2 looks downward at an angle of 40 degrees. Thus, as the spin-stabilized TIROS X satellite orbits the earth, the optical axis of at least one of the two sensors will intercept the earth during part of each spin period. During the period that a sensor scans (or views) the earth, the infrared input level is significantly higher than during the sky-scan periods and an output pulse is generated. This pulse is differentiated and transmitted to ground where it is recorded on a paper chart recorder (the major positive-going pulses indicating sky-to-earth transitions and the major negative-going pulses indicating earth-to-sky transitions). From this recording, the ratio of the earth-scan period (T_E) for each sensor and spin period (T_{SPIN}) is determined and, as detailed in the TIROS X Attitude Handbook, used in conjunction with the gamma-angle data from the solar-aspect indicator to determine the spin-axis attitude in terms of roll angle (φ_{max}) and orbit phasing angle (τ). A block diagram of the attitude horizon scanner is shown in Figure II-8.

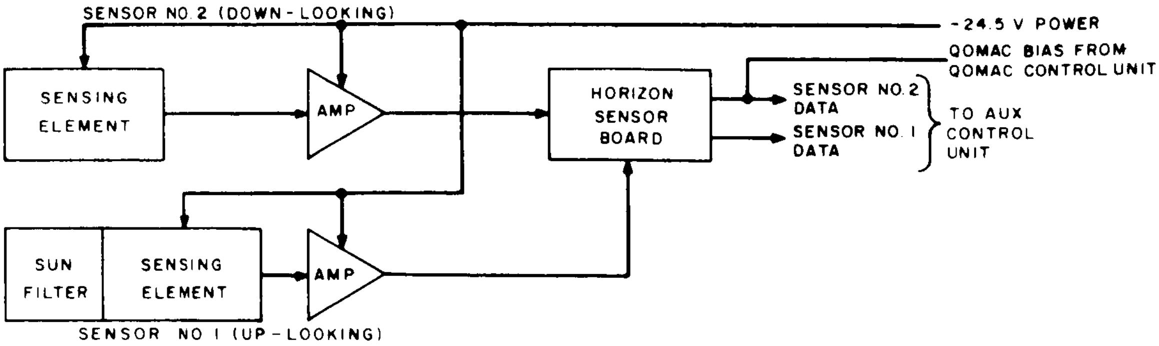


Figure II-8. Attitude Horizon Scanner, Block Diagram

Each sensor in the attitude horizon scanner is identical to the TIROS VII horizon sensor, with the exception that the size of the included angle between the two sensors (80 degrees) made it necessary to add a sun filter to the "up-looking" sensor, sensor No. 1.

4. Testing

a. Solar-Aspect Indicator

The solar-aspect indicator was purchased from a manufacturer who flight-qualified the unit prior to delivery to AED.

b. Attitude Horizon Scanner

(1) Qualification Testing

Horizon sensor No. 1, Serial No. 3-901-2, successfully completed vibration testing on December 7, 1963, and thermal-vacuum testing from December 13 through December 15, 1963.

Horizon sensor No. 2, Serial No. 3-901-1, successfully completed vibration testing on November 19, 1963, and thermal-vacuum testing from November 11 through November 14, 1963.

(2) Spectral Response

The optical responses of the two horizon sensors are identical in design, except that an 8-micron long-wavelength pass filter is placed in front of sensor No. 1 to reduce sun interference. Figure II-9 shows the relative spectral response of the two horizon sensors.

In Figure II-10, the relative output (as compared to peak output) of horizon sensor No. 2 is plotted as a function of the angular displacement of the target from the mechanical axis. The data was obtained by moving a small (subtended angle of less than 0.1 degree) radiation source across this field-of-view. Horizon sensor No. 1 provides a similar response, except that elevation between mechanical and optical axes for sensor No. 1 is only 1 minute of arc. The fields-of-view of the sensors, measured in two orthogonal directions for a 50-percent response, ranged between 67 and 81 angular minutes.

F. POWER-SUPPLY SUBSYSTEM

1. Introduction

The power-supply subsystem for the TIROS X satellite consists of an array of 9120 P-on-N silicon solar cells, an energy storage system containing 63 nickel-cadmium storage batteries, voltage regulation circuits, protection circuits, and telemetry sensing networks. A special group of 60 cells, mounted on the top of the satellite hat, are included in the subsystem so that indications of the over-all condition of the solar cells can be telemetered to the CDA stations.

The solar cells are mounted on the top and sides of the satellite structure. During orbital day, when the solar cells are illuminated by the sun, the output of the array is used as a primary power source for the satellite's electrical system; any excess power (power that is not needed by the satellite electronics) is used to charge the storage batteries. In cases where peak power requirements exceed the power output of the solar cells, the batteries automatically supply the power difference. During orbital night, the storage batteries, which have a total capacity of approximately 309 watt-hours, supply all the power required by the satellite. Precautions have been taken to prevent circulating currents in the power source interconnections and to preclude the total loss of power in the event that a short circuit occurs in one of the battery cells.

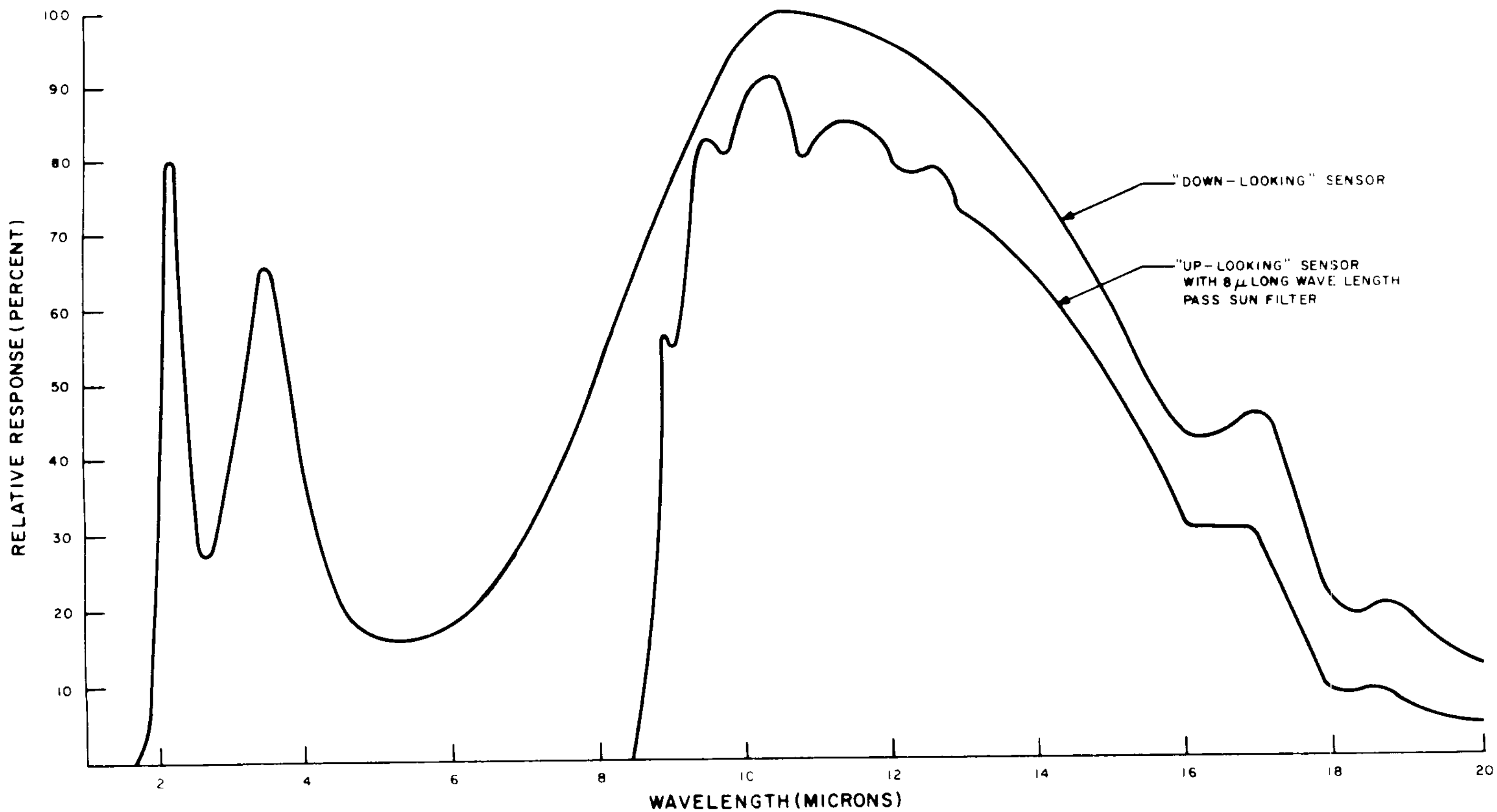


Figure II-9. Spectral Response of Horizon Sensors

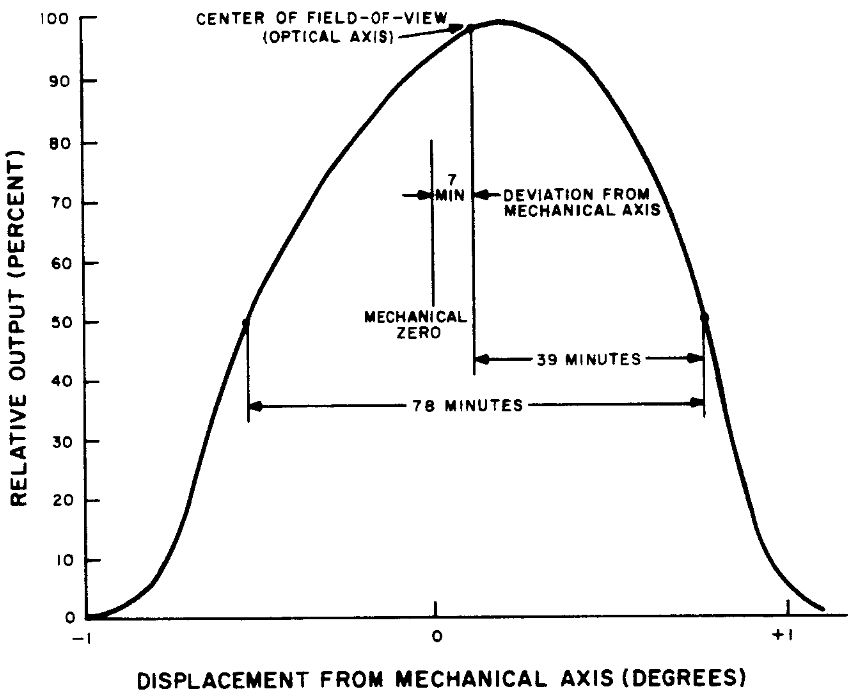


Figure II-10. Relative Output of Horizon Sensor No. 2 over the Sensor's Field-of-View

The storage batteries are electrically connected in three independent groups, each of which is connected to the solar-cell array through its own charge current controller. The charge controllers limit the maximum current into each battery string to a safe charge level. Excess array power is diverted through a bypass regulator to the main battery output bus. The bypass regulator also limits the maximum voltage excursion of the battery output bus so that input limits of the series-row voltage (-24.5 volts dc) are not exceeded. During orbital night, when the solar cells are passive, silicon diodes in each series row of solar cells prevent the storage batteries from discharging into the solar cells.

A similar function is performed by the diodes which are included in each series row of solar cells located on the lateral surface of the satellite. Because of the satellite rotation, each row is alternately illuminated and then darkened. The diodes prevent the darkened solar-cell rows from loading the illuminated rows. The storage batteries, in addition to providing power for the satellite subsystems, provide a relatively constant voltage across the solar cells, isolating them from variations in the electrical load.

Minus 24.5 volts and minus 13 volts are provided to supply the regulated d-c voltages required by the spacecraft. Minus 28 volts (unregulated) is supplied by the main battery output bus or the array bus via the bypass regulator, depending on which is more negative. Its limits are determined by the minimum battery bus power (minus 25.2 volts) and the maximum clamping level of the bypass regulator, minus 33 volts.

A block diagram of the power-supply subsystem is shown in Figure II-11.

2. Equipment Description

a. Solar-Cell Array

Electrical power for the satellite subsystems is generated by an array of 9120 P-on-N silicon solar cells mounted on the top and sides of the satellite structure. Each cell is 1 x 2 centimeters in area, and has a transparent, 0.006-inch, micra platelet bonded to it to improve thermal emissivity. An anti-reflective coating which permits maximum light transmission in the 600- to 800-millimicron range (where the cells are most responsive) is vacuum-deposited on the upper (or outer) surface of each cell. A multilayer, sharp cutoff, blue reflective coating is deposited on the lower or inner surface. This coating reflects all wave-lengths lower than 400 millimicrons and transmits 90 to 95 percent of all longer wave-lengths up to and beyond the 1100-millimicron upper response limit of the cells. The platelet is bonded to each cell with a transparent epoxy adhesive.

The cells are assembled in shingle form, each shingle consisting of 5 cells connected in series. Each shingle has a conversion efficiency of 9.0 percent at 1.95 volts and at a temperature of 27 ± 2°C. A group of sixteen shingles (i.e., 80 series-connected solar cells) are bonded to a flat epoxy-fiberglass board 3.400 inches wide by 7.572 inches long by 0.036 inch thick. The resultant moduleboard assembly weighs approximately 80 grams.

The modules are bonded directly to the top surface of the satellite and to the 18 side panels in parallel groups of four, with the zero (ground), one-quarter, one-half, three-quarters, and full voltage points connected in series. This method of interconnection tends to minimize effects of a shorted or open solar cell.

The physical characteristics of the fully assembled solar array are as follows:

Area	No. of Cells	Active Area
Top	3520	6.8 square feet
Sides	5600	10.9 square feet

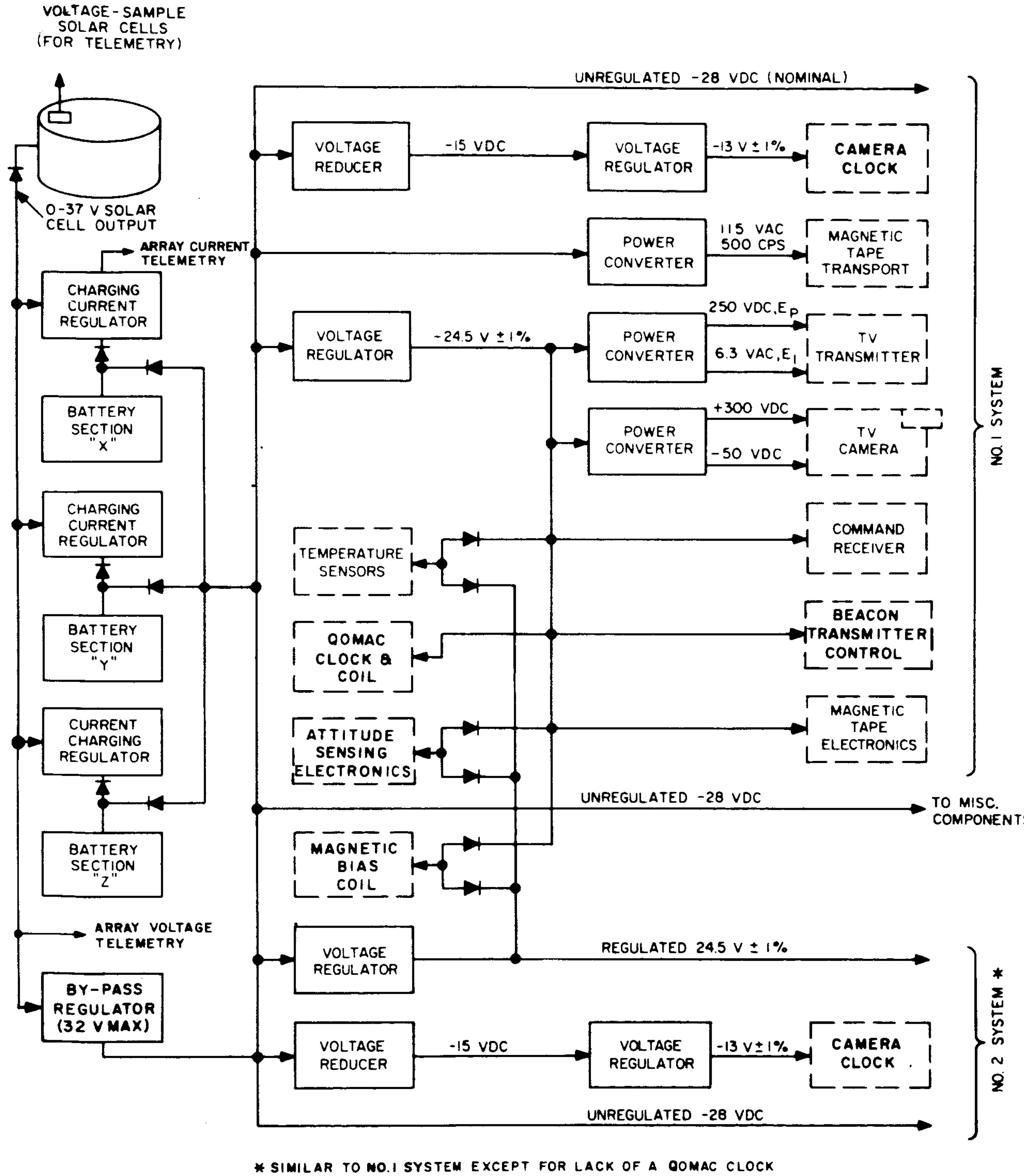


Figure II-11. Power Supply Subsystem, Block Diagram

b. **Solar-Cell Telemetry Patch**

A group of solar cells mounted on the top of the hat is electrically isolated from the power supply and provided with a fixed resistive load that provides a d-c signal to the telemetry subsystem. When the top surface of the hat is illuminated, this group of solar cells provides an uninterrupted output voltage whose level is a function of solar-cell temperature and sun angle.

In orbit, the telemetered voltage output, V_T, of the solar-cell patch and the calibration voltage, V_C, for the specific solar-cell temperature and sun angle are periodically examined. This comparison, expressed by the ratio V_T/V_C, provides a measure of solar-cell performance and thereby, a means of checking solar-cell degradation due to radiation damage and micrometeorite bombardment.

c. **Storage Cells and Battery Pack**

The characteristics of the individual storage cells are as follows:

(1) Type of cell: Sintered plate glass-to-metal seal; hermetically sealed nickel-cadmium storage cell, F-size, with paper separators.

(2) Ampere-Hour Capacity:

(a) 3.9 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 25°C.

(b) 3.1 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 0°C.

(c) 2.7 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 40°C.

(3) Watt-hour capacity: 4.9 watt-hours at 25°C for discharge conditions listed in (a) above.

(4) Watt-hours per pound: 8.91 watt-hours at 25°C for discharge conditions listed in (a) above.

(5) Maximum recharge rate: 0.600 ampere between 0°C and 40°C.

(6) Watt-hour efficiency: 70 percent.

(7) Physical Characteristics of Cell:

(a) Weight: 0.53 to 0.55 pound.

(b) Diameter: 1.3 inches.

(c) Length: 3.5 inches.

(d) Volume: 4.7 cubic inches.

The characteristics of the overall battery pack are as follows:

(1) Number of F-size cells: 63

(2) Number of parallel rows: 3

(3) Number of series cells per row: 21

(4) Total watt-hour capacity: 309 at discharge conditions defined in paragraph (2) (a) above.

(5) Total ampere-hour capacity: 1.7 at discharge conditions defined in paragraph (2) (a) above.

(6) Total weight of assembled battery including all packaging: 39 to 41 pounds.

The following battery operational parameters are telemetered to the ground:

(1) The terminal voltage of each of the three rows of storage cells.

(2) The voltage supplied to the input of one of the voltage regulators by the three rows of storage cells, each supplying power to the regulator through isolating diodes.

(3) The temperature of the battery pack.

(4) The output voltage of two minus 24.5 volt d-c voltage regulators and two minus 13.0 volt d-c regulators.

Each of these parameters is employed to determine whether the storage batteries and associated equipment are performing within the limits of the electrical design specifications.

d. Current Telemetry Board

The array-current telemetry circuit permits an evaluation of the electrical performance of the solar-cell array under orbital conditions. The magnitude of current flowing in the solar-cell array is measured, and the measured value is converted to a signal voltage suitable for transmission to the ground by means of the telemetry system. The maximum output signal of the array-current telemetry circuit, with 2.5 amperes flowing through the array current sensing resistors, is set at -5 volts dc. The circuit is normally in the OFF state, and is commanded ON by a -24.5 volt d-c signal.

The array-voltage telemetry circuit is also included in the current telemetry board. This circuit is a precision, resistive divider network connected directly across the output of the solar array, and provides telemetry signals directly proportional to the array output.

e. Power Supply Protection Unit

(1) General

The TIROS X power supply protection unit (PSPU) is basically similar to that used on TIROS VII. PSPU comprises charging regulators, voltage dividers, a bypass regulator, pre-regulators, and an external mounting block for the 12-ampere fuses.

(2) Charging Regulators

The PSPU includes a charging regulator for each battery string. The charging regulators limit the charge rate of each bank to 440 ± 40 milliamperes whenever the battery voltage is from 1 to 14 volts below the solar-cell array output voltage.

(3) Fuse-Wire Mountings

Five 12-ampere fuse wires are connected between the unregulated bus and the major subsystem elements, and two 1.2-ampere fuse wires are placed between the unregulated bus and the two pre-regulators.

For the TIROS IX program, a new fiberglass fuse housing which completely encloses the individual fuses was designed for mounting on a bracket on top of the PSPU. This type of housing was used for mounting the 12-ampere fuses on TIROS X.

(4) Isolating Diodes

Six high-current transistors, connected as diodes, are used to isolate the charging and load-delivering paths of each 21-cell battery string. These diodes prevent (1) battery discharge into the satellite loads during the daytime (except during periods of high-current requirements), and (2) battery discharge into the quiescent solar-cell array during the orbital night. The units also prevent circulating currents between the parallel-connected battery strings and, in the event of failure in one string, preclude total loss of power by isolating the defective battery string.

(5) Bypass Regulator

Under normal conditions, the solar-cell array can deliver 2.2 amperes to the satellite. Of this total, whatever is required for direct use by the satellite loads will be supplied, and the remainder, to a maximum of 1.2 amperes will be supplied to the batteries by means of the charging regulators.

A simple two-transistor bypass regulator prevents the voltage level of the unregulated power bus from exceeding 32 volts. When the input drops below 32 volts, the bypass regulator functions as a diode with a forward drop of approximately 1.0 volt and passes the current to the load without regulation.

(6) Pre-Regulators

Two pre-regulators (voltage reducers), connected across the unregulated bus, clamp their outputs to -16 ± 1 volts dc, providing pre-regulated inputs to the two -13 volt dc series voltage regulators.

f. Voltage Regulators

The TIROS X power-supply subsystem also includes two -13 volt regulators and two -24.5 volt regulators. The regulators used on TIROS X are the same series-type units as were used on TIROS VII, with a -13 volt unit and a -24.5 volt unit housed together in an open, 0.25-inch aluminum chassis, and temperature-compensated, single-ended amplifiers used for sensing.

3. Testing

a. Battery Pack

The flight-model battery pack, Serial No. 3-721-06, was successfully subjected to vibration testing on August 31, 1964 and was electrically tested on September 1, 1964. It was delivered for integration on September 4, 1964. The battery was later returned to engineering for the installation of improved terminal feed-through pins on each of the nine modules. After completion of the work, the battery was requalified on January 11, 1965, and again delivered ready for integration with the spacecraft. In May, 1965, the flight-model battery pack was subjected to a final checkout prior to installation on spacecraft OT-1. It was first subjected to a conditioning charge and discharge from May 4 to 6, 1965, and then to a capacity test at 25°C on May 7, 1965. The battery was then discharged completely and subjected to a second capacity test on May 11, 1965. The test voltages for each of the three rows after charge and discharge are listed in Table II-4, along with the acceptable maximum and minimum values. On May 12, 1965, as a result of these tests, the battery was again declared to be flight-qualified.

b. Voltage Regulators

Voltage Regulators, Systems Nos. 1 and 2, Serial Nos. 3-704-01 and 3-704-02, respectively, were subjected to vibration testing on August 14, 1963, and thermal-vacuum testing from August 18, 1963 to August 21, in each case with satisfactory results.

c. PSPU

The PSPU, Serial No. 3-703-02, was successfully subjected to vibration testing on September 30, 1963. Thermal-vacuum testing on the unit was completed on October 3, 1963, with satisfactory results.

d. Current Telemetry Board

The current telemetry board was successfully subjected to vibration testing on April 28, 1964. On October 3, 1963, thermal-vacuum tests on this unit were completed with satisfactory results.

e. Solar-Cell Array

The TIROS X solar-cell array was tested under sunlight at AED on April 29, 1965. Based on these measurements, a power prediction was developed (see Appendix A), which further demonstrated that the array was capable of sustaining the mission.

G. ANTENNA SUBSYSTEM

1. General

The TIROS X antenna subsystem provides for the reception of command signals from the CDA ground stations and for the simultaneous radiation of energy from three of the four separate satellite transmitters (two beacon transmitters and either one of

TABLE II-4. POST-QUALIFICATION TEST RESULTS ON BATTERY PACK
SERIAL NO. 3-701-06

Test Date	Conditions	Test Results (Volts)			Acceptable Levels (Volts)	
		Row X	Row Y	Row Z	Maximum	Minimum
May 4 to 6, 1965	After conditioning charge at 180 ma for 40 hours	29.65	29.74	29.76	31.1	--
May 6, 1965	After discharge at 1.3 amp for 3 hours	25.58	25.69	25.73	--	25.2
May 7, 1965	After charge at 1 amp for 6 hours	30.57	30.65	30.80	31.1	--
May 7, 1965	After discharge at 1.3 amp for 3 hours	25.83	25.86	25.90	--	25.2
May 11, 1965	After charge at 1 amp for 6 hours	30.40	30.57	30.37	31.1	--
May 11, 1965	After discharge at 1.3 amp for 3 hours	25.84	25.87	25.84	--	25.2

the two TV transmitters). The subsystem also provides for coupling and matching the receivers and transmitters to the antennas and isolating the three active transmitters and the two command receivers.

The subsystem consists of (1) a single dipole receiving antenna for the command receivers, (2) two crossed-dipole transmitting antennas, (3) the associated RF matching and coupling network for the transmitters, and (4) a notch filter* for further isolating the receivers from the beacon transmitters.

2. Functional Description

a. Receiving Antenna

The receiving antenna is a separate 1/4-wavelength dipole at the command frequency. It is positioned in the neutral plane of the crossed transmitting dipoles. This positioning causes an approximate 45-db attenuation to exist between the transmitting and receiving antenna terminals. The receiving antenna is vertically mounted on top of the satellite at the spin axis and is coupled to the two receivers through a 1/2-wavelength transmission line, a four-way cross adapter, and 3/8-wavelength transmission lines.

b. Transmitting Antenna

The two crossed-dipole transmitting antennas form a composite antenna operating at both the beacon and the TV frequencies. It consists of four elements mounted to the satellite baseplate. Each element is a coaxial structure with rods extending through canted sleeves. The total element length is approximately 32.5 inches, which is equivalent to 0.36 wavelength at the 136-Mc beacon frequency. At the TV frequency (235.0 Mc), the rod extensions are isolated from the coaxial sleeves by a short-circuited stub within each coaxial structure. Thus, at that frequency, the element length is that of the sleeves, which extend 12 inches from the drive end of the dipole. This length is equivalent to 0.25 wavelength at 235 Mc. The dipoles are fed in quadrature to achieve circular polarization.

c. Matching and Coupling Networks

The TIROS X matching and coupling network is the same as that used in TIROS II through VII and in TIROS IX. The network couples the three transmitters to the radiating elements, provides an impedance match for this coupling, minimizes interaction and feedback between the transmitters, and effects circular polarization by exciting the antenna elements in phase quadrature.

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*The notch filter was added immediately prior to launch to eliminate an interference condition observed during testing at the launch site.

Figure II-12 shows the schematic diagram of the matching and coupling network. The network is composed of two similar sections, each consisting of a diplexer and two baluns. The RF transmitters (shown as a-c generators in Figure II-12) are connected symmetrically across the diplexer inputs, while the matched loads are driven from the output. The currents in the two loads are phase-displaced by 180 degrees. Theoretically, the two transmitters in each section are completely isolated, resulting in a perfect load match.

The two baluns in each section are essentially delay lines that selectively phase the RF currents in the dipoles. To achieve circular polarization, the two line lengths between the diplexer outputs and their respective baluns differ in electrical length by 90 degrees.

d. Notch Filter

By NASA directive, a notch filter was added to TIROS X at the launch site to reduce RF-interference effects experienced at the gantry. The notch filter is connected to one input of the coaxial adapter to prevent operation of the beacon transmitters from affecting the command receiver. The filter is identical to those used on TIROS VII and TIROS IX, and is in the form of a shunt stub-line placed across the antenna line. The stub is open-circuited and one-quarter wavelength at the frequency to be rejected, thereby providing a high conductance at the rejected frequency, but an extremely low conductance at the command-receiver frequency. The achievement of a high conductance at the rejection frequency in the 137-Mc band and a negligible conductance at the relatively close command frequency required a line having both a high Q and high characteristic impedance.

The RF filter used to satisfy these requirements utilizes a coaxial transmission line having a helical inner conductor, the diameter of which is approximately half the inner diameter of the outer conductor.

The TIROS X notch filter was tuned for maximum isolation at 136.575 Mc, i.e., the median of the beacon frequencies.

3. Testing

a. Transmitting Antenna

The transmitting antenna (Serial No. 3-408-01) was combined with the matching and coupling network on September 3, 1963. Final tuning of the antenna, which constitutes its electrical acceptance test, was performed on October 8, 1963.

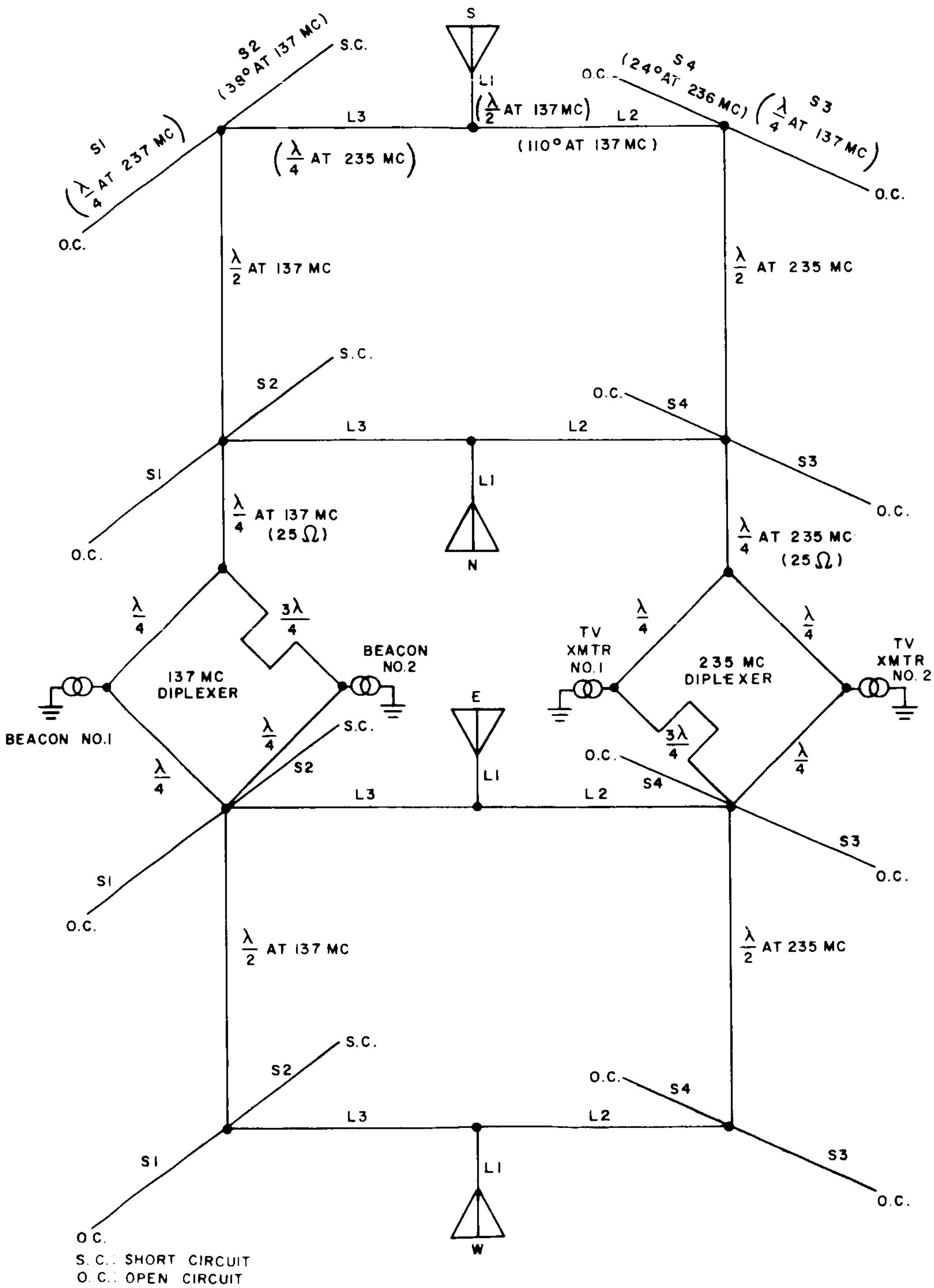

Figure II-12. Antenna-Coupling Network, Schematic Diagram

b. Matching and Coupling Network

Both the 136-Mc section of the matching and coupling network, Serial No. 3-409-01, and the 235-Mc section passed electrical acceptance tests on July 1, 1963, and the sections were accepted for final fabrication on the same date. Electrical acceptance tests of the final configuration of both sections were successfully passed on July 25, 1963.

c. Notch Filter

The notch filter successfully completed acceptance tests on November 16, 1964. During these tests, a final measurement of the filter's response characteristics was made. The results obtained are plotted in Figure II-13.

H. DYNAMICS-CONTROL SUBSYSTEM

1. General

The TIROS X dynamics-control subsystem comprises the following units:

- precession dampers,
- despin device,
- spin-up rockets,
- QOMAC (Quarter Orbit Magnetic Attitude Control) coil, and
- MBC coil.

On TIROS X, the QOMAC coil was added to the standard TIROS dynamics-control subsystem to permit spin-axis attitude control. In addition, the voltage rating and current capability of the MBC coil were increased for all positions of the MBC switch, and two high-torque switch positions were provided to permit a back-up capability for spin-axis control in the event of a failure in the QOMAC coil after the turn-around maneuver is accomplished. (The MBC coil still provides, however, for cancellation of the satellite's residual dipole moment.)

2. Precession-Damping Devices

The initial wobble of the satellite, caused by precession or nutation after release from the third-stage rocket, is damped-out rapidly by two tuned-energy-absorption-masses (TEAM) that oppose the forces that tend to oscillate the satellite body. The precession-damping devices are shown in Figure II-14. Two similar mechanisms are installed

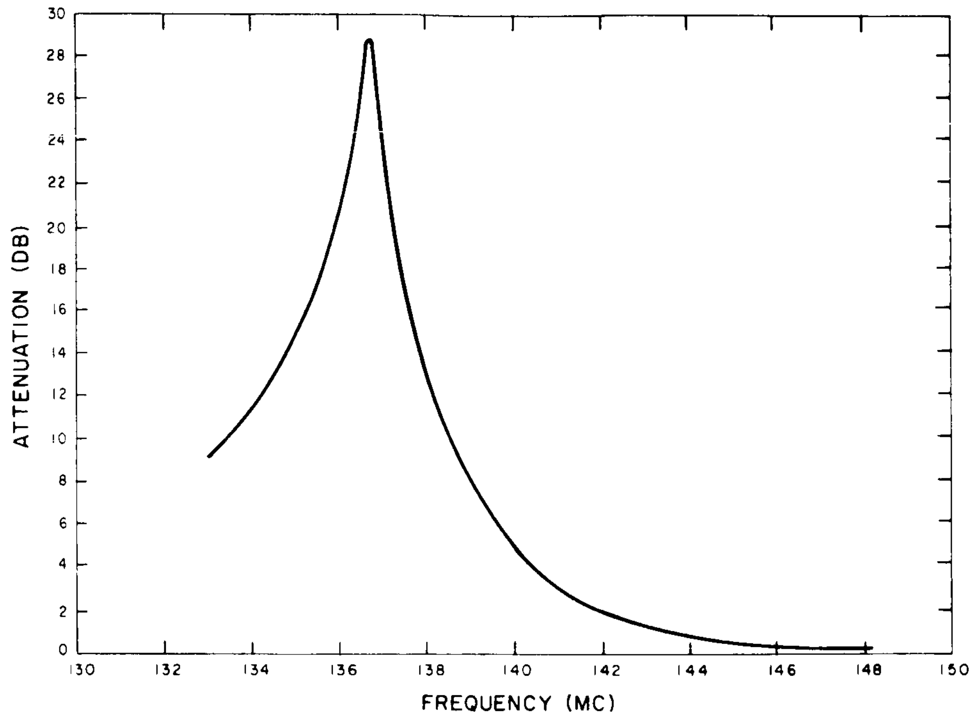

Figure II-13. Response Characteristics of TIROS X Notch Filter

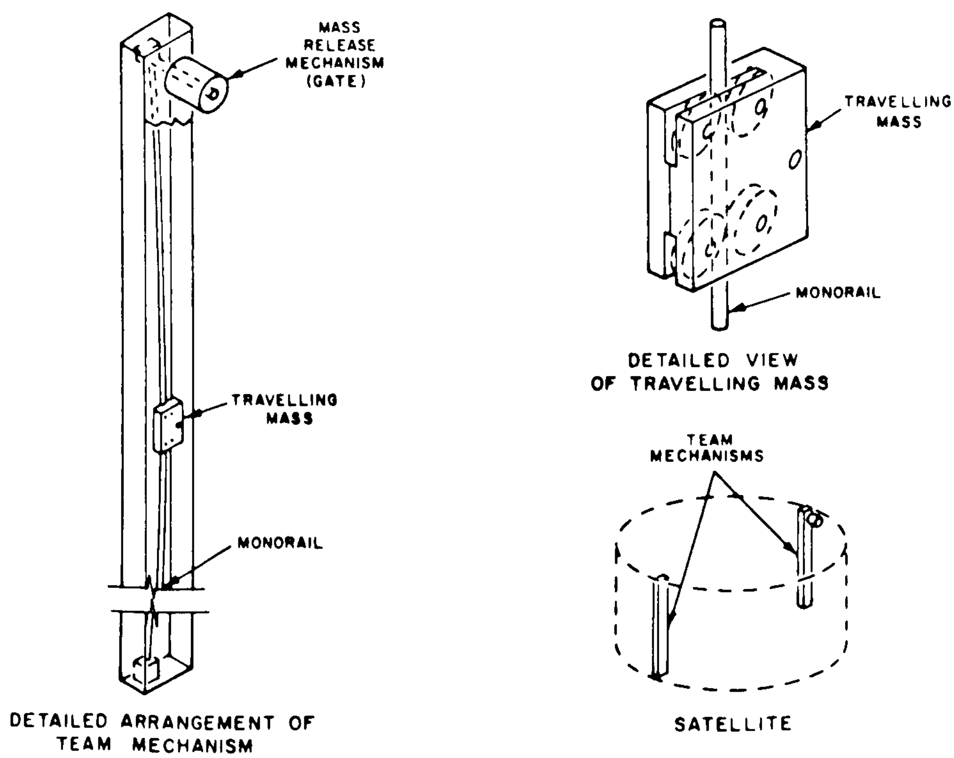

Figure II-14. TEAM Precession-Damping Device vertically along the inside wall of the satellite hat, 180 degrees apart. Each mechanism weighs a little under one pound; each mass, approximately 3 ounces. During launch, each travelling mass is restrained by a mechanical gate. Upon separation of the satellite from the third stage of the launch vehicle, the gates are opened by automatic firing of squibs, permitting the masses to roll freely along the rods. Manual firing of squibs from the ground station, utilizing switch contacts on the rocket-firing switch in the satellite, is also provided as a back-up control measure. The device is tuned to the natural precession frequency of the satellite, i.e., 2.74 cps, and rapidly absorbs the energy causing the wobble, dissipating this energy in the form of heat. Upon stabilization, the masses rest at or near the center of the rods.

3. Despin Mechanism

When the satellite separates from the third stage of the launch vehicle, its spin rate is approximately 126 rpm. The despin mechanism is used to reduce this spin rate to an operational rate in approximately 0.5 second. The mechanism consists of a pair of 1-pound weights attached to light steel cables, wrapped once around the satellite, about the periphery of the baseplate, and attached to the satellite structure by means of hook-and-eye devices. Approximately 9 minutes after separation of the satellite from the launch vehicle, the masses are released by the automatic firing of a pair of squibs. (A capability for manually firing these squibs in response to ground command is also provided.) The released masses move radially from the satellite, unwrapping the cables and resulting in a significant increase in the satellite's moment of inertia. This, in turn, reduces the spin rate to approximately 10 rpm. When the cables unwrap completely, they disengage from the hook-and-eye devices and the weights and cables are cast off, carrying a portion of the satellite's kinetic energy into space. Thus, the satellite's moment of inertia is returned to essentially its initial value, but the lower spin rate is maintained.

4. Spin-Up Rockets

As the satellite spins, the ferrous materials used in its construction are acted upon by the earth's magnetic field. This action produces a drag that slowly reduces the spin rate. To restore the spin rate to its optimum range (8 to 12 rpm) five pairs of solid-propellant rockets (each delivering an impulse of approximately 1.4 pound-seconds) are mounted around the periphery of the baseplate. The firing of a pair of spin-up rockets is programmed from a ground station when measurements indicate the need for an increase in the spin rate. When fired, each pair of spin-up rockets increases the spin rate between 3 and 3.5 rpm.

The firing of the rockets is controlled by means of a seven-position stepping switch. The first position provides for back-up firing of squibs that control activation of the precession dampers; the second for back-up firing of the despin squibs; and the remainder for firing the spin-up rockets in pairs.

5. QOMAC Coil

The center-tapped QOMAC coil is used in conjunction with the QOMAC clock and QOMAC coil unit* to provide control over the attitude of the satellite's spin axis. This control is effected by means of the torque resulting from the interaction of the earth's magnetic field and the magnetic field which results from current through the QOMAC coil. The operation of the QOMAC coil is utilized both in accomplishing the initial turn-around maneuver and in ensuring that the minimum nadir angle occurs at the desired point along the satellite subtrack.

The direction of current through the coil is reversed at quarter-orbit intervals such that for each QOMAC cycle (see Figure II-15) a positive dipole is induced during the the first quarter orbit and a negative dipole is induced during the second quarter orbit. Sixteen QOMAC cycles are programmed for a given QOMAC sequence; a 16-cycle (8-orbit) sequence will change the spin axis attitude by approximately 80 degrees. If less than 16 cycles are desired, a CDA station must shut the QOMAC system off by a realtime command after the desired number of cycles have been completed. Unlike TIROS IX, which could be commanded for either high- or low-torque QOMAC, TIROS X is equipped to provide high-torque operation.

When a QOMAC sequence is desired, two separate command operations are required**: First, camera clock No. 1*** must be set to the delay time required for the start of the QOMAC sequence; second, a QOMAC ON command must be sent to the QOMAC control unit.**** When the camera clock alarms, the QOMAC control unit allows current through the coil in a direction such that a positive dipole moment results. Approximately 25 minutes later, the QOMAC clock sends a pulse to the control unit, reversing the current through the coil and resulting in a negative dipole moment. This reversing of current at 25-minute intervals continues until the sequence is ended by ground command or until 16 cycles are completed and the QOMAC clock turns off automatically.

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*Detailed functional descriptions of the QOMAC clock and QOMAC control unit are provided in Volume II of this report, the classified supplement.

**The methods used in determining the need for a QOMAC sequence and the type of sequence required are described in the TIROS X Attitude Handbook.

***The setting of camera clock No. 1 results in a Remote Picture Sequence. Depending upon the required QOMAC alarm time, the remote sequence might or might not result in meteorologically useful pictures.

****Unless it is already on.

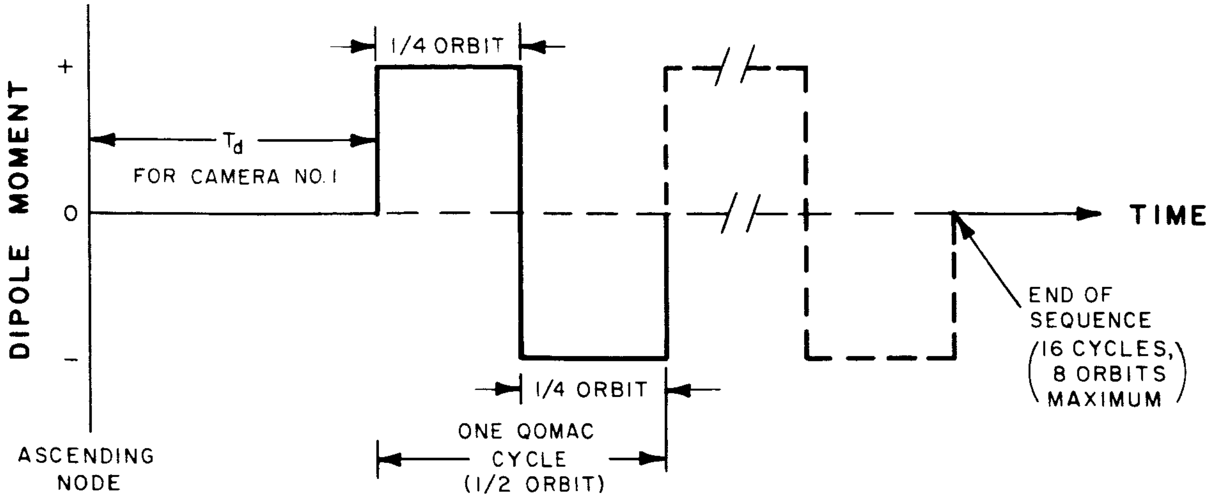


Figure II-15. Depiction of QOMAC Sequence

Normally, a QOMAC sequence will be terminated by ground command and will consist of much fewer than 16 cycles. In such cases, the sequence may be ended by sending either a QOMAC power "OFF" command to the satellite (as outlined in Volume II) or a playback command which resets both the QOMAC and the camera clock.

6. MBC Coil

The MBC (Magnetic Bias Control) device is composed of a 250-turn coil of wire wrapped around the periphery of the satellite and a solenoid-operated stepping switch that controls the direction and the amount of current in the coil. The MBC coil is used to cancel the satellite's small inherent magnetic dipole due to the ferromagnetic materials used in the satellite's construction and to the electromagnetism produced by the satellite's operating currents. After a number of days in orbit the satellite begins to show an attitude drift due to interaction of the residual dipole with the earth's magnetic field. This drift is analyzed and the residual dipole calculated; the MBC switch is then commanded by the ground station to step to a position to cancel the residual dipole. After cancellation of the dipole, the MBC coil is used to generate a constant dipole to cause the spacecraft attitude to drift, in order to compensate for the precession rate of approximately 1 degree per day that occurs with a sun-synchronous orbit. (This reduces the amount of QOMAC programming required.)

By use of this device, any one of eleven different dipole moments (ranging from -10 to +10 ampere-turns-meter^2) can be programmed into the satellite by ground-initiated commands. As noted previously, two high-torque switch positions permit the MBC coil to be used as back-up for the QOMAC coil once the turn-around maneuver has been accomplished. These positions provide either a positive or negative magnetic dipole moment of approximately 9.53 ampere-turns-meter^2.

When the MBC switch is in its sixth position, it connects a zener diode across the coil of the ON latching relays for the beacon transmitters. Although this diode will have no effect on the relay if it is already energized, it will prevent the relay from being re-energized after a "beacon-kill" sequence. Thus, when the switch is set to position 6, the satellite's TV picture subsystem can be programmed for operation without causing the reactivation of the previously disabled beacons.

7. Testing

a. General

The components of the dynamics-control subsystem were individually subjected to vibration testing prior to integration on the spacecraft and subjected to thermal-vacuum testing with the environmental qualification tests of the integrated spacecraft.

b. Precession Dampers

The TIROS X precession dampers, Serial Nos. 3-503-01 and 3-503-12, underwent vibration testing on August 5, 1964, with satisfactory results. The units were then subjected to a break-away friction test, and both produced results well within specifications.

c. Despin Mechanisms

The despin mechanisms, Serial Nos. 3-507-3 and 3-507-4, were successfully subjected to vibration testing on October 24, 1963.

d. MBC Switch

The MBC switch, Serial No. 3-502-03, was successfully subjected to vibration testing on August 6, 1963.

e. QOMAC and MBC Coils

The QOMAC and MBC coils, which are wound on the same frame, were subjected to vibration testing on May 5, 1965, with satisfactory results.

I. SATELLITE STRUCTURE

1. General

The structure and configuration of TIROS X is basically similar to that of TIROS VII. The structure comprises (1) a baseplate assembly, upon which the operating components are mounted, and (2) a cover ("hat") assembly, which is used as a mounting surface for the solar-cell array.

2. Spacecraft Layout

Figure II-16 shows the component layout of TIROS X. As was the case in TIROS VII, TIROS X utilizes an axial camera assembly in which the cameras view the earth through ports in the baseplate. The distribution of components was dictated by the electrical system requirements and by the requirements of dynamic balance. As standard procedure, the batteries were located at the center of the baseplate, along the spin axis, and the two tape transports were located at diametrically opposite positions.

The beacon timers were removed from spacecraft OT-1 in April 1965, in response to a NASA directive. This directive also set the included angle between the two sensors comprising the attitude horizon scanner at 80 degrees. This change necessitated the use of new wedge brackets for the sensors, modification of the sensor viewing ports, re-routing of the wiring harness, and the addition of a special sun filter to the "up-looking" sensor.

3. Temperature Sensors

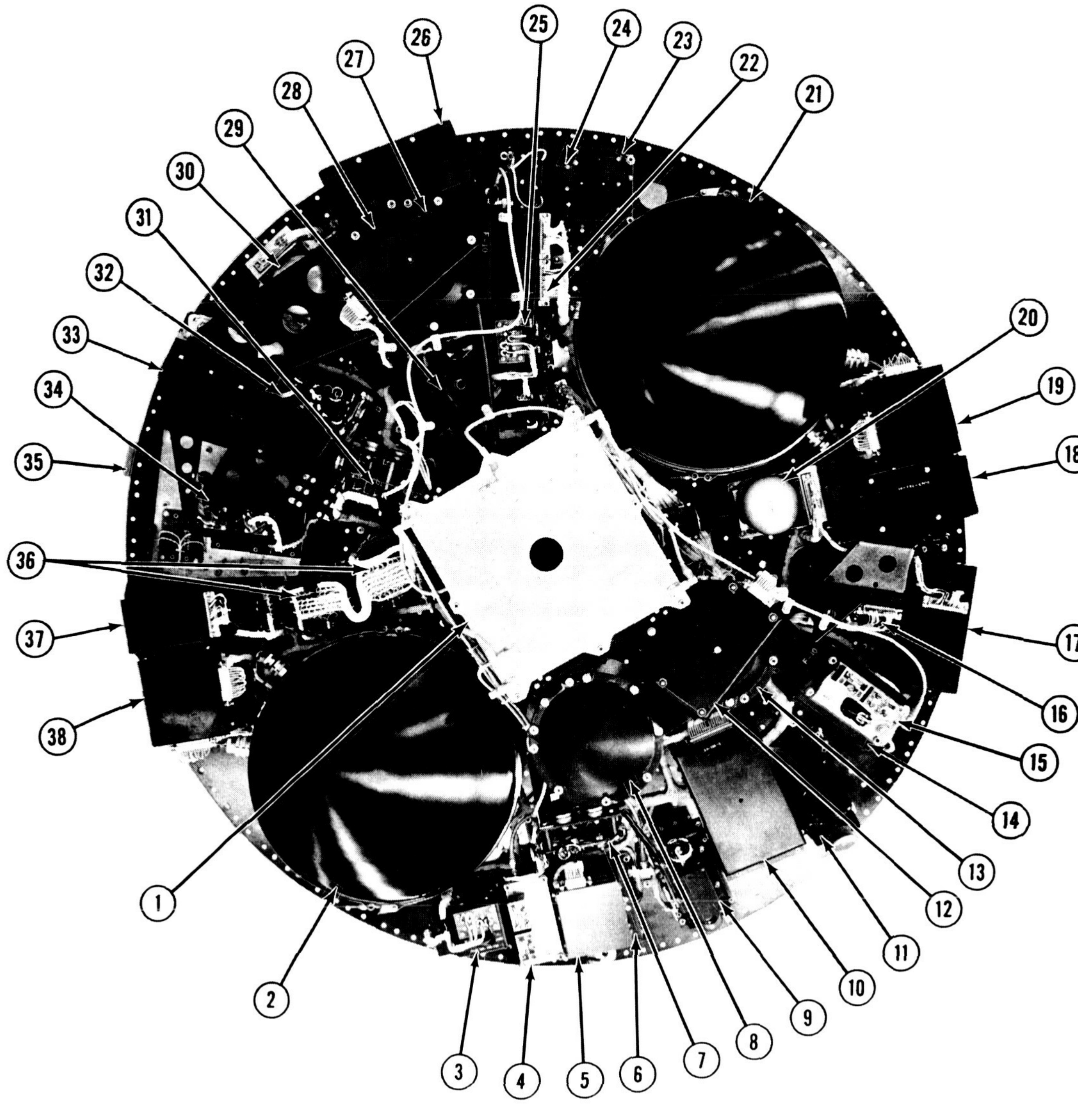
To permit an accurate means of monitoring the actual component operating temperature while the spacecraft is orbiting in the space environment, 13 temperature sensors were mounted on spacecraft OT-1. The locations of the sensors, and the operating range of each, are presented in Table II-5. TIROS X utilized temperature sensors of three ranges, +10 to +40°C, -20 to +10°C, and -30 to +100°C.

4. Thermal Analysis

The spacecraft OT-1 temperature profiles during the turn-around maneuver were determined based on initial gamma angles of 116 degrees and 128 degrees, with a value of 66.5 percent for the sun time. A temperature profile was also determined based on an initial gamma angle of 126 degrees, a sun time of 64.0 percent, and a "tip-off" error of about 2.5 degrees. These profiles are shown in Figures II-17, -18, and -19.

A fourth profile was determined, using the case of a 126-degree initial gamma angle, for the steady-state temperatures which would occur in the event that the turn-around maneuver would be prolonged. The results of this computation are shown in Figure II-20.

For the case shown in Figure II-19, the maximum average component temperature after 3 orbits would be 47°C. From Figure II-20, it can be seen that the steady-state condition encountered with a 126-degree initial gamma angle, in the absence of the turn-around maneuver, would yield an average component temperature of 53°C after 8 orbits.



LEGEND

1. RF Matching and Coupling Network and (Hidden) Battery Pack
2. TV Tape Transport No. 2
3. TV Transmitter Power Converter
4. Beacon Transmitter and SCO Assembly
5. Magnetic Bias Switch
6. TV Transmitter No. 1
7. Voltage Regulator No. 1
8. QOMAC Clock
9. Horizon Sensor No. 2
10. QOMAC Control Unit
11. Horizon Sensor No. 1
12. Camera Clock Board No. 1 and (Hidden) Command Address Unit
13. Camera Clock No. 1
14. Command Receivers
15. Beacon Transmitter and SCO Assembly
16. Command Control Unit No. 1
17. TV Tape Recorder No. 1, Power Converter
18. TV Camera No. 1, Electronics Unit
19. TV Tape Recorder No. 1, Signal Electronics
20. TV Camera No. 1
21. TV Tape Transport No. 1
22. Command Control Unit No. 2
23. TV Transmitter No. 2
24. TV Transmitter RF Filter
25. TV Transmitter Power Converter
26. TV Camera No. 2, Electronics Unit
27. Camera Clock Board No. 2
28. Camera Clock No. 2
29. Rocket Switch (Hidden)
30. TV Camera No. 2
31. Power Supply Protection Unit
32. Voltage Regulator No. 2
33. Auxiliary Control Unit
34. DSAI Electronics Unit
35. DSAI
36. Telemetry Switches (2)
37. TV Tape Recorder No. 2, Power Converter
38. TV Tape Recorder No. 2, Signal Electronics

Figure II-16. Component Layout of Spacecraft OT-1

TABLE II-5. LOCATIONS AND OPERATING RANGES OF TIROS X TEMPERATURE SENSORS

Sensor Location	Operating Range
Hat, 3-in. Radial	-30 to +100°C
Hat, 12-in. Radial	-30 to +100°C
QOMAC Coil	-30 to +100°C
Side Panel	-30 to +100°C
Solar-Cell Patch	-30 to +100°C
Baseplate	-30 to +100°C
Baseplate	-20 to +10°C
Camera No. 2	+10 to +40°C
TV Xmtr No. 1	+10 to +40°C
Beacon Xmtr No. 2	+10 to +40°C
Camera Clock No. 2	+10 to +40°C
Battery Pack	+10 to +40°C
Baseplate	+10 to +40°C

5. Mechanical Integration

The integration of components on spacecraft OT-1 followed the assembly procedures used on previous TIROS spacecraft. A chronology of major events in the mechanical integration of spacecraft OT-1 is presented in Table II-6. Table II-7 lists the serial numbers and weights of spacecraft components.

6. Interface Check

On March 30, 1965, the X-258 attach fitting assembly was mounted and centered on a rotary table which, in turn, was affixed to a surface plate. Flatness and concentricity measurements were made of the spacecraft interface and reference surfaces.

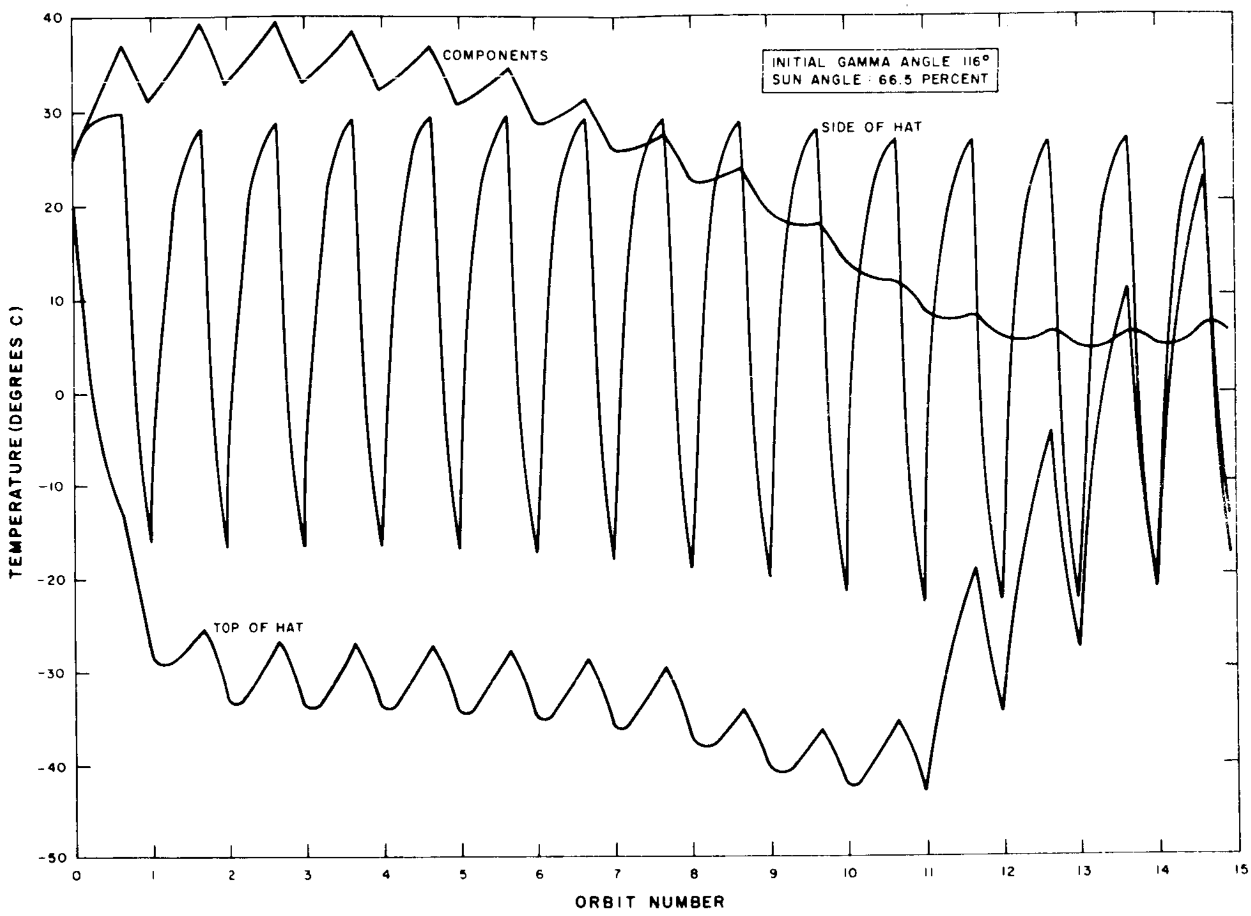


Figure II-17. Temperature Profile for TIROS X During Initial Torquing, Assuming Gamma Angle of 116 Degrees

The flatness of the interface surface was well within 0.003 inch TIR, and the concentricity of the three radial reference surfaces was also within this value.

Spacecraft OT-1 was then mounted on the attach fitting and the interface mating between the two was within 0.0015 inch at every point around the periphery.

Mating of the lift-off and separation switch actuators with the contact plate in the attach fitting was observed; and, in all four cases, the actuators struck the plate satisfactorily, i.e., at least 0.25 inch from the clearance slot in the center of the plate.

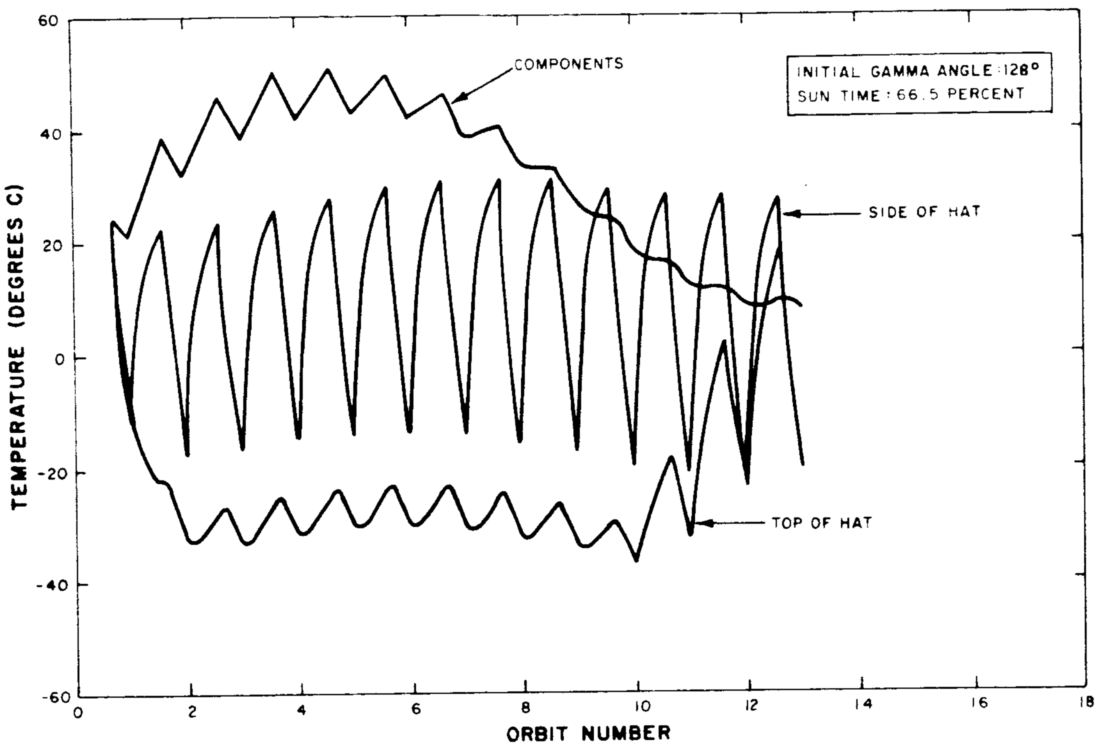


Figure II-18. Temperature Profile for TIROS X During Initial Torquing, Assuming Gamma Angle of 128 Degrees

A runout measurement was made at the top of the spacecraft, employing the eccentric plug which is used at the launch site during balancing of the spacecraft/third-stage rocket assembly. The eccentrics in the plug were set at "0", and the runout measurement was 0.024 inch TIR, well within the allowable limit.

The interface check was attended by representatives from NASA, the Douglas Aircraft Company, and AED. At the completion of the check, the group reviewed the results and agreed that the interface mated properly and that no difficulties at the launch site should be anticipated.

During this check, it was noted that the fairing for spacecraft OT-1 would be identical to that for TIROS IX, with the addition of camera illuminating lights and "in-line" translucent panels in the smoke shield.

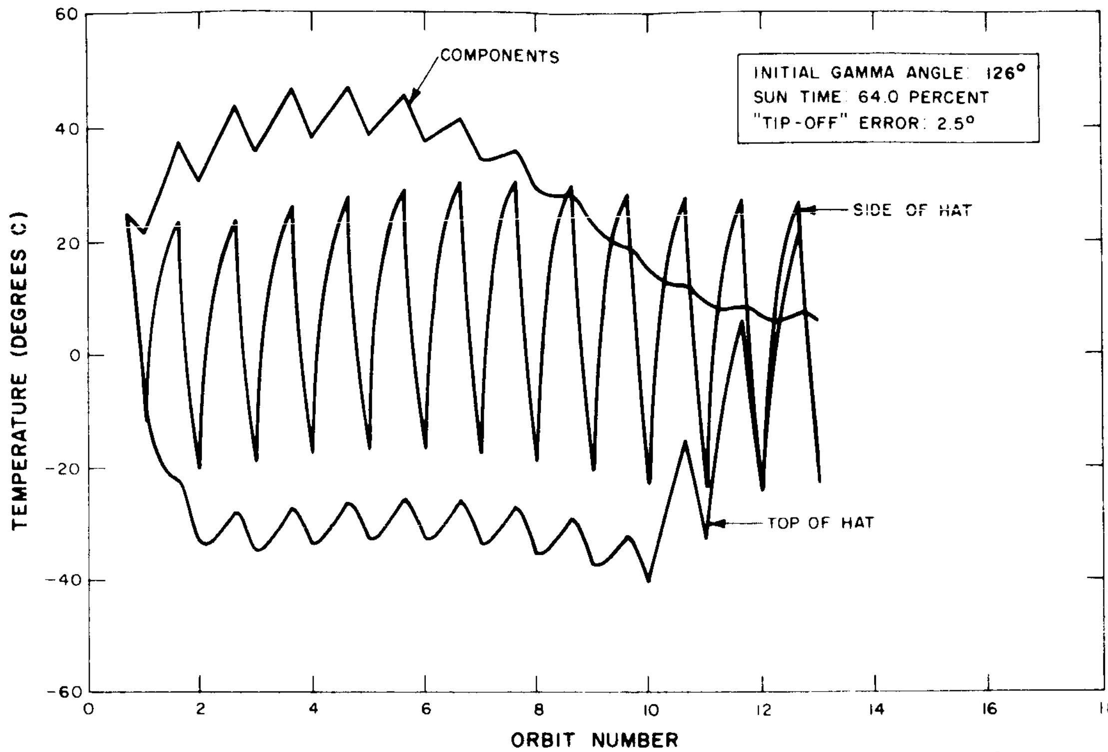


Figure II-19. Temperature Profile for TIROS X During Initial Torquing, Assuming Gamma Angle of 126 Degrees

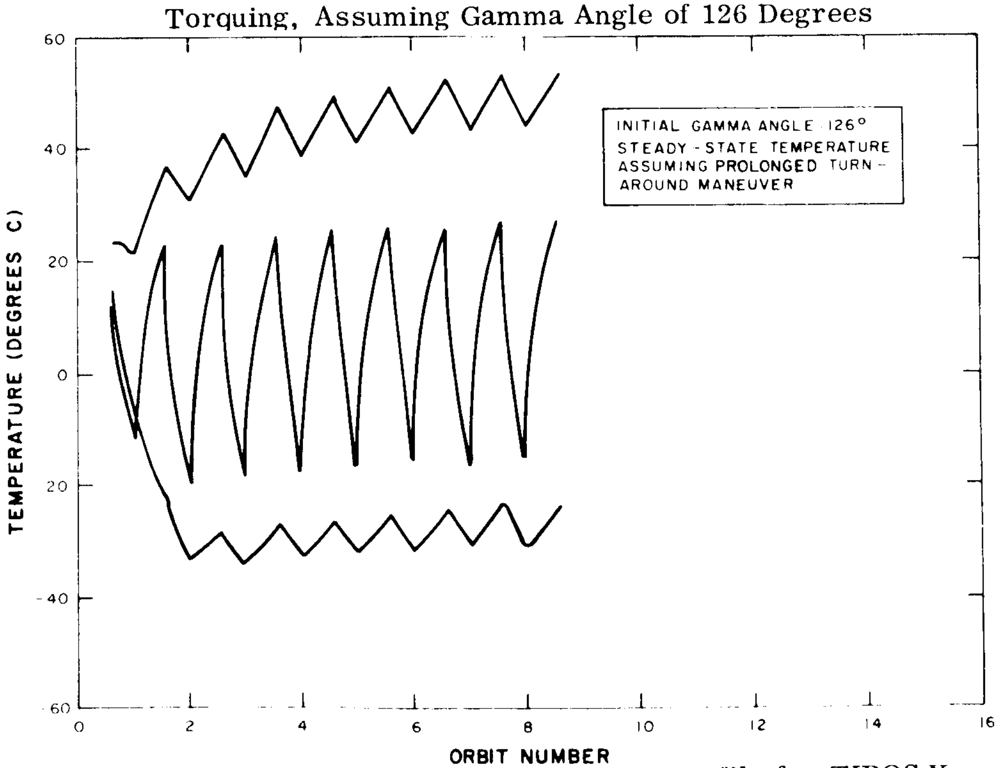


Figure II-20. Steady-State Temperature Profile for TIROS X, Assuming Gamma Angle of 126 Degrees

TABLE II-6. CHRONOLOGY OF MECHANICAL INTEGRATION OF SPACECRAFT OT-1

1965	
Date	Action
Feb 24	Tape recorder, R-2 box, co-axial relay, telemetry switch, command receiver, and camera S/N 01 removed.
Mar 1	Horizon sensor S/N 01 removed.
Mar 4	TV camera electronics removed.
Mar 8	"HOLD" STATUS FORMALLY RESCINDED BY NASA DIRECTIVE.
Mar 9	Horizon-sensor wedges installed.
Mar 18	Installation and "tie-down" of all subsystems initiated.
Mar 30	Spacecraft OT-1 and Douglas attach fitting mated.
April 5	Electrical integration check-out of spacecraft performed.
April 9	Spacecraft placed into Tenney chamber.
April 21	One-year timers removed.
April 23	Optical alignment of sensors measured.
April 27	Fine dynamic balance performed.
April 28	RF tests performed.
May 1	Spacecraft placed in Tenney chamber, then removed at 3:00 P.M., and SEPET begun.
May 2	Initial camera alignment and distortion test performed.
May 3	Spacecraft placed in thermal-vacuum chamber for qualification tests.
May 5	Spacecraft removed from chamber for replacement of R-2 boxes, and camera system No. 1.
May 6	Spacecraft reinstalled in thermal-vacuum chamber.
May 13	Spacecraft removed from thermal-vacuum chamber. Flight-model battery pack installed on spacecraft.
May 15	Vibration phase of qualification tests performed.
May 18	Spacecraft installed in thermal-vacuum chamber.
May 22	Spacecraft removed from chamber because of problem with TV transmitter No. 2.
May 25	Spacecraft installed in thermal-vacuum chamber.
May 27	Spacecraft removed from thermal-vacuum chamber.
June 1	Magnetometer test conducted.
June 3	Final optical alignment and distortion tests, camera scene-brightness tests, and final balancing performed.
June 4	Outdoor solar-cell test performed.
June 11	Spacecraft placed in shipping container.

TABLE II-6. CHRONOLOGY OF MECHANICAL INTEGRATION OF SPACECRAFT OT-1 (Continued)

1963	
Date	Action
June 11	Structure received from Purchased Material Inspection.
June 24	Side panels removed from spacecraft hat for attachment of solar-cell shingles. Despin pans fitted to the hat structure.
Aug 7	Solar-cell shingles installed on the top of spacecraft hat.
Sept 17	Baseplate drilled for mounting of components; subsystem installation started.
Oct 30	Hat harness installed.
Nov 22	All subsystems installed except for cameras, diplexer, and north-indicator sun sensors.
Dec 2	Despin pans and dampers installed.
Dec 5	Cameras installed. Hat wiring completed, except for magnetic attitude coil.
Dec 6	All hat wiring bonded.
Dec 10	Magnetic attitude coil installed in hat.
Dec 19	Baseplate dampening brackets fitted and installed.

1964	
Date	Action
Jan 22	Hat taken outdoors, into sunlight, for solar-cell test.
Jan 27	Both tape-recorder transports and R-3 boxes removed.
Feb 13	TV transmitter No. 1 replaced.
Mar 2	SPACECRAFT PLACED IN "HOLD" STATUS BY NASA DIRECTIVE
Mar 9	Dummy weights replacing IR equipment installed. Spacecraft assembly completed.
Mar 10	Rough dynamic balance performed, and balance weights installed.
Mar 16	Initial optical alignment performed.
Mar 23	Hat taken outdoors, into sunlight, for solar-cell test.
Mar 25	Fine dynamic balance performed.

1965	
Date	Action
Feb 16	Reconfiguration activity initiated in anticipation of removal of the "hold" status on spacecraft OT-1.
Feb 23	TEAM precession dampers installed.

TABLE II-7. SERIAL NUMBERS AND WEIGHTS OF SPACECRAFT OT-1 COMPONENTS

Components	Serial No.	Weight (Grams)
Camera System (Sys.) No. 1 (Shutter S/N 06)	3-201-01	2675
Camera Sys. No. 2 (Shutter S/N 03)	3-202-02	3165
Camera Electronics, Sys. No. 1	3-203-01	1792
Camera Electronics, Sys. No. 2	3-203-02	1810
Tape Transport (R-1 Box), Sys. No. 2	3-301-01	4716
Tape Transport (R-1 Box), Sys. No. 1	3-301-02	4695
Tape Recorder Power Conv (R-2 Box), Sys. No. 2	3-302-01	1410
Tape Recorder Power Conv (R-2 Box), Sys. No. 1	3-302-03	1413
Tape Recorder Signal Cond (R-3 Box), Sys. No. 1	3-303-03	1077
Tape Recorder Signal Cond (R-3 Box), Sys. No. 2	3-303-01	1078
TV Transmitter, Sys. No. 1	3-401-07	860
TV Transmitter, Sys. No. 2	3-401-04	860
TV Transmitter Filter Sys. No. 1	3-402-02	29
TV Transmitter Filter Sys. No. 2	3-402-03	28
Command Receiver	3-403-04	1115
Beacon, Sys. No. 2	3-405-02	305
Beacon, Sys. No. 1	3-406-01	305
Antenna Receiving	3-407-02	123
Antenna Transmitting	3-408-01	893
Antenna Coupling and Matching Network	3-409-01	936
Harness, RF, Receiving	3-410-01	864.7
Harness, RF, Transmitting	3-411-01	864.7
MBC Switch	3-502-03	290
Precession Damper	3-503-08	549
Precession Damper	3-503-12	548
Despin Timer	3-504-01	155
Lift-Off Switch Assembly	3-505-21	360
Rocket Switch	3-506-05	181
Despin Mechanism	3-507-3	672

TABLE II-7. SERIAL NUMBERS AND WEIGHTS OF SPACECRAFT OT-1 COMPONENTS (Continued)

Components	Serial No.	Weight (Grams)
Despin Mechanism	3-507-4	673
QOMAC Coil	None	2977
Camera Control Unit, Sys. No. 1	3-601-2	998
Camera Control Unit, Sys. No. 2	3-602-2	
Auxiliary Control Unit	3-603-2	1121
Command Address Unit	3-607-2	465
Camera Clock, Sys. No. 1	3-608-19	1691
Camera Clock, Sys. No. 2	3-609-22	1747
Oscillator	3-610-S722	66
Oscillator	3-610-S723	84
QOMAC Clock	3-614-24	1786
QOMAC Control Unit	3-615-02	614
Camera Clock Board, Sys. No. 2	3-616-03	230
Camera Clock Board, Sys. No. 1	3-616-02	208
Battery Pack	3-701-06	17,756
DC/DC Converter, Sys. No. 1	3-702-01	873
DC/DC Converter, Sys. No. 2	3-702-05	866
Power Supply Protection Unit	3-703-02	914
Voltage Regulator, Sys. No. 1	3-704-01	718
Voltage Regulator, Sys. No. 2	3-704-02	701
Current Telemetry Board	None	84.5
Fuse Board	3-728-03	39
Attitude Horizon Scanner		
"Down-Looking" Sensor	3-901-1	256
"Up-Looking" Sensor	3-901-2	204
Telemetry Switch	3-902-05	886
Telemetry Switch	3-902-06	886
North-Indicator Sensor (Not Connected)	---	960
Solar Aspect Indicator	3-923-106	37
Solar Aspect Indicator Electronics	3-924-106	647

SECTION II. SPACECRAFT TESTS

A. INTRODUCTION

Each new or modified unit for the reconfigured spacecraft OT-1 was extensively tested to ensure that it met the TIROS subsystem prototype- and flight-level specifications before that unit was accepted for integration with the spacecraft. After spacecraft integration, electrical checkout, and initial alignment and calibration had been completed, the spacecraft was subjected to the TIROS environmental qualification-test sequence.

The Environmental Test Committee, comprising four engineering representatives from NASA and three from AED, was established before the start of the environmental test program, to review and approve spacecraft performance and test results.

The initial assembly and debugging of spacecraft OT-1 in its original configuration was completed in March, 1964, and the spacecraft was maintained in a "hold" status until February 1965, when the reconfiguration was directed by NASA. Assembly and debugging of the reconfigured spacecraft was completed by April, 1965, and qualification procedures were initiated at that time.

Environmental testing of spacecraft OT-1 was initiated on May 3, 1965, and completed on May 26. Upon completion of environmental testing, final alignment and focus of the TV cameras were carefully checked, and the field-of-view and distortion characteristics were recorded by using the cameras to photograph a special test target. At that time the final alignment checks of the attitude horizon scanner and the solar-aspect indicator were also performed.

The spacecraft was then dynamically balanced and weighed; the center-of-gravity was determined; and the moments of inertia and the magnetic dipole moments were measured. On June 9, 1965, based upon results of these checks and measurements, NASA declared spacecraft OT-1 ready for delivery. The spacecraft was shipped to the Eastern Test Range on June 14.

After its arrival at the launch site, the spacecraft was carefully tested on a regular basis. Based on the performance of the spacecraft during these tests, NASA accepted the spacecraft as being flight-qualified. The spacecraft was successfully launched and orbited on July 1, 1965.

B. COMPONENT TESTS

In line with the test philosophy used for the previous TIROS programs, all new components were subjected to prototype-level testing. Upon receipt of each group of purchased satellite parts, a complete quality inspection and test was conducted by the Purchase Material Inspection activity to ensure that the parts met specifications.

It was required that the prototype model of each new component satisfactorily complete the following series of tests:

(1) Vibration

  (a) Thrust Axis: 21 g rms, 2 minutes, 20 to 2000 cps.

  (b) Lateral Axis No. 1: 15 g rms, 2 minutes, 20 to 2000 cps, and

  (c) Lateral Axis No. 2: 15 g rms, 2 minutes, 20 to 2000 cps.

(2) Sustained Acceleration (Positive and Negative)

  (a) Thrust Axis: 50 g, 5 minutes,

  (b) Lateral Axis 1: 50 g, 5 minutes, and

  (c) Lateral Axis 2: 50 g, 5 minutes.

(3) Thermal Vacuum (5×10^{-5} mm Hg)

  (a) Hot cycle: +60 °C, 12 hours,

  (b) Ambient cycle: +25 °C, 12 hours, and

  (c) Cold cycle: -15 °C, 12 hours.

The flight models of new components were subjected to vibrations of 10 g rms in the thrust direction and in each of two, mutually perpendicular lateral (transverse) axes. The frequency spectrum for these vibrations was 20 to 2000 cps, and the duration of each test was 2 minutes.

After vibration, the new or modified flight-model components were mounted in a thermal-vacuum chamber (5 x 10^{-5} mm Hg) and checked for satisfactory operation at the following temperatures and periods:

- -10°C for 12 hours,
- +25°C for 12 hours, and
- +55°C for 12 hours.

C. QUALIFICATION TESTING OF SPACECRAFT OT-1

1. General

Before the initiation of the test program on spacecraft OT-1, and after each major phase of testing, a complete SEPET (Standard Electrical Performance Evaluation Test) was performed to provide a thorough electrical checkout of the spacecraft. (The SEPET is described in detail in Volume II of this report.) The environmental qualification tests were also preceded by the initial balancing of the spacecraft and the initial alignment and calibration effort.

The environmental test requirements for spacecraft OT-1 are listed in Table II-8. The sequence followed in the TIROS X test program was in accordance with qualification test procedures and was as follows:

(1) SEPET

(2) Initial Alignment and Calibration

(3) Initial Balancing

(4) SEPET

(5) ENVIRONMENTAL TESTING: First Phase of Thermal-Vacuum Test

(6) SEPET

(7) ENVIRONMENTAL TESTING: Vibration Test

(8) SEPET

(9) ENVIRONMENTAL TESTING: Second Phase of Thermal-Vacuum Testing

TABLE II-8. SPACECRAFT OT-1 ENVIRONMENTAL TEST REQUIREMENTS

Thermal-Vacuum Test: First Phase		
Duration	Temperature Level	Pressure
3 days 2 days	+50 °C 0 °C	Not exceeding 5×10^{-5} mm Hg at either temperature
Vibration Test		
Test Axis	Vibration	Test Parameters
Three mutually perpendicular axes (one thrust axis and two lateral axes)	Random	7.7 g rms (equalized to ± 3 db) 20 to 2000 cps, 2-minute duration, with sharp roll-off above 2000 cps
Thermal-Vacuum Test: Second Phase		
Duration	Temperature Level	Pressure
1 day 3 days	+35 °C 0 °C	Not exceeding 5×10^{-5} mm Hg at either temperature

(10) SEPET

(11) Final Mechanical Inspection and Electrical Test

(12) Measurement of Magnetic-Dipole Moments

(13) Final Alignment and Calibration

(14) Final Balancing and Weighing

(15) Determination of Moment of Inertia

As can be noted in this listing, thermal-vacuum environmental testing in the TIROS X test program was divided into two phases, interspersed by vibration testing. The initiation of vibration testing was dependent on the successful completion of the first phase of thermal-vacuum testing, while the initiation of the second phase of thermal-vacuum testing was dependent on the successful completion of vibration testing.

For the first phase of thermal-vacuum testing, the flight-model battery pack was removed from the spacecraft and a spare pack substituted. The flight-model pack was reinstalled for vibration testing and the second phase of thermal-vacuum testing.

The purpose of this procedure, which was also followed in the TIROS VIII and TIROS IX programs, was to avoid subjecting the flight-model battery pack to long exposure to the high test temperature involved in the first phase of thermal-vacuum testing, since such temperature would not be experienced by the battery pack in the space environment.

For the TIROS X test program, the requirements were made similar to those of the TIROS VII test program because of the similarity between the two spacecraft. (The requirements for a vibration survey and flight-level sinewave vibration test of the integrated spacecraft, such as had been performed on the TIROS IX test program were removed by NASA directive.) During vibration testing, power was applied only to those components that would be operative during the launch phase. During thermal-vacuum testing, the spacecraft was tested on an operational basis at 2-hour intervals and its responses were carefully monitored and recorded.

Table II-9 presents a summary of the various tests, checks, and measurements performed in the TIROS test program, and the respective completion dates.

2. Initial Alignment and Calibration

The initial alignment of both the attitude horizon scanner and the solar-aspect indicator was performed on April 23, 1965.

The initial alignment and calibration procedures on the TV cameras were performed on May 2.

The procedures followed in both the initial and final alignment and calibration efforts on spacecraft OT-1 were identical, and are detailed in the TIROS X Alignment and Calibration Handbook.*

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*"Alignment and Calibration Data for the TIROS X Meteorological Satellite System," Astro-Electronics Division Radio Corporation of America, AED M-2058, 14 June 1965.

TABLE II-9. SUMMARY OF TIROS X TEST PROGRAM

Test	Completion Date (1965)
Initial Balancing	April 27
Initial Alignment and Calibration	
Attitude Horizon Scanner	April 23
Solar-Aspect Indicator	April 23
TV Cameras	May 2
Environmental Tests	
Thermal-Vacuum (First Phase)	May 12
Vibration	May 15
Thermal-Vacuum (Second Phase)	May 27
Measurement of Magnetic-Dipole Moments	June 2
Final Alignment and Calibration	
Attitude Horizon Scanner	June 3
Solar-Aspect Indicator	June 2
TV Cameras	June 3
Final Balancing and Weighing	June 4
Determination of Moment of Inertia	June 4

3. Initial Balancing

On April 27, 1965, initial balancing of spacecraft OT-1 was performed. Balance weights were added as follows:

- Top of Hat: 690 gms at the 144.5-degree radial, 19.25-inch radius;
- Bottom of Hat: 493 gms at the 10-degree radial, 20.16-inch radius;
- Bottom of Hat: 1640 gms at the 350-degree radial, 20.16-inch radius;
- Edge of Baseplate: 216 gms at the 298-degree radial, 19.90-inch radius;

- Top of Baseplate: 3710 gms at the 20-degree radial, 8.63-inch radius;
- Top of Baseplate: 2360 gms at the 130-degree radial, 15.63-inch radius.

The last two items on this list were counterweights added to the spacecraft to replace the IR system equipment which had been part of the original configuration fo spacecraft OT-1.

4. Thermal-Vacuum Phases of Environmental Tests

The first phase of thermal-vacuum testing of spacecraft OT-1, i.e., the previbration phase, was initiated on May 3, 1965, at the 50 °C temperature level. Testing was halted on May 4, when it was observed that the high temperature resulted in marginal performance of TV camera No. 1 and excessive current drain in the tape recorder No. 1 power supply.*

The spacecraft was removed from the thermal-vacuum chamber on May 5. On May 6, following an investigation of the problem and the implementation of the required minor modification, the spacecraft was reinstalled in the chamber. Thermal-vacuum testing was resumed on May 7, and on May 12 the first phase of thermal-vacuum testing was successfully completed. On May 13, the flight-model battery pack was installed, and on May 14 a full SEPET was satisfactorily performed on the spacecraft.

On May 18, following the successful completion of vibration testing and a full SEPET, the second phase of thermal-vacuum testing was initiated. Testing was halted on May 22 because a change had been noted in the deviation sensitivity and carrier frequency of TV transmitter No. 2 during the first several seconds of transmitter operation. The condition was investigated, and it was decided to resume the qualification cycle when it was determined that the condition was not indicative of a problem in the transmitter and did not result in the loss or degradation of normal data.

Thermal-vacuum testing was re-initiated on May 25 and successfully completed on May 27. On May 28, the spacecraft was successfully subjected to a full SEPET.

Figure II-21 shows spacecraft OT-1 prepared for thermal-vacuum testing.

5. Vibration Phase of Environmental Tests

On May 15, 1965, after the flight-model battery pack had been installed, spacecraft OT-1 was subjected to vibration testing in the thrust axis and both of two mutually

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*This condition and the subsequent actions are detailed later in this section under "History of Testing and Qualification".

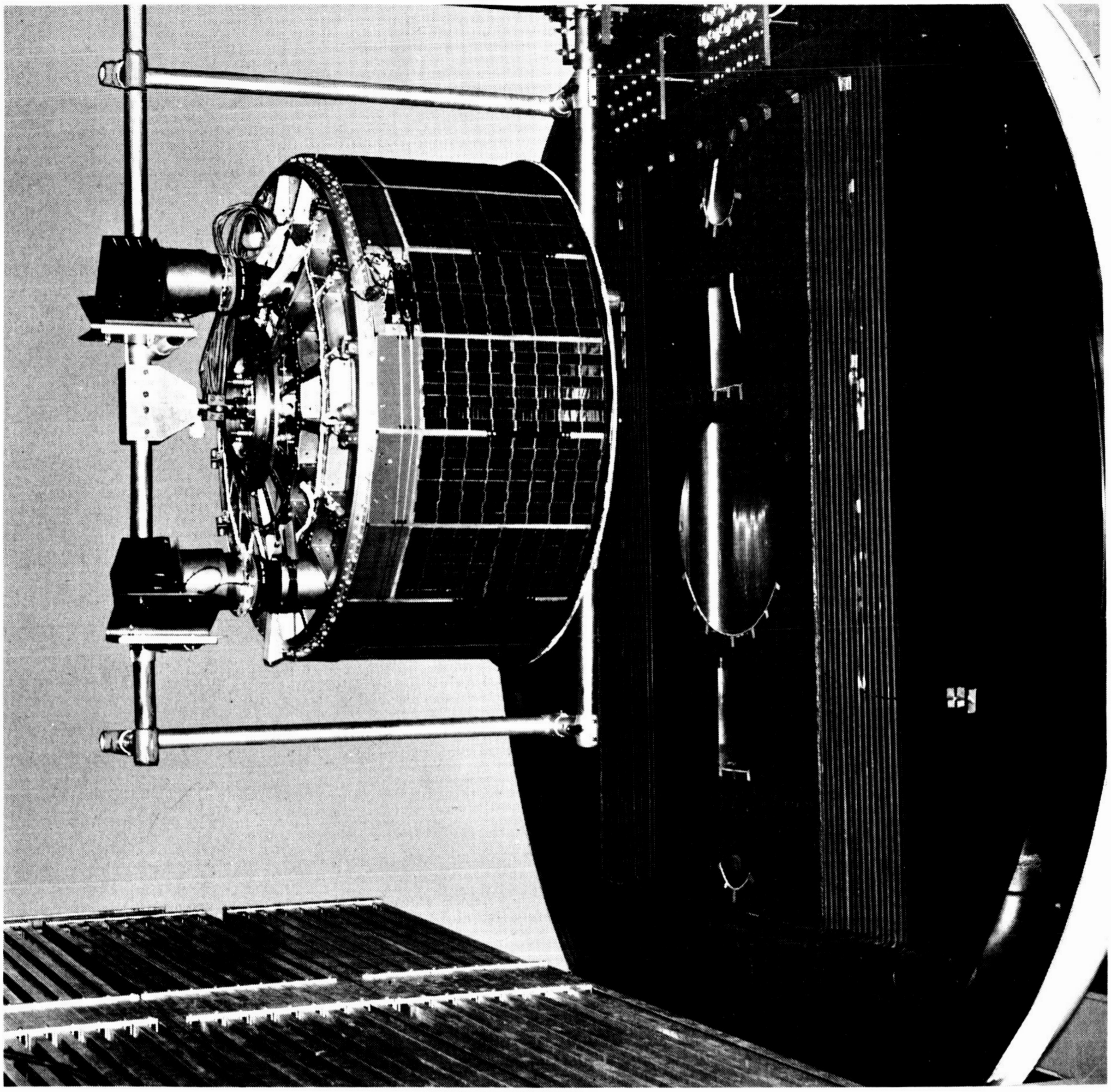


Figure II-21. Spacecraft OT-1 Prepared for Thermal-Vacuum Testing perpendicular lateral axes, with satisfactory results. Random vibration was utilized, and the test parameters, which were the same in each axis of vibration, were as follows: 7.7 gms (equalized to ± 3 db), 20 to 2000 cps, 2-min duration, with sharp roll-off above 2000 cps.

For vibration testing, the spacecraft was mounted to the 6-inch Douglas attach fitting (including the 1-inch spacer and separation spring) by means of a Marman clamp. Post-vibration examination of the Marman clamp and separation plane interfaces uncovered indications of galling. However, the extent of the galling was minor, bordering on simple burnishing of the metal surfaces.

On May 17, following the successful completion of vibration testing, a full SEPET was performed on spacecraft OT-1, with satisfactory results.

Figure II-22 is a block diagram of the test set-up used in the vibration testing of spacecraft OT-1.

6. Final Checks and Measurements

a. General

A series of final checks was performed on spacecraft OT-1 after the completion of environmental testing. These checks included:

(1) a mechanical inspection to ensure that all fasteners were tight, that all electrical connections were secure, and that the spacecraft was free from foreign material, and

(2) an electrical test of systems operation (basically a repeat of the SEPET).

After the final checks were completed, the spacecraft's magnetic dipole moments were measured, final alignment and calibration procedures were completed, final balancing and weighing was performed, and the spacecraft's moment-of-inertia was determined.

The post-environmental-testing procedures on spacecraft OT-1 were initiated on June 2, 1965, and completed on June 8, with a final inspection of the spacecraft. On June 9, at a meeting held at AED with NASA representatives, the spacecraft was accepted by NASA; and on June 11, after the spin-up rockets had been installed, the spacecraft was prepared for shipment. On June 14, spacecraft OT-1 was shipped to the launch site at Cape Kennedy.

The procedures used in the final alignment and calibration efforts were essentially the same as those used in the initial efforts. Figure II-23 shows the location and

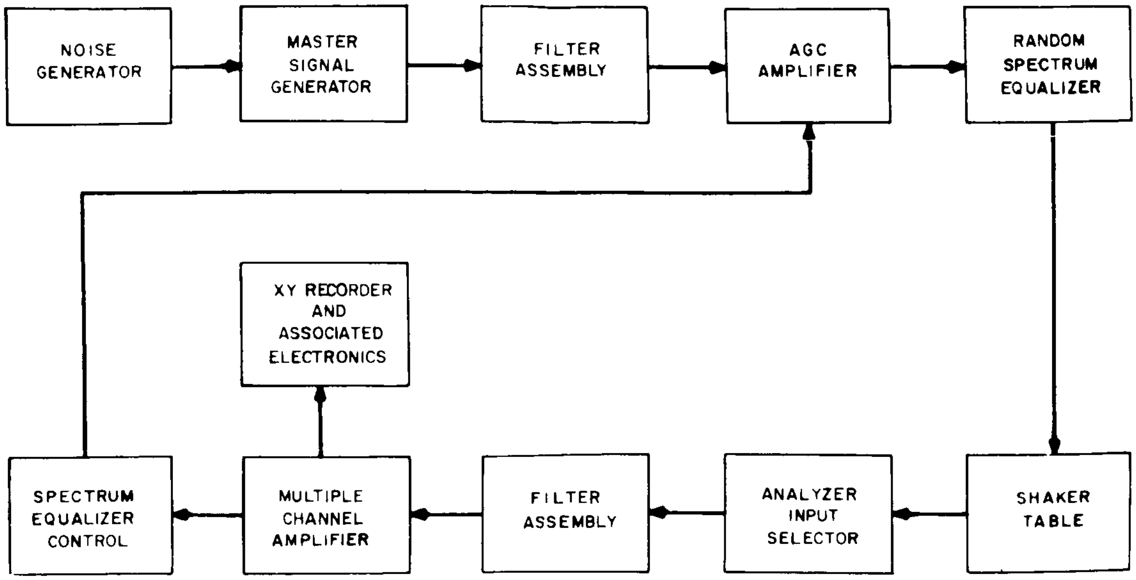


Figure II-22. Vibration Test Set-Up

orientation of (1) the TV cameras, (2) the sensors composing the attitude horizon scanner, and (3) the solar-aspect indicator on TIROS X.

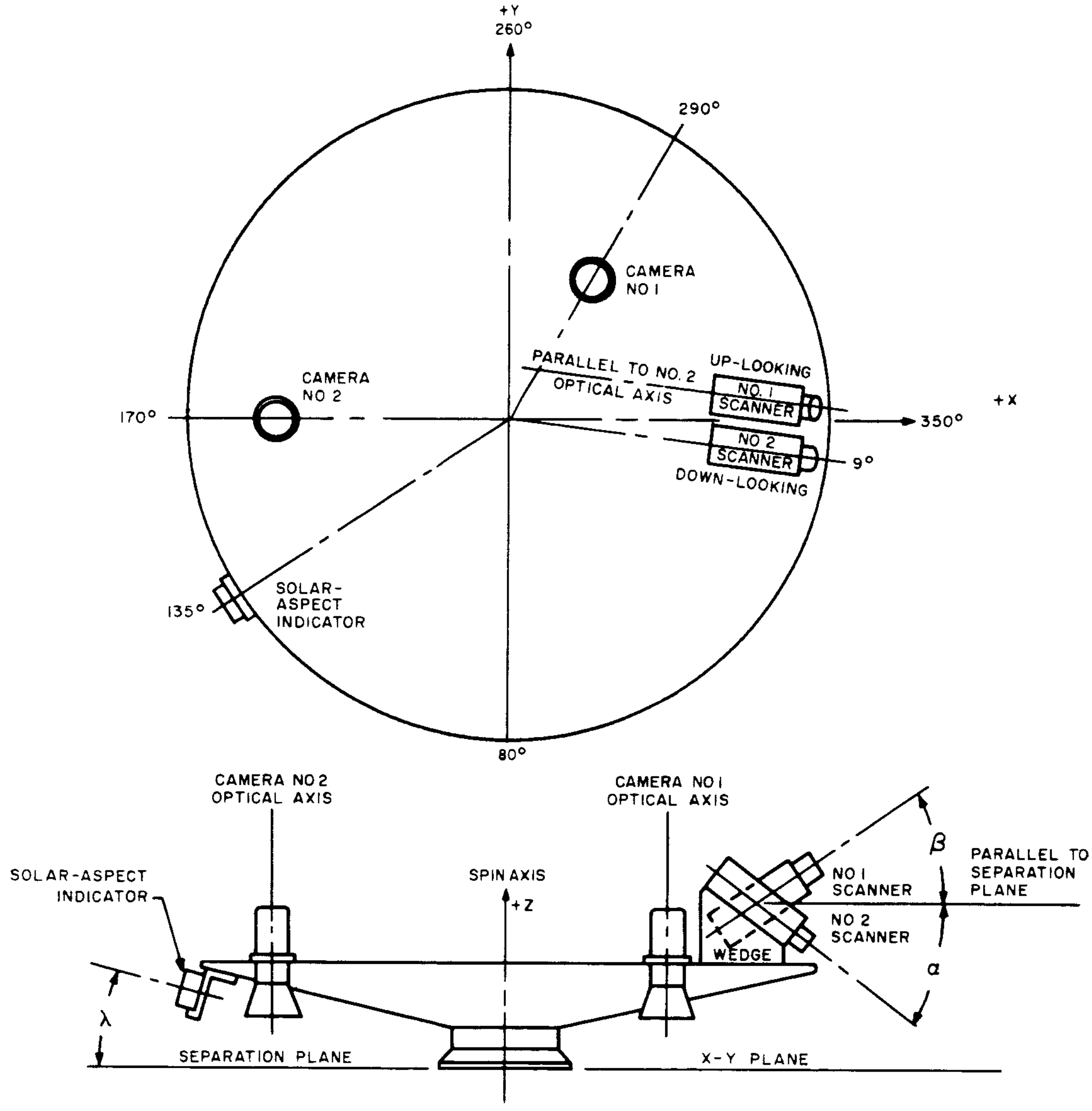
b. **Measurement of Magnetic-Dipole Moments**

On June 2, 1965, the residual spin axis magnetic moment of spacecraft OT-1 was measured, with the spacecraft operating in the standby mode. The residual moment was then cancelled by the addition of a permanent magnet with a total strength of 1.14 ampere-turns-meter^2. The net magnetic moments for the various operating modes of the spacecraft were then measured. The values obtained are listed in Table II-10. The magnetic dipole moments generated by the MBC coil for each position of the MBC switch were also measured. The values obtained are listed in Table II-11, along with the associated telemetry voltages. The sum of the applicable values in Tables II-10 and II-11 added to the residual magnetic moment for the standby mode, yields the instantaneous, total magnetic dipole moment along the spin axis for any single operating mode or combination of modes.

c. **Final Alignment and Calibration**

(1) Final Alignment of TV Cameras

The deviation between the optical and nodal axes was measured on both TV cameras on June 3, 1965. (The nodal axis is defined as a line which is parallel



α	β	λ	DIVERGENCE OF CAMERA NO.1 OPTICS VS. SPIN AXIS	DIVERGENCE OF CAMERA NO. 2 OPTICS VS. SPIN AXIS
39° 55' (NOMINALLY 40°)	40° 01' (NOMINALLY 40°)	25° 12' (NOMINALLY 25° 30')	0° 14' (ON 70°-250° AXIS POINTING TOWARDS 250°)	0° 8' (ON 80° - 260° AXIS POINTING TOWARDS 80°)

Figure II-23. Location and Orientation of TIROS X Sensing Elements

TABLE II-10. SPIN AXIS MAGNETIC DIPOLE MOMENTS ON TIROS X

Operating Mode			Magnetic Moment (ampere-turns-meter^2)
Standby:	night		+ 0.07
Clock I:	remote		0.00
	timing		0.00
Clock II:	remote		+ 0.06
	timing		0.00
Direct Camera I			- 0.03
Direct Camera II			+ 0.13
QOMAC Coil:			
Clock I:	remote	- positive	+27.80
		- negative	-27.50
Clock I:	off	- positive	+27.80
		- negative	-27.50

TABLE II-11. MAGNETIC DIPOLE MOMENT PRODUCED FOR EACH POSITION OF THE MBC SWITCH

Switch Position	Magnetic Dipole Moment (ampere-turns-meter^2)	Telemetered Voltage (volts)
0 (or 12)	0.00	0
1	+0.19	0.25
2	+0.44*	0.75
3	+0.76*	1.25
4	+1.56*	1.75
5	+8.10	2.25
6	0.00	0
7	-0.19	2.50
8	-0.44*	2.00
9	-0.76*	1.50
10	-1.56*	1.00
11	-8.41	0.50

*Calculated Values

to the spin axis and which passes through the front nodal point of the camera lens, as shown in Figure II-24.) Both the deviation angle, ρ, and the direction of deviation, θ, were measured. The values obtained were as follows:

Camera	ρ	θ
No. 1	0° 14' 32"	302°
No. 2	0° 7' 52"	283°

(2) Final Calibration of TV Cameras

Figures II-25a and b are direct mode photographs taken with cameras No. 1 and 2, respectively, on June 3, 1965. These photographs were used to determine the amount of distortion being introduced by the lenses of the two cameras, and to ensure the centering of the vidicon fiducial markings on the optical axis of each lens.

In addition, the subcarrier frequency-deviation for each camera system was checked to determine the effects of length of operating time. These checks were performed at various levels of scene brightness, and the results are presented in Figures II-26 and II-27.

Both cameras were then checked to determine the effects, in each direction of shutter motion, of variations in scene brightness upon subcarrier frequency. The results of these tests are shown in Figures II-28 and II-29.

Figure II-30 shows spacecraft OT-1 mounted in the calibration test fixture.

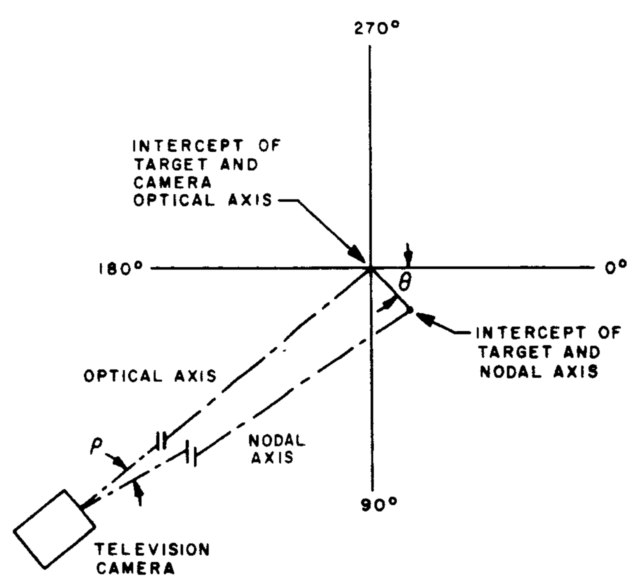


Figure II-24. Deviation of TV Camera Optical and Nodal Axes

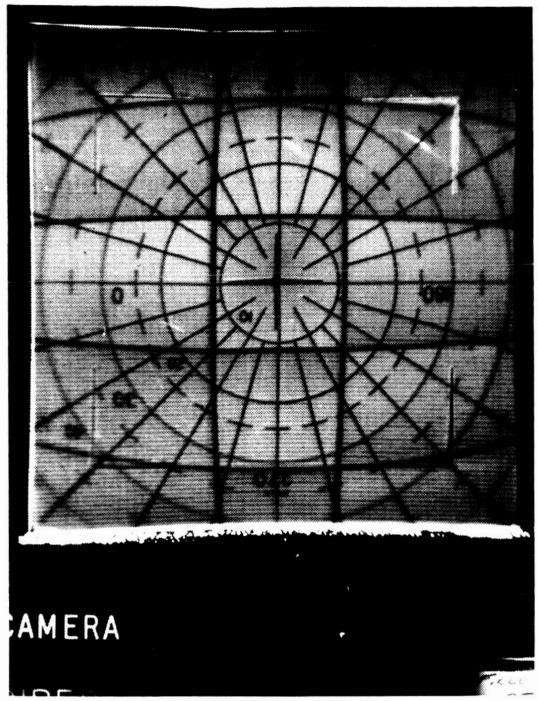


Figure II-25a. Distortion/Calibration Photograph, Camera No. 1 (Direct Mode)

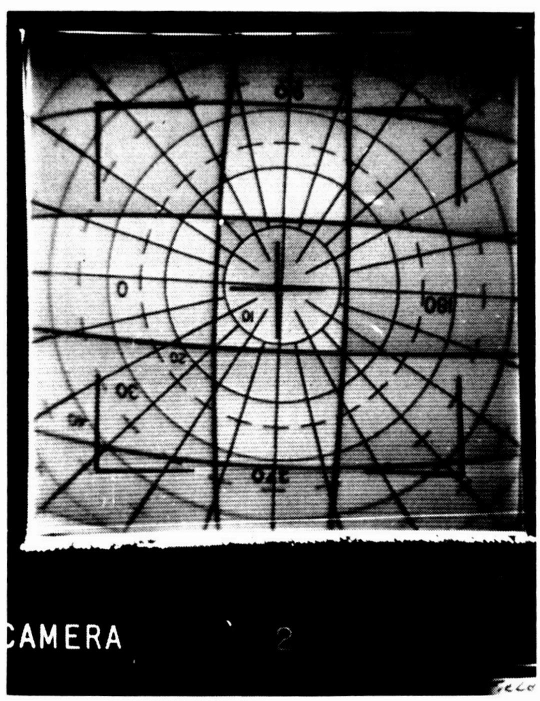


Figure II-25b. Distortion/Calibration Photograph, Camera No. 2 (Direct Mode)

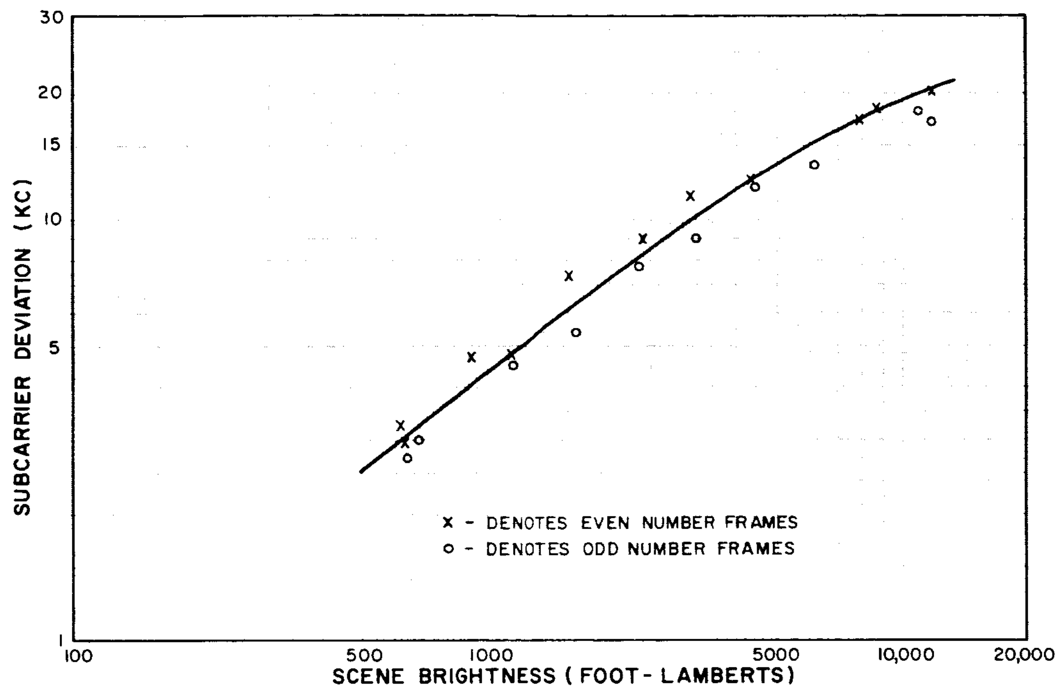


Figure II-26. Camera No. 1; Effects of Scene Brightness and Shutter-Motion Direction on Subcarrier Frequency

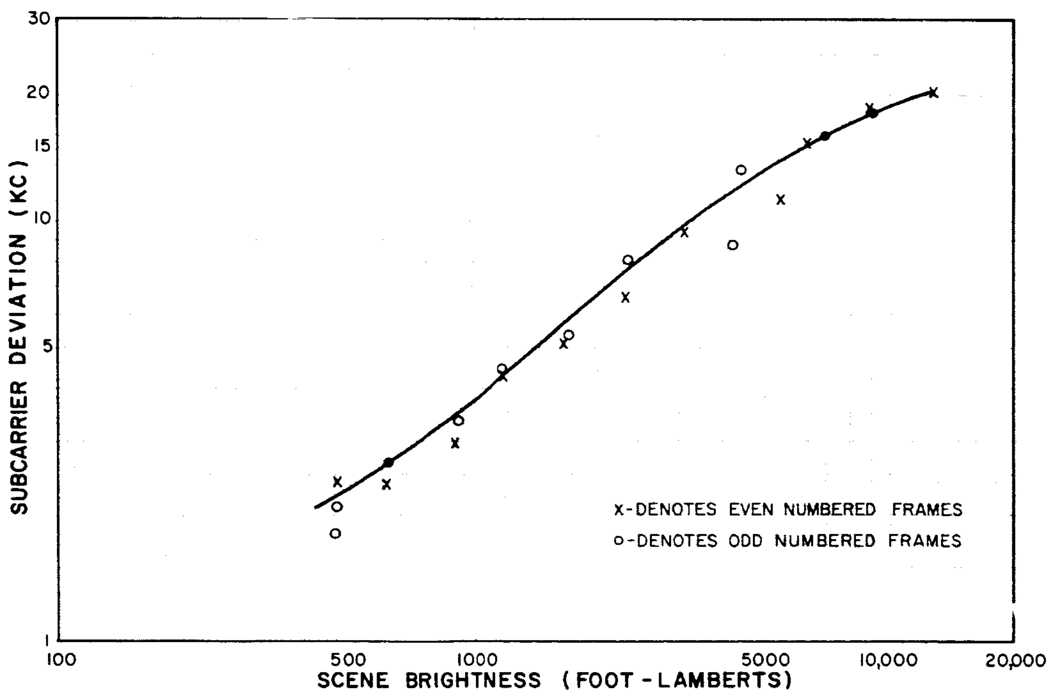


Figure II-27. Camera No. 2; Effects of Scene Brightness and Shutter-Motion Direction on Subcarrier Frequency

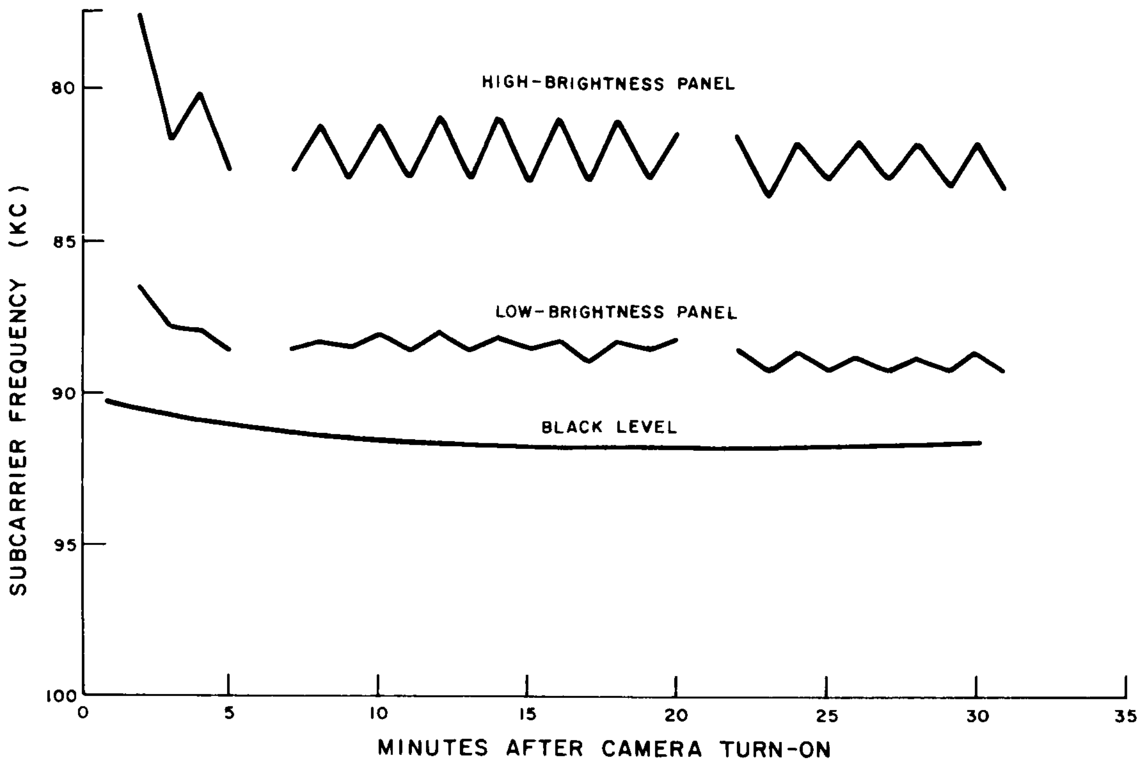


Figure II-28. Camera No. 1; Effects of Camera Operating Time on Subcarrier Frequency, at Various Levels of Scene Brightness

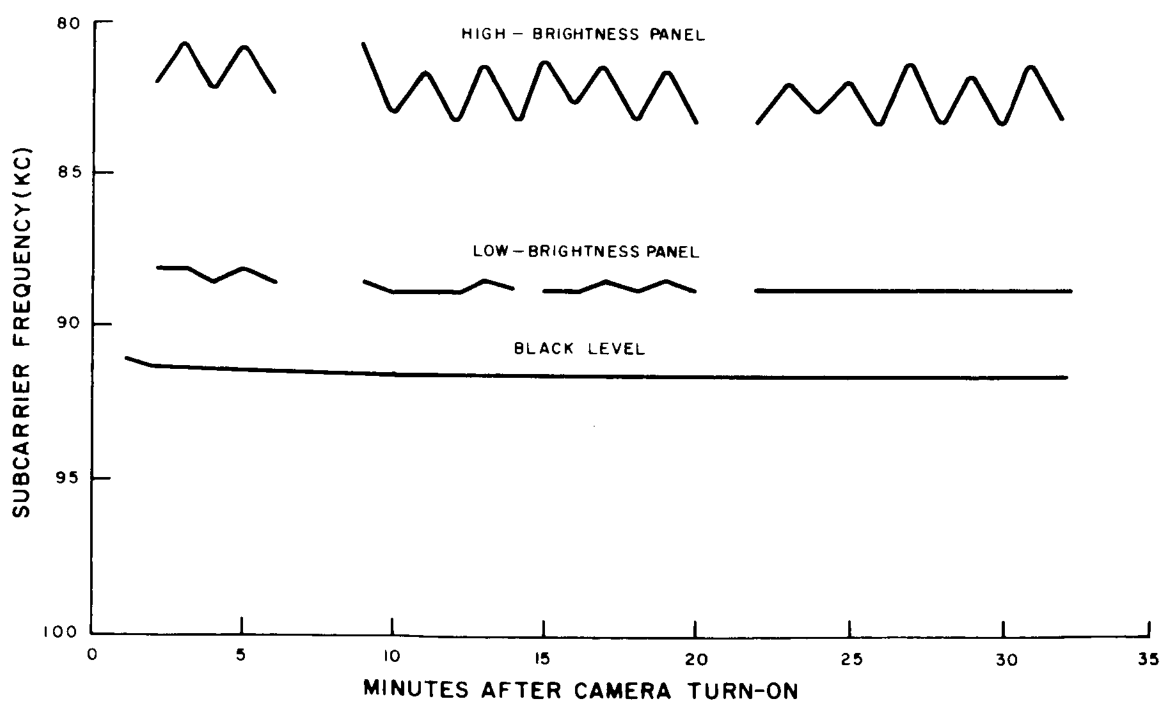


Figure II-29. Camera No. 2; Effects of Camera Operating Time on Subcarrier Frequency, at Various Levels of Scene Brightness

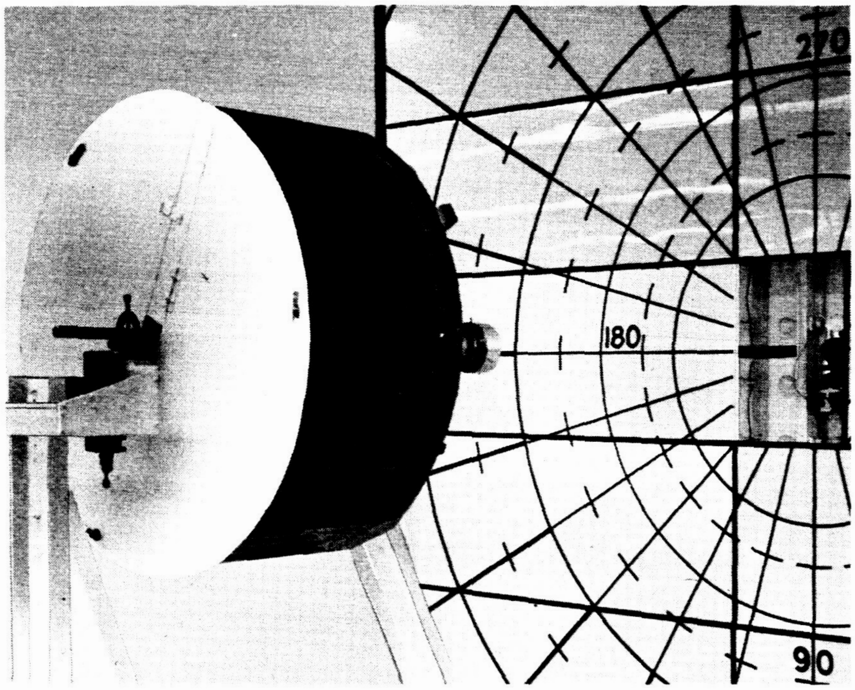


Figure II-30. Spacecraft OT-1 Prepared for Camera Calibration Tests

(3) Final Alignment of Solar-Aspect Indicator

On June 2, 1965, the angle between the x-y plane and the mechanical axis of the solar-aspect indicator, Serial No. 3-924-06, was measured and found to be 25° 12'. (The orientation of the projection on the x-y plane of the normal to the solar-aspect sensor is not important in the determination of γ and, therefore, was not measured during final calibration.)

The manufacturer of the solar-aspect indicator had previously supplied calibration data which gave the binary-coded input from the solar-aspect indicator as a function of the sun angle, based upon a nominal value of 25° 12' for the alignment of the sensor axis with respect to the x-y plane. Following final alignment of the solar-aspect indicator, the data was modified to take into account the measured value (i.e., 25° 12') of the sensor orientation.

(4) Final Alignment of the Attitude Horizon Scanner

On June 3, 1965, the orientation of the two sensors comprising the attitude horizon scanner was measured, and the values obtained were as follows:

	Sensor No.1 (Serial No.3-901-2 Up-looking Sensor)	Sensor No.2 (Serial No.3-901-1 (Down-looking Sensor)
• Displacement of mechanical axis from x-y plane	40° 02'	40° 02'
• Deviation of optical axis from mechanical axis	-1'	-7'
• Displacement of optical axis from x-y plane	40° 01'	39° 55'

(5) Final Balancing and Weighing of Spacecraft OT-1

On June 4, 1965, final balancing and weighing of spacecraft OT-1 was performed. Balance weights were added as follows:

• Edge of Baseplate:	96.0 gms at 322-degree radial; 3.1 gms at 176-degree radial;
• Top of Hat (19 inches above baseplate plane):	11.0 gms at 97-degree radial; 15.0 gms at 101-degree radial; 3.0 gms at 310-degree radial.

The total weight of the balance weights added was 128.1 gms, and the dynamic balance of the spacecraft was then within a maximum of 80.4 ounce-inches^2, i.e., within the specified limits of 110 ounce-inches^2.

The final weight of the spacecraft (without the despin weights) was then measured and was found to be 287.55 pounds, and the center-of-gravity of the spacecraft in the space configuration was found to be 10.01 inches above the separation plane.

(6) Determination of Moment of Inertia

The spacecraft's moments of inertia were also determined on June 4. The data required for making these determinations was obtained by attaching the spacecraft to a bifilar suspension system as shown in Figure II-31 and measuring the period of the resultant pendulum. In the measurements of the moment of inertia, various configurations and attitudes were covered. First, the moment of inertia about the spin axis was measured while the despin weights were not attached, i.e., the spacecraft was in the space configuration. Next the moment of inertia was again measured about the spin axis, but in this case the despin weights were included so that the spacecraft would be in the same configuration as before separation of the third-stage rocket.

The maximum and minimum transverse moments were then measured with and without the despin weights, and the ratio of the maximum transverse axis moment to the spin-axis moment was determined for both cases, i.e., with and without despin weights.

The results obtained th this series of measurements are presented in Table II-12.

7. History of Testing and Qualification

a. Efforts Prior to Reconfiguration of Spacecraft OT-1

On October 25, 1963, preliminary partial debugging of spacecraft OT-1 in the original configuration was initiated. At that point the spacecraft was configured essentially like TIROS VII, and was planned for use in a 58-degree orbit. Functional checks of the spacecraft subsystems were performed during November 1963, and debugging continued through December.

In January 1964, the spacecraft hat and baseplate were integrated and tested, and a preliminary SEPET and an RF-interference test were performed in February. During March, dummy weights for the IR equipment were installed, and the initial balancing of the spacecraft was performed. At this time the spacecraft was also subjected to thermal tests in a Tenney chamber, with satisfactory results. In addition satisfactory performance of the rocket-firing circuits, the precession-damper circuits, and the despin mechanism squib-firing circuits was verified, and initial optical alignment of sensing units was performed.

At the end of March, 1964, assembly and debugging of spacecraft OT-1 had been completed and the spacecraft was ready for environmental qualification tests; however, the spacecraft was placed in a "hold" status by NASA directive.

The "hold" on spacecraft OT-1 was rescinded in March, 1965. During the period that the "hold" had been in effect, the spacecraft was subjected to periodic checks to ensure its readiness for qualification testing.

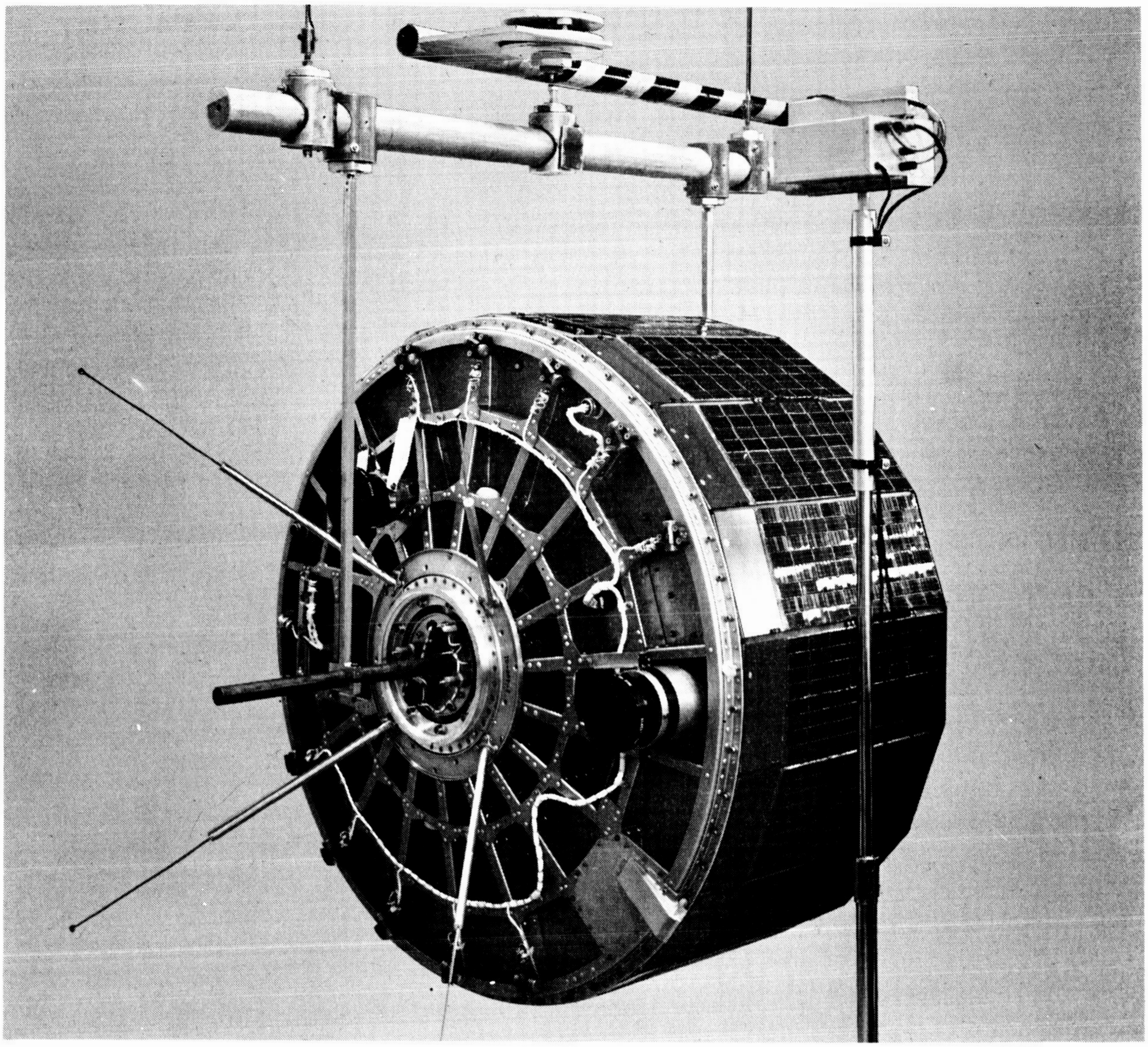


Figure II-31. Spacecraft OT-1 in Bifilar Suspension System

TABLE II-12. MECHANICAL AND PHYSICAL PARAMETERS OF TIROS X

Parameter	With Despin Weights	Without Despin Weights
Spacecraft Weight	290.07 lbs	287.55 lbs
Moment-of-Inertia		
(a) About Spin Axis	163.241 lbs-in-sec^2	160.453 lbs-in-sec^2
(b) About Transverse Axis (Maximum)	118.468 lbs-in-sec^2	118.388 lbs-in-sec^2
(c) About Transverse Axis (Minimum)	112.786 lbs-in-sec^2	109.83 lbs-in-sec^2
(d) Ratio of (b)/(a)	72.5 percent	73.8 percent
(e) Location of Maximum Moment of Inertia	245° (with keyway in separation ring at 90° and angles increasing clockwise with spacecraft viewed from hat)	
Center of Gravity		
(above separation plane)	9.99 inches	10.01 inches
Dynamic Balance		
Residual dynamic imbalance (maximum)	80.4 oz-in^2 (0.00041 radian displacement between geometric and inertial axes)	

b. **Reconfiguration and Preparation of the Spacecraft for Calibration and Qualification Tests**

In March 1965, spacecraft OT-1 was reconfigured for use in a near-polar orbit. This effort included the following:

(1) addition of the QOMAC system, the solar-aspect indicator, and the attitude horizon scanner,

(2) modification of the MBC switch,

(3) modification of the camera clocks to change the picture interval in remote sequence from 0.5 minute to 1.0 minute (increasing the duration of a 32-picture remote sequence to 32 minutes), and

(4) removal of the IR-equipment dummy weights from the assembled spacecraft and disconnection of the north - indicator subsystem.

The assembly of the reconfigured spacecraft was completed on March 18, 1965, and on March 23, 1965, debugging was initiated. On April 11, after debugging had been completed, a functional check of all subsystems was performed with satisfactory results. On April 12 and 13, Spacecraft OT-1 was successfully subjected to thermal checks at +50 °C, 0 °C, and -5 °C in a Tenney chamber.

On April 15, 1965, an RF-interference check was successfully performed on Spacecraft OT-1, and the mechanical "tie-down" of spacecraft units was performed in the period from April 16 through 26.

c. **Calibration and Qualification of Spacecraft OT-1**

On April 27, 1965, rough balancing of Spacecraft OT-1 was completed. On April 28, an RF-interference check was performed at the RF tower. When the spacecraft was returned from the RF tower, it was observed that a resistor in camera clock No. 2 had been shorted because of insufficient clearance between a metal holding clip on the clock case and the case of a silicon-controlled rectifier. The clips on all three clocks, i.e., the two camera clocks and the QOMAC clock, were modified to provide greater clearance, and camera clock No. 2 was replaced with a flight-qualified spare.

On May 1, 1965, a full SEPET was successfully performed; and, on May 2, the field-of-view test on spacecraft OT-1 was completed with satisfactory results. On May 3, the spacecraft was installed in the thermal-vacuum chamber for the initiation of environmental testing.

On May 4, after the spacecraft temperature had been stabilized at +50 °C, it was observed that (1) the video level from camera No. 1 was 10 percent of the normal value, and (2) an exponential increase in the load-bus current occurred at a point about 80 percent through the playback of video from tape recorder No. 1. Both conditions were repeatable, and at +45 °C camera No. 1 video was at about 30 percent of the level at room temperature. At this temperature level, the exponential increase in the load-bus current did not occur.

Thermal-vacuum testing was, therefore, halted; and, on May 5, the spacecraft was removed from the thermal-vacuum chamber and camera No. 1 and the tape recorder power converter in camera system No. 1 were removed to permit an investigation of the problem area.

A transistor amplifier stage in the video amplifier section of camera No. 1 was found to be improperly biased. When the camera was operated at elevated temperatures, the transistor approached saturation, causing "clipping" of the video signal. The biasing resistor was changed in order to center-bias the amplifier and permit sufficient operating margin over the temperature range from -10 °C to +65 °C. Following this modification, the camera was successfully subjected to thermal tests in the Tenney chamber, and reinstalled on the spacecraft.

An examination of the tape recorder power converter indicated that one of the two 2N174A power transistors in the DC inverter had excessive collector-to-baseplate leakage. In order to provide a path for the collector to base leakage current for both 2N174A DC inverter transistors, a 68-ohm resistor was added to "tie down" the common base point to ground.

The same modification had been made to the tape recorder power converters on TIROS IX, but had not been made to those on spacecraft OT-1 since the duty cycle for the units on the latter was only approximately one-half that of the units on TIROS IX. Tests of the unit after the 68-ohm resistor was added failed to duplicate the condition. A qualified spare power converter was modified accordingly and mounted on the spacecraft in place of the unit which had been removed from side one. In addition, the 68-ohm resistor was also added to the side two power converter.

On May 6, the spacecraft was re-installed in the thermal-vacuum chamber for the resumption of environmental tests. The 36-hour test at +50 °C was initiated on May 7 and completed on May 10, with satisfactory results. On May 12, the 48-hour test at 0 °C was successfully completed, marking the end of the first phase of thermal-vacuum testing.

On May 13, the spacecraft was removed from the thermal-vacuum chamber and the flight-model battery pack was installed. On May 14, a SEPET was satisfactorily performed on spacecraft OT-1.

On May 15, random vibration testing in three mutually perpendicular planes was performed with satisfactory results (at the test levels detailed earlier in this discussion). On May 17, a SEPET was satisfactorily performed on the spacecraft.

On May 18, spacecraft OT-1 was installed in the thermal-vacuum chamber for the second phase of thermal-vacuum testing. On May 19, the 24-hour test at +35 °C was successfully performed. On May 20, testing at 0 °C was initiated. However, on May 22, after 48 hours of testing at this level had been accomplished, testing was halted because of an anomoly in the operation of TV transmitter No. 2, and the spacecraft was removed from the thermal-vacuum chamber.

Carrier deviation with TV transmitter No. 2 had been observed to take a sudden increase of approximately 30 percent approximately 15 seconds after the start of playback. Further tests showed that the length of time from initiation of transmitter B+ until the deviation increased varied with each playback from less than 1 second to as much as 18 seconds. A check of the transmitter showed that (1) the carrier frequency was 30-kc low until the time that the deviation increased, and (2) the carrier frequency shift also occurred when the camera system was operated in the direct mode. TV transmitter No. 1 was tested and showed similar results, along with four flight-qualified spare transmitters. The anomaly was traced to a cold cathode voltage regulator tube in the transmitter. This tube was found to have an inconsistent firing time which was, to a great extent, independent of starting voltage, temperature, and frequency of operation.

Further tests performed by means of interrogations of TIROS VII and VIII indicated the condition was, in fact, normal and had no adverse effect on picture transmission. Based on these findings, the environmental committee decided that the qualification cycle should be resumed.

Spacecraft OT-1 was placed in the thermal-vacuum chamber on May 25, and environmental testing was reinitiated on May 26. On May 27, the spacecraft completed 36 hours of testing at 0 °C with satisfactory results. This marked the successful completion of environmental tests on the spacecraft. Table II-13 presents a summary of thermal-vacuum tests on spacecraft OT-1, while Table II-14 presents a chronological summary of all environmental testing of spacecraft OT-1.

On May 28, a full SEPET was performed on spacecraft OT-1 with satisfactory results.

On June 1, 1965, the magnetic dipole measurements on the spacecraft were made, utilizing the magnetometer and the spherical dipole testing machine, and a permanent magnet was added to cancel out the spacecraft's residual dipole.

TABLE II-13. SUMMARY OF SPACECRAFT OT-1 THERMAL-VACUUM TESTS

Temperature Level (Degrees C)	Test Duration (hours)				Total Test Duration (hours)	Specification Requirements (hours)
	Test No. 1	Test No. 2	Test No. 3	Test No. 4		
+50	7.5	72	--	--	79.5	72
+35	--	--	24	--	24.0	24
0	--	48	54	36	138.0	120

TABLE II-14. CHRONOLOGICAL LISTING OF SPACECRAFT OT-1 ENVIRONMENTAL TESTS

Test Initiation Date (1965)	Environmental Test
May 4	Thermal-Vacuum Testing at 50° C
May 7	Thermal-Vacuum Testing at 50° C
May 10	Thermal-Vacuum Testing at 0° C
	(1st Phase of Thermal-Vacuum Testing Completed on May 12)
May 15	Vibration Testing, Random Vibration in Three Mutually Perpendicular Axes
May 18	Thermal-Vacuum Testing at 35° C
May 20	Thermal-Vacuum Testing at 0° C
May 25	Thermal-Vacuum Testing at 0° C
	(Environmental Testing Completed on May 27)

On June 2, camera brightness and final alignment tests were performed; and, on June 3, the camera field-of-view tests were completed.

Final alignment of the attitude horizon scanner and the solar-aspect indicator were performed on June 3, 1965.

On June 4, the final balancing of the spacecraft and the center-of-gravity, final-weight, and moment-of-inertia measurements were performed.

On June 5 and 7, respectively, an abbreviated SEPET and a Task 4 test were successfully completed. On June 8, the spacecraft was subjected to a final inspection.

On June 9, the Environmental Test Committee met and agreed that spacecraft OT-1 had successfully passed all environmental test requirements. The spin-up rockets were added to the spacecraft on June 11, and the spacecraft was installed in a shipping container.

On June 14, 1965, at NASA direction, spacecraft OT-1 was shipped to the launch site at the Eastern Test Range.

PART III. GROUND STATIONS

PART III. GROUND STATIONS

SECTION I. DESIGN OF GROUND-STATION COMPONENTS

A. INTRODUCTION

In general, ground operations for the TIROS X satellite system consist of (1) tracking the position of the satellite; (2) commanding the satellite instrumentation to perform specific functions in a given order; and (3) receiving, storing, and processing data received from the satellite. These operations are performed and coordinated by a ground complex which includes the following: (1) two primary Command and Data Acquisition (CDA) stations, one located at Fairbanks, Alaska and the other at Wallops Island, Virginia; (2) a secondary CDA station located at the AED Space Center near Princeton, New Jersey; and (3) selected stations of the NASA Minitrack Network, including an auxiliary station (for the initiation of clock-start) at the Minitrack station in Santiago, Chile. In addition, a checkout (Go, No-Go) station, installed at the launch site at the Eastern Test Range, Cape Kennedy, Florida, is included in the ground complex for performing prelaunch checkout of the satellite.* The CDA stations and the Go, No-Go equipment were used in previous TIROS programs.

The primary purposes of the CDA ground stations are as follows:

(1) To transmit radio signals to the satellite for programming its operation and data transmission.

(2) To receive signals carrying the television, attitude, and telemetry data from the satellite.

(3) To extract the television, attitude, and telemetry data from the carrier signals.

(4) To record and reproduce, in permanent form, the received data and to provide a means of identifying that date.

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*The ground complex includes other facilities involved in satellite command programming and in data processing. However, the facilities mentioned here are those that are in direct communication with the satellite.

(5) To relay the transmitted attitude and telemetry data, along with the station's status reports, to the NASA TIROS Technical Control Center (TTCC) located at the Goddard Space Flight Center, Greenbelt, Maryland.

The precision tracking capabilities of the selected stations of the Minitrack Network are used to permit an accurate determination of the satellite's orbital parameters.

B. FUNCTIONAL DESCRIPTION

The components of the CDA stations are divided into three functional groups: namely, the satellite command and control equipment; the data-receiving components; and the data-processing, display, and recording components.

The command and control equipment controls the satellite functions by means of an amplitude-modulated command transmitter. Audio control tones, each tone representing a different command function, are used for modulating the command transmitter. Three modes of operation are provided for commanding the satellite; namely, manual-operate, manual-start, and automatic. In the manual-operate mode, which is normally used only during testing, all satellite commands are initiated manually. In the manual-start mode, only the program sequences are started manually; once the sequence starts, the commands within the sequence are initiated automatically. In the automatic mode, all sequences and all commands within a sequence are transmitted without manual intervention and at preselected times. The times are synchronized with the time signals transmitted by WWV.

The TV-picture receiving circuit consists of two receivers that are connected in polarization diversity to minimize signal fading due to satellite spin and attitude. The telemetry receiving circuit consists of four receivers. Two of the receivers are tuned to the upper telemetry frequency, and two are tuned to the lower telemetry frequency. Two receivers, one of each pair, are connected in polarization diversity.

Each TV picture received is displayed on a kinescope which is mounted in the display unit. A panel framing the kinescope is equipped with (1) a clock that provides a real-time indication and (2) legends and numbers that are illuminated to indicate the mode (direct camera or tape playback), camera source (1 or 2), frame number, and orbit number for each TV picture. Mode and camera-source information are derived from outputs of the command and control equipment. The frame number is generated by a binary counter, which is stepped by the vertical-sync pulse of each TV picture received. The frame number, consisting of six binary bits, and the mode and camera-source data, consisting of three binary bits, are stored in a shift register from which a serial output and a parallel set of outputs are taken. The parallel outputs control read-out lamps which are photographed along with the kinescope display; the serial output keys tone oscillators whose outputs are recorded on magnetic tape to indicate camera source and frame information. The orbit number is displayed by means of illuminated, manually set dials. A camera, mounted on the display unit, is used to photograph both the picture displayed on the kinescope and the associated identification data.

The recording devices used at the primary TIROS X ground stations are (1) two Ampex Model FR100K, seven-channel, tape recorders; (2) an Esterline Angus Model AW events recorder; and (3) a Sanborn, two-channel, paper-chart recorder.

The tape recorders record the TV pictures received from the satellite and the related identification information. Each recorder operates at a speed of 60-inches-per-second, using tape 1/2 inch wide. The tape recorders are remotely controlled by the command and control equipment to start automatically at the beginning of each ground-to-satellite contact.

The events recorder provides a real-time recording of the initiation of both the various satellite commands and the other vital ground-system operations.

The two beacon subcarrier signals containing telemetry information are sent from the beacon receivers to the Sanborn recorder, and the beacon No. 2 signal is applied through the Frequency Shifter Unit (FSU) before being mixed with the beacon No. 1 signal for transmission over the SCAMA line.

C. PHYSICAL CONFIGURATION

Ground-station components are mounted on roll-out assemblies located in vertical racks which have an overall height of 65-3/8 inches. Each roll-out assembly consists of a front panel and two vertically mounted chasses; the vertical chasses are arranged so that the tubes face inward and the wiring faces outward. This combination of chassis mounting and roll-out slides facilitates maintenance and trouble-shooting. The vertical mounting of the chasses also provides a chimney effect which materially assists in component cooling. (The equipment racks at Wallops Island are mounted in the telemetry building.)

D. GROUP FUNCTIONAL DESCRIPTIONS

1. Introduction

Since the TIROS X CDA ground stations are essentially identical to those used for the TIROS IV, V, VI, and VII programs, the following functional descriptions are limited to block-diagram discussions. Detailed circuit descriptions are contained in the TIROS I through VII final reports.

2. Satellite Command and Control Equipment

a. Introduction

The satellite command and control equipment provides a reliable means of turning on the command transmitter, programming the antenna to follow the predicted path of the satellite, turning on the TV and data recorders, initiating the transmission of control tones to the satellite, and turning off the equipment at the end of a satellite-to-ground contact. Figure III-1 is a functional block diagram of this equipment.

Functionally, the equipment consists of two separate programming circuits, one program selector, and one timing circuit. The timing circuit, consisting of the master clock, the WWV receiver, the WWV comparator, and the frequency standard, is common to both programming circuits. The positioning of switches and control relays on the program selector determines which programming circuit is used for a specific satellite pass.

The ground-station components of the satellite command and control equipment provide for the selection of three modes of operation; namely, automatic, manual-start, and manual-operate. Briefly, the system operation during these operational modes is as follows:

(1) Automatic: During this mode, the program is set up in advance on the control equipment. Each program sequence starts in response to an alarm signal from the master clock and proceeds to its conclusion without the aid of an operator.

(2) Manual-Start: During this mode, the program is also set up in advance. The only difference between manual-start and automatic is the use of pushbutton controls in place of the master clock for the initiation of the alarm signals.

(3) Manual-Operate: During this mode, the program is not set up in advance. Instead, the program sequences are initiated by the use of pushbutton controls which are further used to carry the sequence through to completion.

The use of two, separate, programming circuits permits two complete programs (A and B) to be set up in advance. The program sequences used in TIROS X and the alarms controlling these sequences are as follows:

(1) Direct Camera Sequence I: Direct Camera Sequence I (DCS-I) is controlled by alarm number 1. This program sequence is used when the TV pictures are to be taken while the satellite is in range of a ground station.

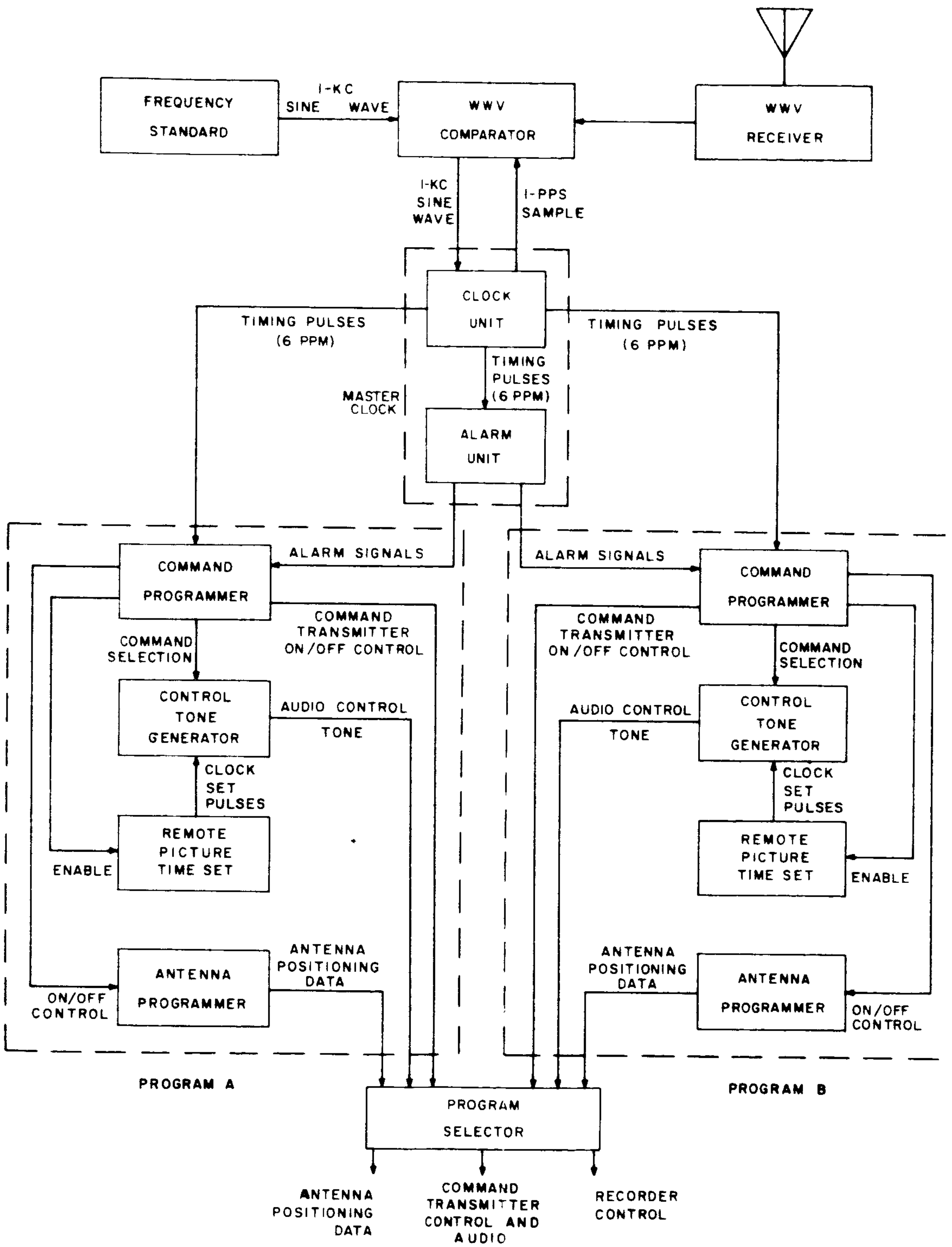


Figure III-1. Command and Control Subsystem, Block Diagram

In DCS-I, the pictures are transmitted directly to ground, bypassing the satellite's tape recorders. Either one of the satellite's TV cameras can be commanded to take pictures at 30-second intervals. Picture-taking commands can be alternated from one camera to the other at intervals of 30 seconds. The length of the sequence can be varied between 0.5 and 8.0 minutes in 0.5-minute increments.

(2) Playback and Clock-Set Sequence: The playback and clock-set sequence is initiated by alarm number 2 and includes the following: (1) commanding the satellite to read out pictures which have been recorded on the satellite's tape recorders since the last ground-to-satellite contact, and (2) sending set pulses to the satellite clocks and, at the conclusion of the set pulses, a start pulse for both clocks.

(3) Direct Camera Sequence II: Whenever Direct Camera Sequence II (DCS-II) is programmed, it follows directly after the playback sequence. In addition to providing for the same program variations as DCS-I, this sequence provides for the automatic transmission of a predicted number of attitude-control pulses.

(4) Clock-Start (Santiago, Chile). A clock-start capacity is provided at the NASA Minitrack station at Santiago, Chile. This capability can be used to start the clocks only during a direct-camera sequence.

In addition to the functions listed for each sequence, any of the program sequences can include the sending of manually initiated "Fire Rockets" commands, as well as manually initiated commands for stepping the satellite's MBC switch.

b. Timing Circuits

The timing circuits consist of the alarm unit, the master clock, the WWV receiver, the WWV comparator, and the frequency standard. These circuits provide the time standard for both ground station and satellite, and generate the alarm signals which initiate the various sequences of an automatic program.

c. Control-Tone Generator

This unit generates the audio-control tones used to represent the various command functions.

d. Remote-Picture Time-Set Unit

This unit generates, during the playback sequence, the pulses required to set each of the satellite clocks.

e. Antenna Programmer

The antenna programmer utilizes a linear interpolation of ephemeris data, taken at 1-minute intervals, to aim the tracking antenna at the point where the satellite is expected to appear over the horizon. In addition, the programmer can be used to direct the tracking antenna to follow the predicted path of the satellite. Use of the antenna programmer ensures faster antenna lock-on by elimination of the need for horizon scanning at the start of each satellite pass. The antenna programmer also prevents loss of data when the auto-tracking contact is prematurely interrupted.

f. Program-Selector and Power-Control Unit

This unit provides for (1) selection of either Program A or Program B for transmission to the satellite,and (2) control of both filament and plate voltages to the two programmer circuits.

g. Relay Power Supply

This unit provides the 24-volt power required for the energization of the relays in the ground-station command and control equipment; it also provides for distribution of the 115-volt, 60-cps, a-c power to these components.

h. Command Transmitter and Remote-Control Panel

The Collins Model 242F-2, 200-watt, amplitude-modulated, VHF transmitters are used for transmission of the command signals. Though two of these transmitters are located at each ground station, only one can be used at any given time. The transmitters are operated remotely, due to the requirement that they be located within 100 feet of the transmitting antenna to avoid excessive power loss in the RF cabling.

A remotely operated coaxial switch is used to switch the antenna from one transmitter to the other. A low-pass filter is located at the output of the switch for the reduction of the spurious radiation found above the command frequency.

Each command transmitter is equipped with an RF detector used for alignment purposes. The output of the detector, a d-c current, is indicative of the transmitter power output. The detector output also contains the detected command tones, which are amplified to drive a loudspeaker in the transmitter control panel; the loudspeaker permits monitoring of the outgoing command tones.

i. Command Programmer

The command programmer provides the means for setting-up and storing the desired satellite program. When an alarm signal is received from the alarm unit, the command programmer supplies related portions of the stored program to the control-tone generator, the antenna programmer, the remote-picture time-set, the tape recorders, and the command transmitter. Two command programmers are installed at each CDA station.

The programmer permits presetting of program sequences and provides for automatic read-out of these sequences at preselected, electrically computed times. The design of the programmer reduces the possibility of human error by affording an opportunity to check preset programs and by minimizing the need for human operations during a satellite-to-ground contact.

Other design features include provisions for manually starting and controlling each of the program sequences. These features are not intended to be used under normal conditions; they are included to provide control of the satellite during special or emergency programming.

j. Clock-Set-Pulse Demodulator

This unit is used in conjunction with back-up Berkeley counters and provides an accurate count of the set pulses sent to the satellite clocks. The input for the clock-set-pulse demodulator is received from the RF-detector circuit of the command transmitter. Circuits within the demodulator separate the detected clock-set pulses from the remainder of the detected command tones and then apply these pulses to the Berkeley counters.

k. Clock-Start Timer

The clock-starting equipment installed at the NASA Minitrack station at Santiago, Chile, permits pictures to be taken over otherwise unobtainable areas of the earth. The equipment includes an appropriate command transmitter and the necessary timer. The clock-start timer is used to generate address tones, direct-camera tones, and start-clock tones.

3. Data Receiving Components

a. Introduction

The TIROS X data receiving equipment at the ground stations comprises the TV receivers, the beacon and telemetry receivers, and the TV diversity combiner. Except for the diversity combiner, the equipment selected for data receiving was in either military or commercial use at the start of the TIROS programs.

b. **TV Receiving Circuits**

The TV receiving circuits (shown in the block diagram in Figure III-2) consist of the following: (1) two bandpass filters; (2) two distribution amplifiers; (3) two TV receivers, tuned to 235 Mc; and (4) a diversity combiner. The horizontally and vertically polarized outputs of the tracking antenna are applied through bandpass filters before being applied to the respective TV receivers. The bandpass filters are used to prevent interference from the command transmitter.

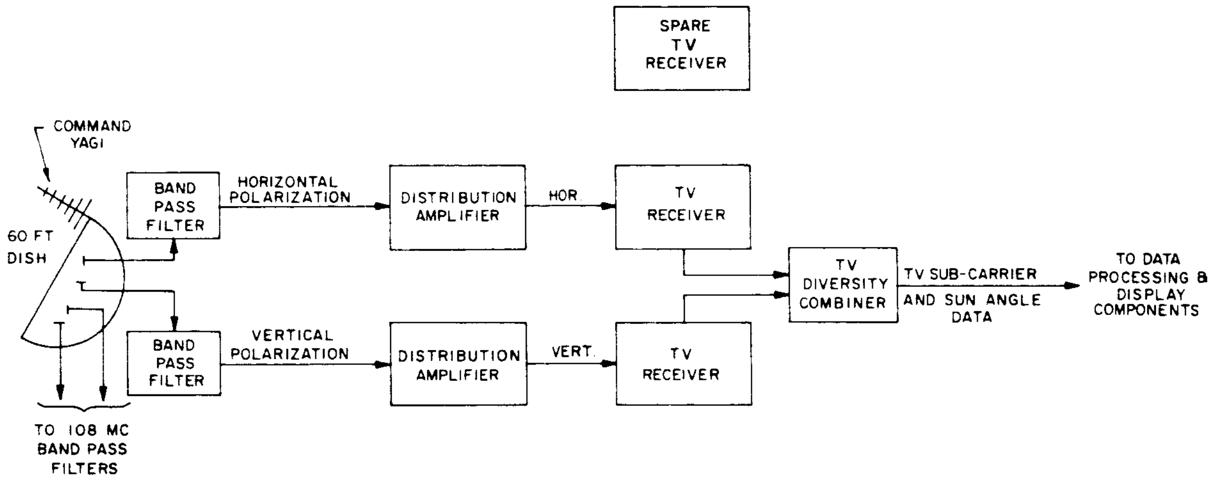


Figure III-2. TV Receiving Circuits, Block Diagram

The outputs of the two TV receivers are applied to the diversity combiner. The diversity combiner either selects the stronger of its two signal inputs or combines the two inputs for application to the succeeding stages of the TV receiving circuits.

c. **Beacon and Telemetry Receiving Circuits**

The beacon and telemetry receiving circuits (shown in the block diagram in Figure III-3) consist of (1) two bandpass filters, (2) two preamplifiers, (3) two frequency converters (136 to 30 Mc), (4) two multicouplers, (5) four R-390A receivers, (6) a telemetry receiver control, and (7) a two-channel Sanborn recorder.

All of this equipment is similar to that used in the TIROS VII program.

For the reception of both beacon and telemetry channels (136.23 and 136.92 Mc), two of the receivers are tuned to 30.23 Mc and the other two to 30.92 Mc. One receiver from each of the two frequency groups receives its signal from the horizontally polarized feed of the antenna system; the other receiver receives its signal from the vertically polarized feed. The two receivers of each frequency group are connected together in polarization-diversity combination.

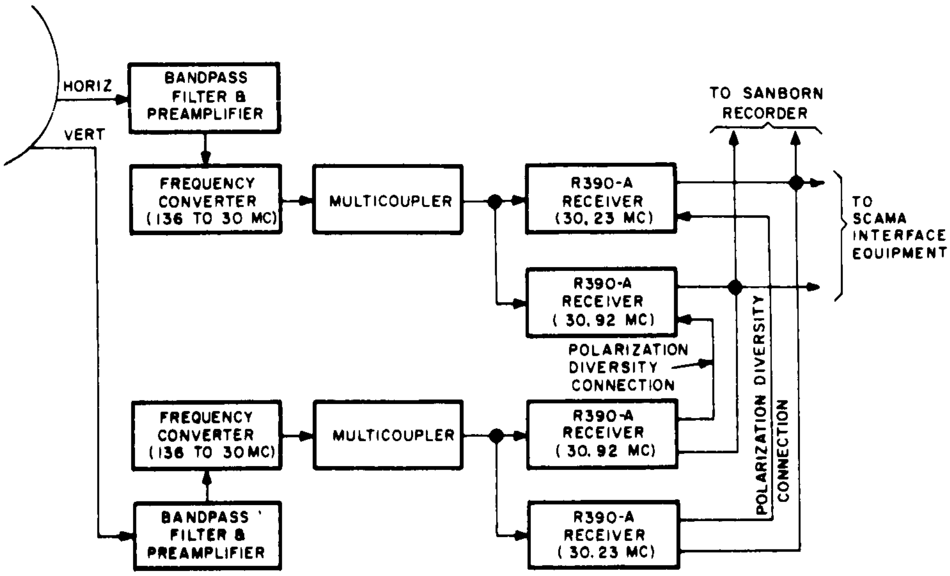


Figure III-3. Beacon and Telemetry Receiving Circuits, Block Diagram

The 1300-cps, frequency-modulated, subcarrier-output signals of the receivers are applied to the Beacon Data Control Unit (BDCU). This unit permits the selection of the desired group of receivers for connection to a particular channel (A or B) of the Sanborn recorder.

The subcarrier signals are demodulated and resulting telemetry information recorded on the Sanborn recorder. The telemetry-receiver control also indicates the present level of AGC and permits monitoring of the receiver audio outputs.

4. Data-Processing, Display, and Recording Components

a. Introduction

The data processing, display, and recording components used for the TIROS X program are the same as those used for TIROS III. Figure III-4 is a block diagram of this equipment. These components provide for (1) the demodulation of the TV signals from the satellite,and (2) the recording of the resultant TV pictures on film and magnetic tape. These components also provide identification and orientation information (frame number, orbit number, satellite-camera identification, picture-taking sequence and real time) for each picture. A playback system, included to facilitate meteorological interpretation of the pictures received from the satellite, permits the generation of both positive and negative duplicate pictures.

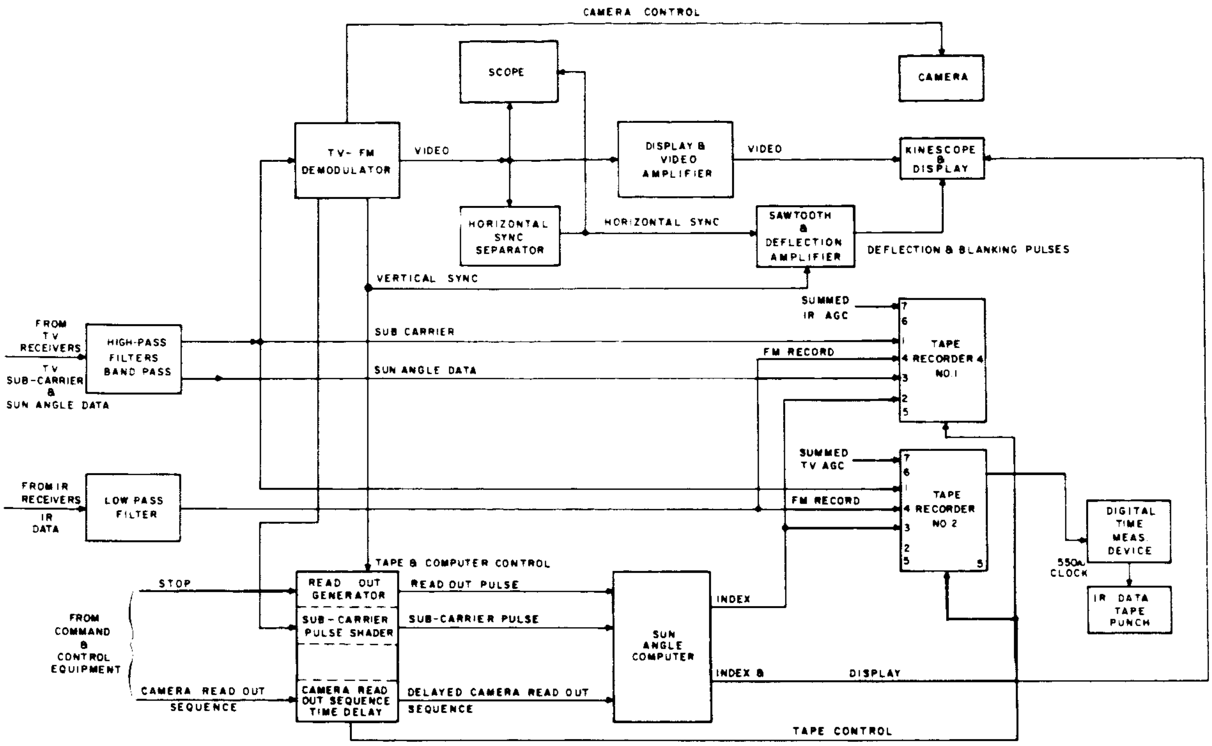


Figure III-4. Data Processing and Display Components, Block Diagram

The input signal to the data-processing, display, and recording components is the frequency-multiplexed TV subcarrier from the TV diversity combiner.

In addition to providing demodulation of the TV subcarrier, the TV-FM demodulator generates the vertical synchronizing signal. The video output of this unit is applied to the horizontal separator, to a monitor scope, and to the display and video amplifier.

The outputs of the horizontal separator and the vertical sync generator are applied to the sawtooth and deflection unit. In turn, the sawtooth and deflection unit provides the vertical and horizontal deflection currents for the deflection yoke of the kinescope.

The TV subcarrier is recorded directly on a magnetic tape.

b. **Display and Video Amplifier**

The function of the display and video amplifier is to provide final amplification of the TV video and to present the TV picture on a display panel. This display panel includes indicator lamps whose operation is controlled by inputs from the other data processing and display components to provide the picture identification data. A camera mount is used to permit taking both 35-mm and polaroid pictures simultaneously. The mount is positioned so that the real-time indicator is in the field-of-view of the 35-mm camera.

c. Sawtooth and Deflection Amplifier

This unit supplies both the horizontal and vertical deflection waveforms and the horizontal and vertical blanking pulses to the kinescope.

d. Horizontal Sync Separator

This unit provides synchronizing pulses to the horizontal sawtooth deflection circuitry. These pulses are in phase and locked to the video horizontal rate. The operation of the sync separator is unique in that it provides these synchronizing pulses in response to video signals which are random in nature, that is, the horizontal rate or horizontal signal is noncoherent from frame to frame even though the frequency within each frame is the same.

e. TV-FM Demodulator

The input to the TV-FM demodulator is the modulated TV subcarrier. This unit separates the video signal from the subcarrier and provides several outputs, one of which is the video signal for the kinescope video circuits. Other outputs are (1) a control signal for the camera shutter, (2) a vertical synchronizing signal to the kinescope deflection circuits, and (3) a video subcarrier pulse to the tape and computer control.

A pulse-counting type of demodulator was used because of its stability and its linear operating characteristics. This type of demodulator has been used successfully on commercial video-tape recorders in which the frequency-spectrum relationships of the video signal, the subcarrier, and their modulation products are the same as those in the TIROS TV-picture subsystem.

f. Tape and Computer Control

On the TIROS X program, this unit provides the source signals for frame identification and acts as a central control for the two tape recorders.

g. Sun-Angle Computer Control

Though sun-angle data is not produced by the north-indicator subsystem on TIROS X, the sun-angle computer control is part of the operating set-up and provides index data (consisting of the frame number, the camera number, and the picture mode) directly to the monitor for immediate display with the TV pictures. It also provides this index data to the tape recorders for the recorded pictures. During playback of the instrumentation recorder, the sun-angle computer provides for either (1) sequential display of all video frames or (2) selection of a single video frame for display.

h. Monitor Control

The monitor control provides switches and relays for controlling the inputs of the data-processing and display components, and for turning on and turning off the 32-volt power supply, the three 300-volt power supplies, the high-voltage power supply, and the sun-angle computer power supply.

i. Calibrator

This unit generates a video test pattern and an 85-kc, frequency-modulated sub-carrier, which are used to test the TV subsystem. Simulated sun-angle bursts are also generated and used to test the sun-angle computer.

j. Digital Time-Measuring Device

The functions of the Digital Time-Measuring Device (DTMD) are (1) to measure and identify the time interval between sequential trigger pulses and (2) to drive a tape punch which presents the time interval in teletype code on paper tape. This unit has an automatic start and stop capability.

k. Attitude-Pulse Selector

The attitude-pulse selector receives the "raw" attitude signals from the Sanborn recorder, selects and re-forms the valid attitude pulses while blocking any spurious pulses, and applies the valid attitude pulses to the DTMD.

l. Quick-Look Demodulator

(Not used on TIROS X)

m. Infrared Buffer

(Not used on TIROS X)

n. Tape Recorders

Seven-channel, Ampex tape recorders (Series FR-100A) have been used on TIROS programs since TIROS IV. These recorders are of standard commercial design and are used to record the received video. (For TIROS satellites containing IR equipment, these recorders also record the composite IR signal, the IR events signal, and the summed AGC voltages of the IR receivers.)

o. Events Recorder

A 20-channel, Esterline-Angus events recorder (Model AW) is used to provide ON-OFF indications versus time, on a paper chart. These indications provide (1) a direct, real-time record of the command program, for the purpose of checking either equipment malfunctions or operator errors prior to a pass and (2) a permanent record of the commands sent during a pass. The recorder is equipped with both manual and automatic start features. Manual start is used when trouble-shooting and maintenance operations are necessary at the ground stations. Automatic operation is used during the normal operation of the equipment.

p. Telemetry Recorder

A two-channel, Sanborn paper-chart recorder is used to provide a permanent record of the time-referenced output from the telemetry receivers. The data recorded includes "housekeeping" telemetry, solar-aspect indicator data, MBC switch position data, etc.

5. CDA - SCAMA Interface Equipment

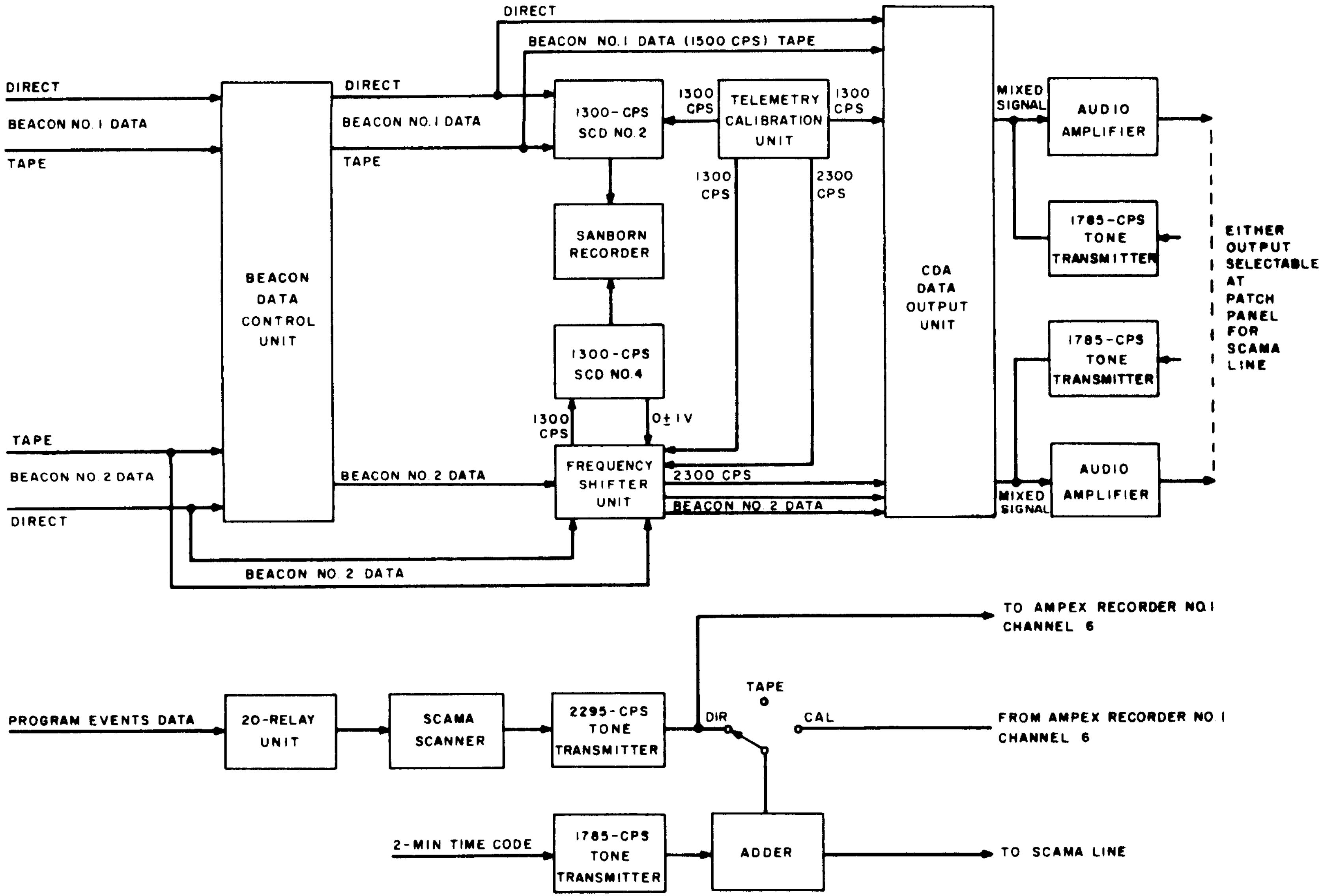
a. General

Figure III-5 shows the CDA-SCAMA interface equipment as used for the TIROS X program. The Frequency Shifter Unit (FSU) was added for this program to shift the beacon No. 2 subcarrier from 1300 cps to 2300 cps, permitting it to be mixed in the CDA Data Output Unit (CDOU) with the beacon No. 1 subcarrier, which is also at 1300 cps, for transmission over a SCAMA line to TTCC.

Test signals, including a series of tuning-fork-controlled tones from the telemetry calibrator, can also be placed on the SCAMA lines for calibration or checking purposes.

For the TIROS X operations, the Beacon Data Control Unit (BDCU) functions as a routing device through which the beacon subcarriers are applied. The beacon No. 1 subcarrier is applied to the BDCU and sent directly to the CDOU and, at the same time, to subcarrier discriminator No. 2, where it is demodulated for recording on the Sanborn recorder.

The beacon No. 2 subcarrier is simultaneously applied (1) through the BDCU to the FSU, and (2) directly to the FSU, before being supplied to the CDOU. (The handling of the beacon No. 2 subcarrier is detailed in the following description of the FSU.)



Figures III-5. CDA-SCAMA Interface Equipment

In the case of beacon No. 1 data, selection of the data to be transmitted is accomplished at the CDOU by means of an input switch which permits the selection of

(1) beacon No. 1 data from beacon receiver No. 1,

(2) beacon No. 1 data from the magnetic tape recorder No. 1, or

(3) a 1300-cps calibration signal from the telemetry calibration unit.

In the case of beacon No. 2 data, this switching function is handled in the FSU.

Bandpass filters in the CDOU permit the selection of the beacon subcarriers. A resistive network provides proper impedance matching for the outputs of the bandpass filters, and allows the outputs of the selected pair of filters to be summed.

Since the beacon-subcarrier signals are attenuated by the bandpass filters and matching-impedance networks, the combined filter outputs are amplified to attain a 0-dbm level for feeding the input to the SCAMA circuits. For this purpose, the unused audio amplifier section on each of the two beacon receivers is connected as a buffer amplifier between the output of each of the two pairs of filters, and the input to the SCAMA circuits. These amplifiers also provide a gain control for setting proper output level and a meter for monitoring that output level.

A QT-30 tone transmitter is also provided, for the output of each pair of filters, to FSK-modulate a 1785-cps subcarrier with a time code (2-pps bit rate). This signal is combined with the beacon signals at the output of the CDOU.

Program events data signals are taken from the Esterline-Angus events recorder and applied to a bank of 20 relays which, in turn, provides contact closures to a scanner-transmitter unit. This unit multiplexes the relay closure data into a signal which is FSK-modulated onto a 2295-cps subcarrier by a QT-30 tone transmitter for transmission over a SCAMA line.

b. FSU

The FSU comprises a 2300-cps voltage-controlled oscillator (VCO), a power supply, and associated circuitry. As noted earlier, the beacon No. 2 subcarrier is applied both through the BDOU to the FSU, and directly to the FSU.

Figure III-6 is a diagram of the switching arrangement in the FSU. The FSU is equipped with two switches: S1 and S2. S1 provides for switching between the SHIFT mode, i.e., TIROS X operations, and the NORMAL mode, i.e., TIROS IX operations.

When S1 is in the NORMAL mode, the beacon No. 2 signal applied through the BDCU bypasses the VCO and is sent to SCD No. 4 and thence to the Sanborn recorder.

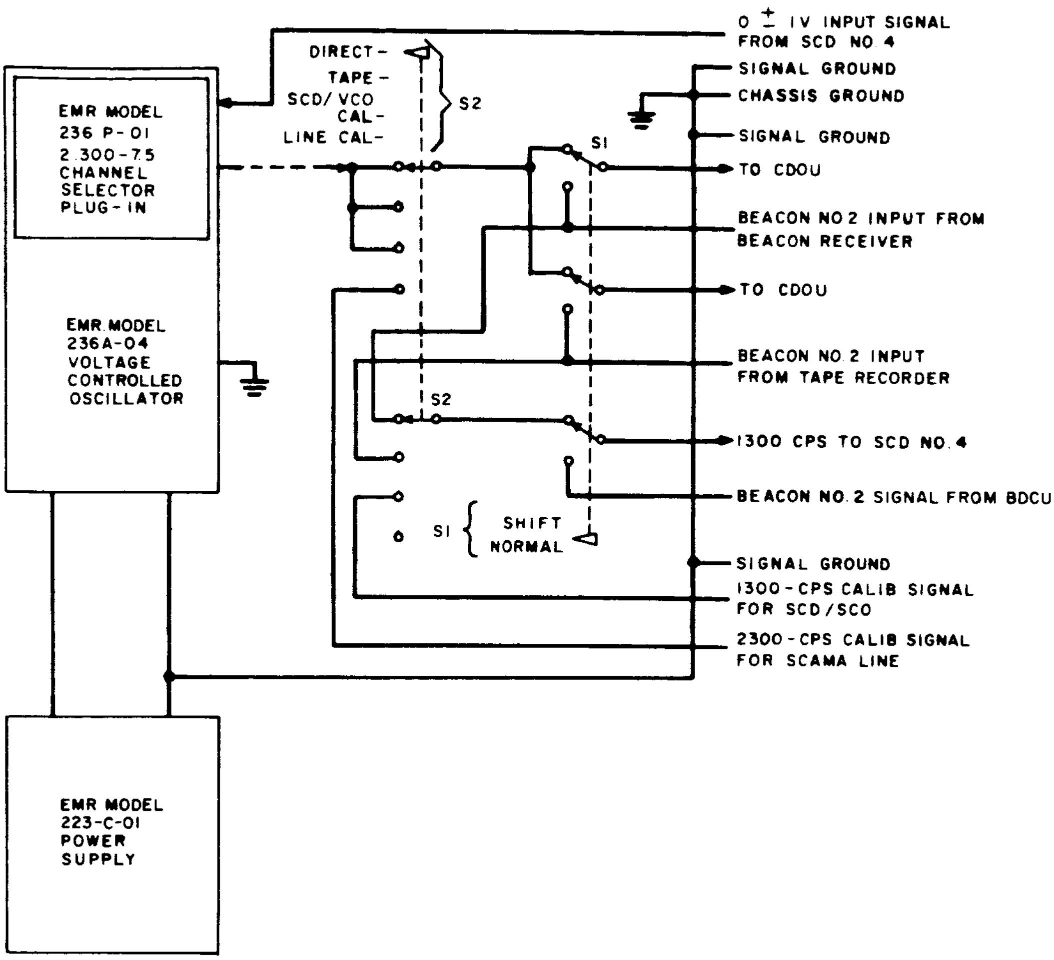


Figure III-6. Switching Arrangement for the Frequency Shifter Unit

The beacon No. 2 signal simultaneously applied directly to the FSU is also unaffected by the VCO and is routed to the CDOU.

When S1 is in the SHIFT mode, the beacon No. 2 signal applied through the BDCU is cut off in the FSU. The signal simultaneously applied directly to the FSU is routed to SCD No. 4; the resultant dc signal is sent to the Sanborn recorder and, at the same time, to the VCO, where it is converted to a 2300-cps signal and from which it is sent to the CDOU.

When S1 is in the SHIFT mode, switch S2 permits the selection of the following direct inputs to the FSU:

(1) beacon No. 2 data from beacon receiver No. 2,

(2) beacon No. 2 data from magnetic tape recorder No. 1,

(3) a 1300-cps calibration signal from the telemetry calibrator, or

(4) a 2300-cps calibration signal from the telemetry calibrator.

The beacon No. 2 data, whether from the receiver or from the magnetic tape recorder, is applied first through SCD No. 4 (where the signal is demodulated), then simultaneously (1) to the Sanborn recorder and (2) through the VCO (where the d-c signal is changed to a 2300-cps signal) to the CDOU.

The 1300-cps calibrator signal is applied to the SCD and, as a demodulated signal, to the VCO to permit calibration of these units.

The 2300-cps calibration signal is immediately routed out of the FSU, bypassing the VCO, and is applied to the CDOU for calibration of the SCAMA circuits.

PART IV. FIELD OPERATIONS

PART IV. FIELD OPERATIONS

SECTION I. GROUND STATIONS

A. GENERAL

Field operations in tracking and commanding the TIROS X satellite and in receiving, storing, and processing retrieved data were similar to those performed for preceding TIROS satellites. The two primary Command and Data Acquisition stations on the TIROS X program were located at Wallops Island, Virginia, and Fairbanks, Alaska. As on previous TIROS programs, the Princeton ground station at the AED Space Center, Princeton, New Jersey, served as a back-up facility. Operations at all TIROS CDA stations were satisfactory for the TIROS X launch.

B. TIROS TECHNICAL CONTROL CENTER (TTCC)

AED personnel assigned to TTCC for the launch assisted and advised TTCC operating personnel in all operational-phase activities as required. In addition, technical support of equipment was provided, with operational support concentrated in the area of attitude control during the turn-around maneuver.

C. WALLOPS ISLAND GROUND STATION

Spacecraft programming at the Wallops Island facility (which was being used to interrogate TIROS VII, TIROS VIII, and TIROS IX) was reduced on June 19 in order to permit the ground station to participate in TIROS X pre-launch exercises.

On June 28, it was necessary to remove the General Bronze 235-Mc antenna from service due to difficulty with the elevation-drive system. However, the difficulty was soon corrected, and the antenna was returned to service later the same day. During the week prior to launch, some problems had been experienced with high-beam reception from the Kennedy 136-Mc antenna. The trouble was attributed to the coaxial switch, which was replaced on June 30.

On June 28, the installation of the telemetry-translator equipment for TIROS X was completed, and the unit was test-operated. The station was then considered in readiness for the TIROS X launch.

D. FAIRBANKS GROUND STATION

The programming effort on TIROS VII, VIII, and IX was reduced at the Fairbanks, Alaska, ground station on June 19 in order to permit this station to participate in the TIROS X pre-launch exercises.

On June 25, the installation of the telemetry-translator equipment for TIROS X was completed and successfully test-operated.

All operations at the Fairbanks station were conducted on a satisfactory basis. The high-power transmitters at this station were used to command the successful spin-down of TIROS X on orbit 002 and, thereafter, to command various phases of the satellite turn-around maneuver and checkout programs.

E. PRINCETON GROUND STATION

The Princeton ground station, located at the AED facility near Princeton, New Jersey, was placed in readiness to support the TIROS X launch on June 29, 1965. Mechanical difficulty with the antenna system at the Princeton station had been experienced late in June. However, the facility was again operational on June 28.

SECTION II. LAUNCH OPERATIONS AT CAPE KENNEDY LAUNCH SITE

Spacecraft OT-1 was shipped, fully qualified, to the Eastern Test Range (ETR) on June 14 along with the RF model and the spacecraft's associated handling and test equipment; the Go, No-Go van had been shipped to the ETR on June 11. Spacecraft OT-1 and the RF model were received at the ETR on June 15.

Shortly thereafter, the RF model was checked out with satisfactory results, and spacecraft OT-1 was subjected to a detailed Go, No-Go test. All phases of this test were completed with good results, except for the check of the MBC coil, which indicated the coil to be "open". The hat assembly, containing the MBC coil, was removed from the spacecraft, and an investigation showed one of the coil ends to be "open" at the point where it connected to the terminal board located on the coil frame.

On June 16, the coil connection was repaired, and it was inspected and approved by AED and NASA representatives. On June 17, a NASA quality-control engineer inspected and approved the repaired connection, and the repaired area was repotted. The hat assembly was then reinstalled on the spacecraft baseplate.

On June 17, the RF model was mated to the inert third stage of the launch vehicle. On the following day, the RF model and the inert third stage were taken to Launch Pad 17B for mating to the second stage of the launch vehicle. In addition, the MBC coil on spacecraft OT-1 was checked out, and a complete Task 4 check was successfully performed. On June 19, the launch-vehicle fairing was installed and an all-systems RF-interference test was conducted using the RF model, both with the gantry around the vehicle and with the gantry away from the vehicle. During these checks, the spacecraft beacon and TV transmitter were observed to interfere with the third-stage telemetry package.

On June 21, an on-stand optical check, a full Task 4 check, and an abbreviated Task 4 check were successfully performed on the spacecraft. On June 22, another Task 4 check was successfully performed. In addition, the RF model was remated to the inert third stage for an RF-interference check with the third-stage telemetry package (which had been modified since the tests made on June 19). The next day, a Task 4 check was performed on the spacecraft, and the third-stage telemetry package was installed on the third stage of the launch vehicle.

On June 24, another Task 4 check was successfully performed and the spacecraft was delivered to the Douglas Aircraft Company (DAC). On June 26, spacecraft OT-1 was mated to the third stage of the launch vehicle and the combined assembly was balanced. (This operation occurred 5 days after the originally scheduled time; however, the postponement is attributable to the additional efforts related to the installation of the third-stage telemetry package by NASA and DAC.)

On June 27, the combined spacecraft and modified third-stage assembly was mated with the second stage of the launch vehicle, and an on-stand optical check was started. However, RF interference (attributed to reflected beacon signals) was noted on the command receivers, and dummy loads had to be installed in place of three of the four elements of the spacecraft's transmitting antenna in order to enable effective programming. With the dummy loads in place, the optical check and a Task 4 check were performed with good results. Thereafter, a special series of tests were conducted which showed the power outputs from the transmitting antennas, the command-receiver sensitivity, and command-receiver bandwidth all to be normal.

The "F-1 Day" effort was started on June 28. The fairing was installed and the gantry was removed. However, the interference observed on June 27 was even more pronounced with the fairing installed, and the spacecraft could not be commanded. Accordingly, the gantry was returned and the fairing removed. After testing showed that command-receiver AGC voltage could be varied by touching the spacecraft's transmitting antennas and various parts of the third-stage rocket and telemetry package, a special series of tests were conducted with the spacecraft separated from the third-stage rocket. These tests showed that, with the spacecraft raised 6 inches above the third stage, the spacecraft could be commanded without difficulty. Upon completion of these tests, the spacecraft was returned to Hanger AE and another series of tests was conducted which provided a further demonstration of the effect produced by the proximity of metal (reflective) surfaces on the command-receiver AGC voltages.

On June 29, at NASA's request, the spacecraft hat was removed and a notch filter, similar to those used on TIROS VIII and IX, was installed in the coaxial cabling used to connect the receiving antenna to the command receivers. Testing of this filter, which was a spare unit for spacecraft OT-3 and had been installed on the RF model, showed that it attenuated 136.65-Mc signals by 26.5 db, while attenuating signals at the command frequency by only 1 db. After the filter installation and checkout was completed, the spacecraft hat was reinstalled on the baseplate and the command sensitivity was checked with good results. Thereafter, a complete Task 4 check was performed with the spacecraft mounted on a metal workstand; satisfactory results were achieved. Then the spacecraft was transported to the launch pad and remated to the third-stage. A complete Task 4 check was then performed; all results were good except the operation of clock No. 1, which did not alarm on time. After this, the fairing was installed and a second Task 4 check was started. Normal command was achieved until the gantry was removed; then the spacecraft could no longer be commanded. When the gantry was returned, normal commanding was again achieved; however, it was decided to discontinue the launch effort until the following day.

On June 30, the possibility of modifying the fairing to reduce the interference problem was discussed but rejected, and "F-1 Day" testing was started at 11:00 AM EST. A Test 4 check was performed with good results, except that both clocks alarmed a few minutes late. (This anomaly was attributed to limitations in the modulator of the NASA/Goddard Launch Operations transmitter that was used in this test, which precluded good-quality clockset pulses.)

The "F-O Day" checks were started on July 1 at 6:00 AM EST. Spacecraft checks were started at 11:30 AM, and a complete Task 4 check was performed with excellent results. Shortly thereafter, the fairing was reinstalled on the launch vehicle. At T-15 minutes, the spacecraft OT-1 was addressed and the "Direct Camera No. 1" command was used to command the spacecraft. In response, the beacon transmitters turned ON and four direct pictures were requested and the appropriate video signals received.

At 11:07 PM EST, on July 1, TIROS X was successfully launched from Cape Kennedy, and injected into a near-polar, sun-synchronous orbit.

APPENDIX

APPENDIX

SOLAR-CELL ARRAY POWER PREDICTION

I. INTRODUCTION

The power-prediction procedure employed for the spacecraft OT-1 solar array was based on:

(1) the development of secondary standard cells using a color filter wheel and primary standard cells,

(2) the performance of array measurements in sunlight (in New Jersey), with linear extrapolation of the readings to outer space values, and

(3) the use of these measurements and extrapolations, plus other inputs, with a computer program to develop a power prediction.

A color filter wheel calibrated by means of four primary standard cells was used to establish six secondary standard cells. The six secondary standards were mounted on a panel and positioned normal to the sun. A digital voltmeter was placed across a 1-ohm shunt and used with a switching arrangement to obtain readings of the I_{SC} of each standard cell.

II. ARRAY MEASUREMENTS

The solar-cell array was placed on an adjustable frame so that the top-hat array and each side panel could, in turn, be positioned normal to the sun. The area under test was exposed to the sun and all other units were shaded. Immediately prior to the test, two different temperature readings of the top-hat array were taken, and the measured values were 25°C and 27°C.

A complete I-V curve was drawn for each unit and the associated isolation diodes; and the curve for the top-hat, as shown in Figure A-1, was taken as representative of the

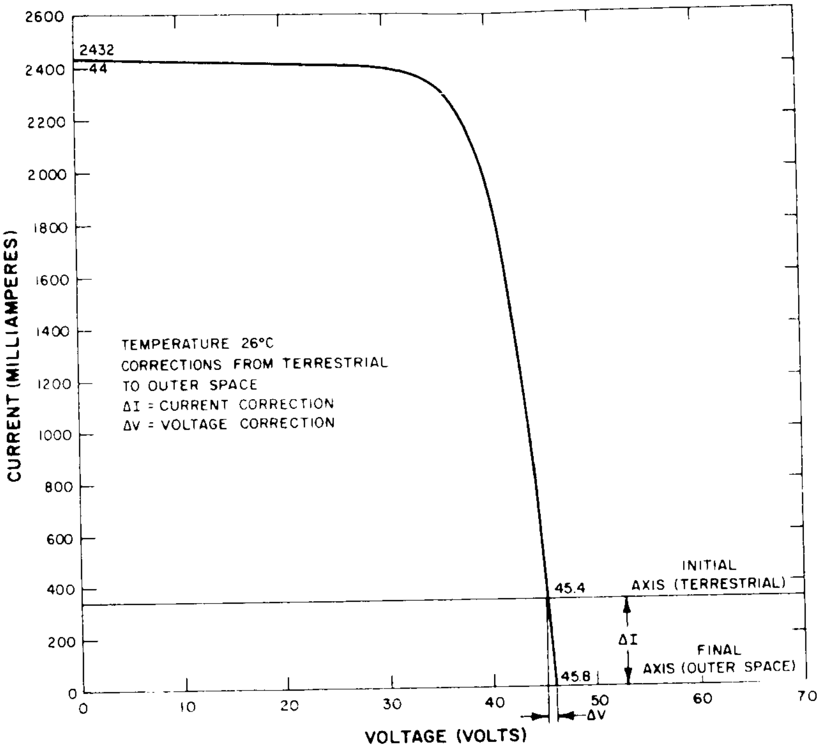


Figure A-1. I-V Curve for Spacecraft OT-1 Top-Hat Solar-Cell Array

entire array. The values obtained in the array measurements were linearly extrapolated to outer-space conditions by means of the following equations:

$$I_{sco} = I_{scm} F_x, \text{ and}$$

$$F_x = \frac{I_{scos}}{I_{scms}},$$

where

I_{sco} is the short-circuit current of the solar array extrapolated to outer space,

I_{scm} is the short-circuit current of the solar array as measured for the earth,

F_x is the extrapolation factor,

I_{scos} is the outer-space value for the short-circuit current of a secondary standard cell, as calibrated by means of the color filter wheel measurements, and

I_{scms} is the short-circuit current of the secondary standard cell as measured for the earth.

F_x for the top-hat array was determined to be 1.161; the average F_x for the side panels was determined to be 1.164. Using these values, a weighted average of the I_{sco} values per solar-cell string was obtained. This value was 54.9 ma.

The V_{oc} for each unit of the array was measured and corrected for outer-space conditions by a correction factor which accounted for the light-intensity difference between terrestrial and outer space conditions. This correction factor was obtained using the following relationship:

$$V_{oc} = K \ ln \frac{I_{sc}}{I_o},$$

where

K is the constant of proportionality and

I_o is the saturation current.

Each of the six P-on-N standard cells were measured under two different light levels, where I_{sc1} and I_{sc2} and V_{oc1} and V_{oc2} are the respective short-circuit currents and open-circuit voltages at the two light levels.

From the above equation then,

$$V_{oc1} - V_{oc2} = K \ ln \frac{I_{sc1}}{I_{sc2}},$$

from which the value K was calculated.

Using $K = 2.94 \times 10^{-2}$ as the average value for the six secondary standard cells, a value ΔV_{oc} was obtained for the top-hat array and for each side panel, where I_{sc1}/I_{sc2} is F_x. For the top-hat array, ΔV_{oc} was determined to be 0.4 volt, and this correction is included in the I-V curve for the top-hat array as shown in Figure A-1.

The weighted average of the corrected V_{oc} values per string was determined to be 45.8 volts. The V_{oc} value per cell, i.e., 0.57 volt, was obtained by dividing the average V_{oc} for the string by the number of cells in the string (45.8/80).

The value for the V_{oc} per cell forms one intercept of the I-V curve for a single cell. The other intercept was obtained by dividing the I_{sco} value for the array by the number of strings (2432 ma/44) and was determined to be approximately 55 ma. The I-V curve for the standard cell, therefore, is mathematically identical to that of the top-hat array, and is linearly proportioned to accommodate the average values for I_{sc} and V_{oc}.

III. USE OF A COMPUTER FOR POWER PREDICTION

A computer program was used to facilitate the power prediction, because the solar-cell array operating in space experiences a variety of gamma angles, each with associated effects on (1) the total cell area exposed to the sun at the particular moment and (2) the consequent temperature conditions. At each given gamma angle, the current contributions from panels at varied angles were computed and summed to yield a value for the total array current. The use of the computer technique provided an accurate and efficient means of performing this procedure.

The factors included as inputs for the computer program were as follows:

(1) the I-V curve of an average cell under outer-space conditions,

(2) gamma angle,

(3) temperature,

(4) the effect of the angle of incidence of the sun's rays on the current produced by the exposed panels, and

(5) the intensity of the sun's rays as affected by distance from the sun. (The computer was originally programmed for 1 astronomical unit; and, in anticipation of a mid-July launch date, a correction of 0.968 was applied.)

Table A-1 is a summary of the results obtained with the computer program. The data included in this summary is presented as current and voltage values at the equilibrium temperature corresponding to a specific sun angle.* Figure A-2 presents a family of curves derived from this data, indicating the current profile for various gamma-angle values.

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*The tolerance applied to the power-prediction values was an r.m.s. value of individual system errors.

TABLE A-1. CURRENT-VOLTAGE VALUES AT VARIOUS GAMMA ANGLES

GAMMA ANGLE (Degrees)		30	40	50	60	70	80	90	100	110	120
TEMPERATURE (°C)	Top	83	75	61	43	22	43	-	-	-	-
	Sides	19	22	23	23	27	27	27	30	30	27
VOLTAGE (Volts)		CURRENT (Amperes)									
24		2.515	2.454	2.298	2.060	1.733	1.369	1.125	1.103	1.057	0.965
26		2.451	2.417	2.285	2.048	1.722	1.363	1.120	1.098	1.052	0.961
28		2.303	2.333	2.263	2.035	1.711	1.357	1.116	1.093	1.047	0.956
30		1.952	2.134	2.205	2.020	1.698	1.349	1.110	1.087	1.041	0.950
32		1.355	1.695	2.073	1.994	1.683	1.339	1.103	1.078	1.032	0.943
34		1.090	1.385	1.763	1.924	1.662	1.323	1.092	1.058	1.013	0.933
36		0.492	0.658	1.181	1.755	1.609	1.284	1.058	1.008	0.964	0.902
38		0.455	0.608	0.861	1.336	1.462	1.188	0.968	0.875	0.835	0.820

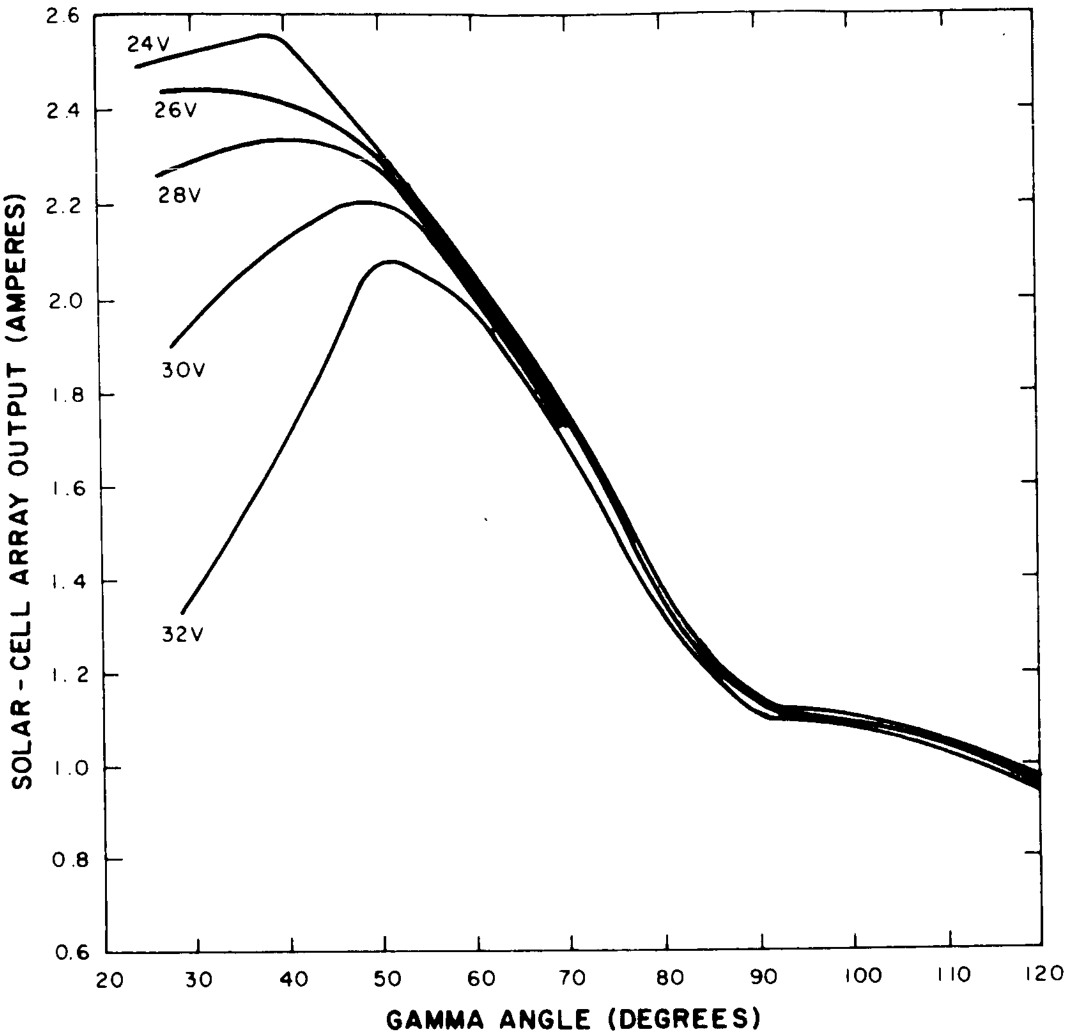
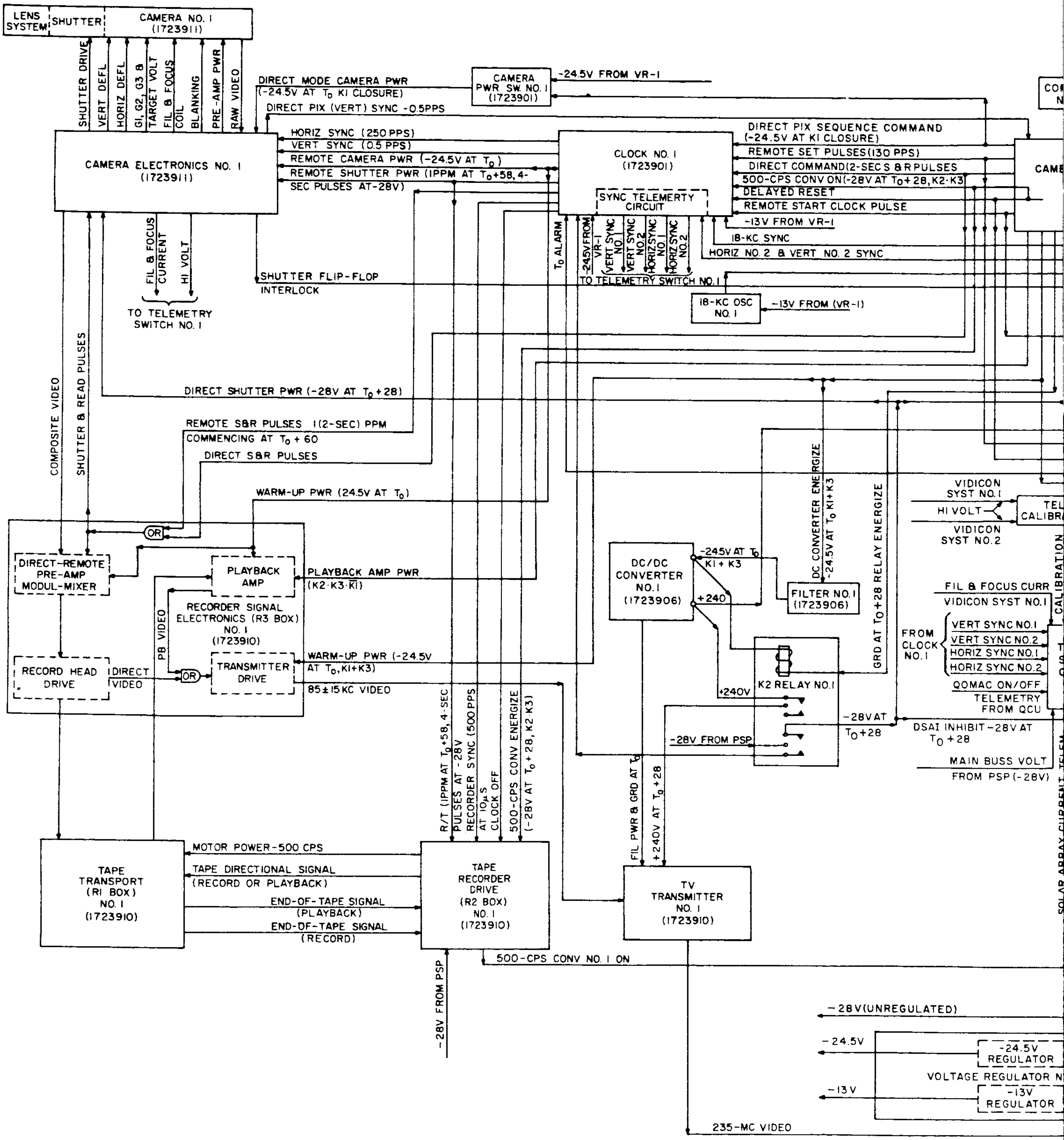
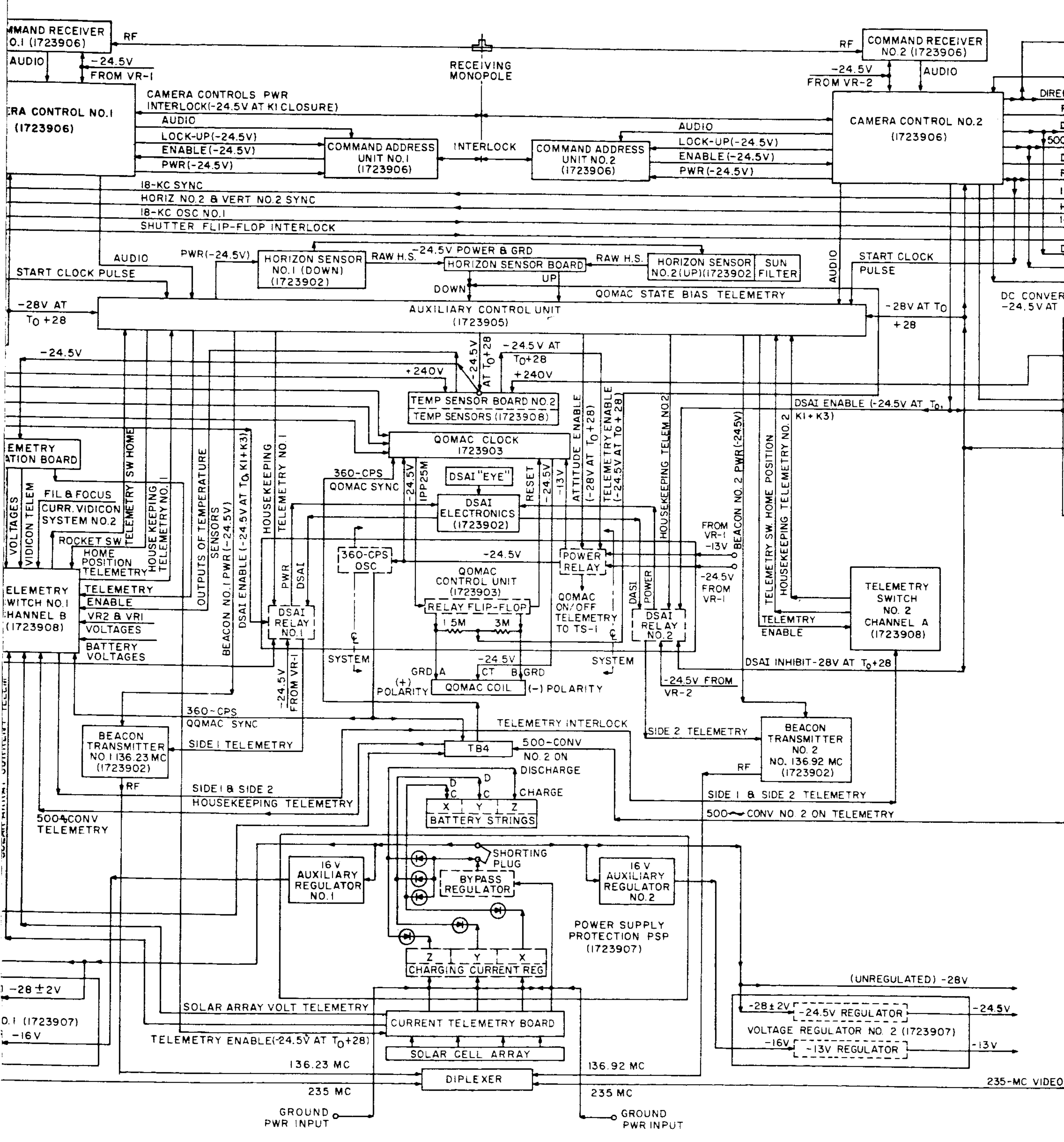


Figure A-2. Solar-Cell Array Output Versus Gamma Angle, for Spacecraft OT-1

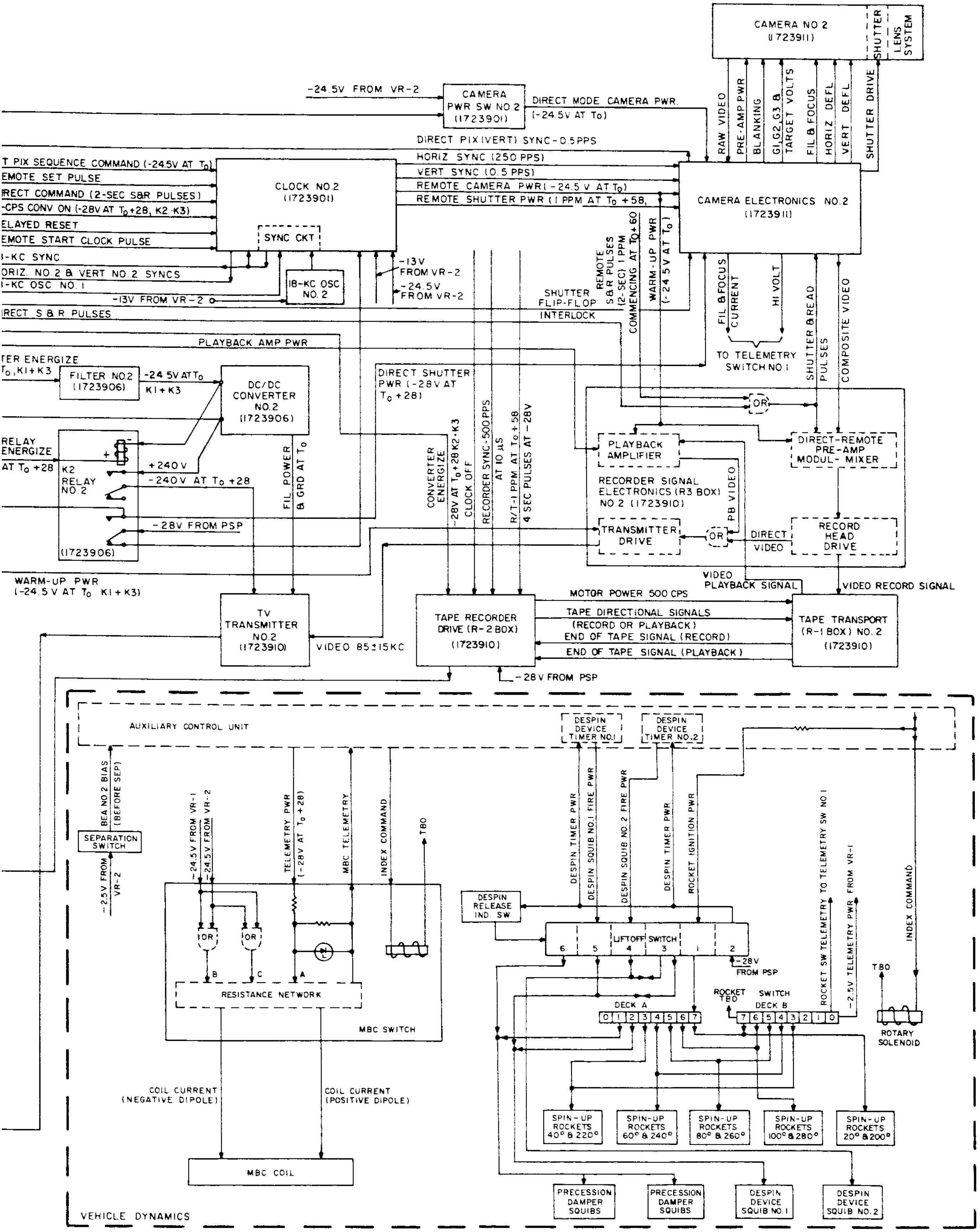


LENS SYSTEM
SHUTTER
CAMERA NO. 1 (1723911)
SHUTTER DRIVE
VERT DEFL
HORIZ DEFL
G1, G2, G3 & TARGET VOLT
FIL & FOCUS COIL
BLANKING
PRE-AMP PWR
RAW VIDEO
DIRECT MODE CAMERA PWR (-24.5V AT T0 K1 CLOSURE)
DIRECT PIX (VERT) SYNC -0.5PPS
CAMERA PWR SW. NO. 1 (1723901)
-24.5V FROM VR-1
CAMERA ELECTRONICS NO. 1 (1723911)
HORIZ SYNC (250 PPS)
VERT SYNC (0.5 PPS)
REMOTE CAMERA PWR (-24.5V AT T0)
REMOTE SHUTTER PWR (1PPM AT T0+58, 4-SEC PULSES AT-28V)
CLOCK NO. 1 (1723901)
SYNC TELEMERTY CIRCUIT
DIRECT PIX SEQUENCE COMMAND (-24.5V AT K1 CLOSURE)
REMOTE SET PULSES (130 PPS)
DIRECT COMMAND (2-SEC S & R PULSES
500-CPS CONV ON (-28V AT T0+28, K2·K3)
DELAYED RESET
REMOTE START CLOCK PULSE
-13V FROM VR-1
18-KC SYNC
HORIZ NO. 2 & VERT NO. 2 SYNC
T0 ALARM
-245V FROM VR-1
VERT SYNC NO. 1
VERT SYNC NO. 2
HORIZ SYNC NO. 1
HORIZ SYNC NO. 2
TO TELEMETRY SWITCH NO. 1
18-KC OSC NO. 1
-13V FROM (VR-1)
FIL & FOCUS CURRENT
HI VOLT
TO TELEMETRY SWITCH NO. 1
SHUTTER FLIP-FLOP INTERLOCK
COMPOSITE VIDEO
SHUTTER & READ PULSES
DIRECT SHUTTER PWR (-28V AT T0+28)
REMOTE S&R PULSES 1(2-SEC) PPM COMMENCING AT T0+60
DIRECT S&R PULSES
WARM-UP PWR (24.5V AT T0)
OR
DIRECT-REMOTE PRE-AMP MODUL-MIXER
PLAYBACK AMP
PLAYBACK AMP PWR (K2·K3·K1)
RECORDER SIGNAL ELECTRONICS (R3 BOX) NO. 1 (1723910)
PB VIDEO
RECORD HEAD DRIVE
DIRECT VIDEO
OR
TRANSMITTER DRIVE
WARM-UP PWR (-24.5V AT T0, K1+K3)
85±15KC VIDEO
DC/DC CONVERTER NO. 1 (1723906)
-245V AT T0 K1+K3
+240
FILTER NO. 1 (1723906)
DC CONVERTER ENERGIZE -24.5V AT T0 K1+K3
GRD AT T0+28 RELAY ENERGIZE
K2 RELAY NO. 1
+240V
-28V FROM PSP
-28V AT T0+28
VIDICON SYST NO. 1
HI VOLT
VIDICON SYST NO. 2
TEL CALIBR
CALIBRATION
FIL & FOCUS CURR VIDICON SYST NO. 1
FROM CLOCK NO. 1
VERT SYNC NO. 1
VERT SYNC NO. 2
HORIZ SYNC NO. 1
HORIZ SYNC NO. 2
QOMAC ON/OFF
TELEMETRY FROM QCU
DSAI INHIBIT-28V AT T0+28
MAIN BUSS VOLT FROM PSP (-28V)
SOLAR ARRAY CURRENT TELEM
R/T (1PPM AT T0+58, 4-SEC PULSES AT -28V
RECORDER SYNC (500 PPS) AT 10µS CLOCK OFF
500-CPS CONV ENERGIZE (-28V AT T0+28, K2·K3)
FIL PWR & GRD AT T0
+240V AT T0+28
TAPE TRANSPORT (R1 BOX) NO. 1 (1723910)
MOTOR POWER-500 CPS
TAPE DIRECTIONAL SIGNAL (RECORD OR PLAYBACK)
END-OF-TAPE SIGNAL (PLAYBACK)
END-OF-TAPE SIGNAL (RECORD)
TAPE RECORDER DRIVE (R2 BOX) NO. 1 (1723910)
TV TRANSMITTER NO. 1 (1723910)
500-CPS CONV NO. 1 ON
-28V FROM PSP
-28V (UNREGULATED)
-24.5V
-24.5V REGULATOR
VOLTAGE REGULATOR N
-13V
-13V REGULATOR
235-MC VIDEO
CAME

47(1)

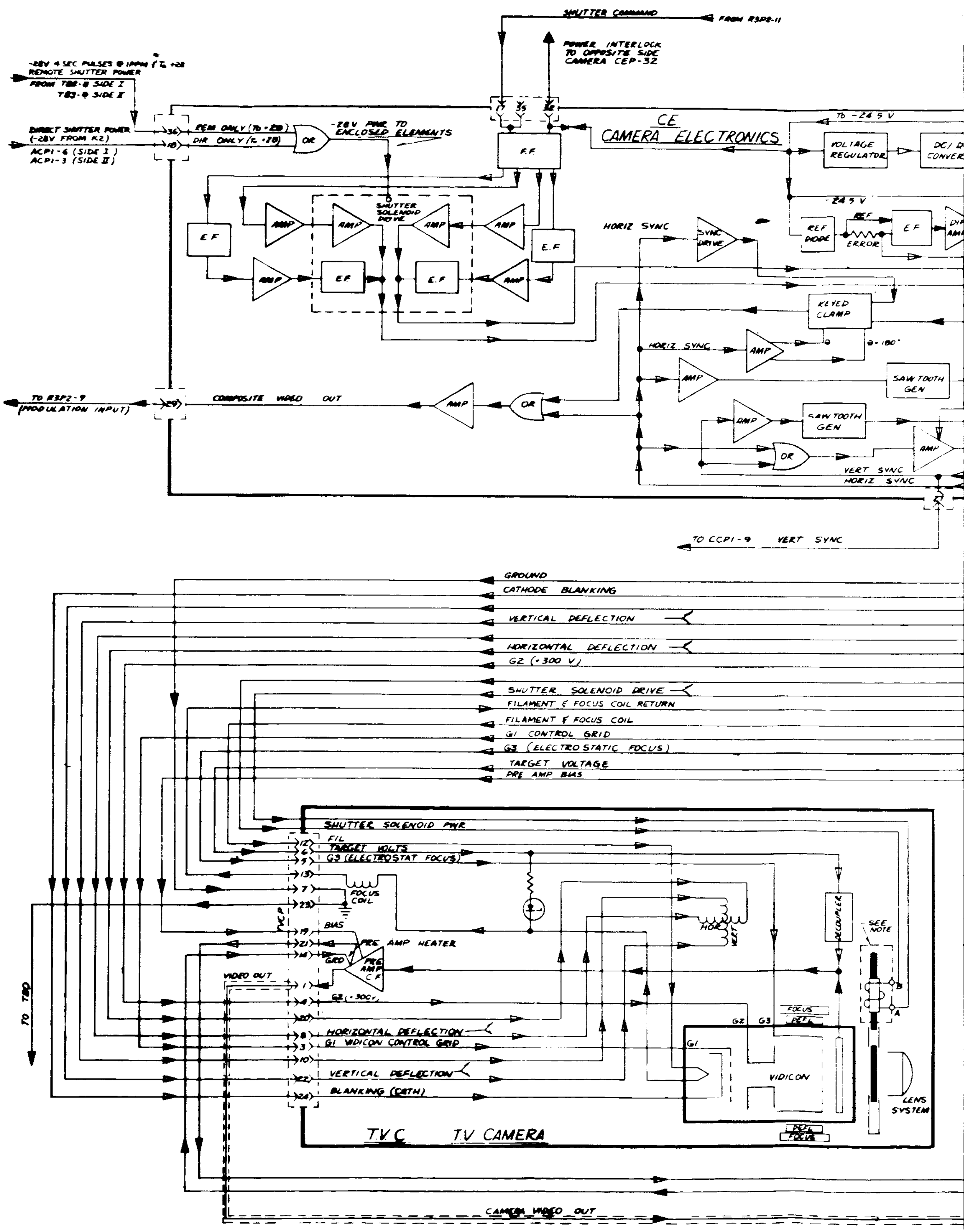

MMAND RECEIVER
O.I (1723906)
RF
AUDIO
-24.5V
FROM VR-I
RECEIVING
MONOPOLE
COMMAND RECEIVER
NO.2 (1723906)
-24.5V
FROM VR-2
AUDIO
ERA CONTROL NO.I
(1723906)
CAMERA CONTROLS PWR
INTERLOCK(-24.5V AT KI CLOSURE)
LOCK-UP(-24.5V)
ENABLE(-24.5V)
PWR(-24.5V)
COMMAND ADDRESS
UNIT NO.I
(1723906)
INTERLOCK
COMMAND ADDRESS
UNIT NO.2
(1723906)
CAMERA CONTROL NO.2
(1723906)
18-KC SYNC
HORIZ NO.2 & VERT NO.2 SYNC
18-KC OSC NO.I
SHUTTER FLIP-FLOP INTERLOCK
START CLOCK PULSE
HORIZON SENSOR
NO.I (DOWN)
(1723902)
RAW H.S.
-24.5V POWER & GRD
HORIZON SENSOR BOARD
UP
DOWN
HORIZON SENSOR
NO.2(UP)(1723902
SUN
FILTER
START CLOCK
PULSE
QOMAC STATE BIAS TELEMETRY
-28V AT
T_0 +28
AUXILIARY CONTROL UNIT
(1723905)
-28V AT T_0
+28
DC CONVER
-24.5V AT
+240V
TEMP SENSOR BOARD NO.2
TEMP SENSORS (1723908)
QOMAC CLOCK
1723903
DSAI ENABLE (-24.5V AT T_0, KI+K3)
EMETRY
ATION BOARD
360-CPS
QOMAC SYNC
DSAI "EYE"
DSAI
ELECTRONICS
(1723902)
RESET
ATTITUDE ENABLE
(-28V AT T_0+28)
TELEMETRY ENABLE
(-24.5V AT T_0+28)
HOUSEKEEPING TELEM NO.2
BEACON NO.2 PWR(-24.5V)
TELEMETRY SW. HOME POSITION
HOUSEKEEPING TELEMETRY NO.2
VOLTAGES
VIDICON TELEM
FIL & FOCUS
CURR. VIDICON
SYSTEM NO.2
ROCKET SW
HOME
POSITION
TELEMETRY
TELEMETRY SW HOME
HOUSE KEEPING TELEMETRY NO.1
OUTPUTS OF TEMPERATURE SENSORS
BEACON NO.I PWR (-24.5V)
DSAI ENABLE (-24.5V AT T_0, KI+K3)
HOUSEKEEPING TELEMETRY NO.I
360-CPS
OSC
POWER
RELAY
FROM VR-I -13V
-24.5V FROM VR-I
QOMAC
CONTROL UNIT
(1723903)
RELAY FLIP-FLOP
1.5M
3M
QOMAC
ON/OFF
TELEMETRY
TO TS-I
ELEMETRY
WITCH NO.I
HANNEL B
(1723908)
TELEMETRY
ENABLE
VR2 & VRI
VOLTAGES
BATTERY
VOLTAGES
DSAI
RELAY
NO.I
DSAI
RELAY
NO.2
TELEMETRY
SWITCH
NO. 2
CHANNEL A
(1723908)
TELEMTRY
ENABLE
SYSTEM
GRD
CT
GRD
(+)
POLARITY
QOMAC COIL
(-) POLARITY
DSAI INHIBIT-28V AT T_0+28
-24.5V FROM
VR-2
-24.5V
FROM VR-I
360-CPS
QQMAC SYNC
TELEMETRY INTERLOCK
BEACON
TRANSMITTER
NO.I 136.23 MC
(1723902)
SIDE I TELEMETRY
TB4
500-CONV
NO. 2 ON
DISCHARGE
CHARGE
SIDE 2 TELEMETRY
BEACON
TRANSMITTER
NO. 2
NO. 136.92 MC
(1723902)
SIDE I & SIDE 2
HOUSEKEEPING TELEMETRY
BATTERY STRINGS
SIDE I & SIDE 2 TELEMETRY
500~CONV NO. 2 ON TELEMETRY
500~CONV
TELEMETRY
SHORTING
PLUG
16 V
AUXILIARY
REGULATOR
NO. I
BYPASS
REGULATOR
16 V
AUXILIARY
REGULATOR
NO. 2
POWER SUPPLY
PROTECTION PSP
(1723907)
CHARGING CURRENT REG
(UNREGULATED) -28V
-28 ± 2V
SOLAR ARRAY VOLT TELEMETRY
CURRENT TELEMETRY BOARD
-24.5V REGULATOR
-24.5V
VOLTAGE REGULATOR NO. 2 (1723907)
-16V
-13V REGULATOR
-13V
TELEMETRY ENABLE(-24.5V AT T_0+28)
SOLAR CELL ARRAY
136.23 MC
136.92 MC
DIPLEXER
235 MC
235-MC VIDEO
GROUND
PWR INPUT

A 73



A 7 3

Figure II-1. Logic Diagram of Spacecraft OT-1



NOTE
SHUTTER SOLENOID IS MOUNTED EXTERNAL TO CAMERA HEAD.

A8D

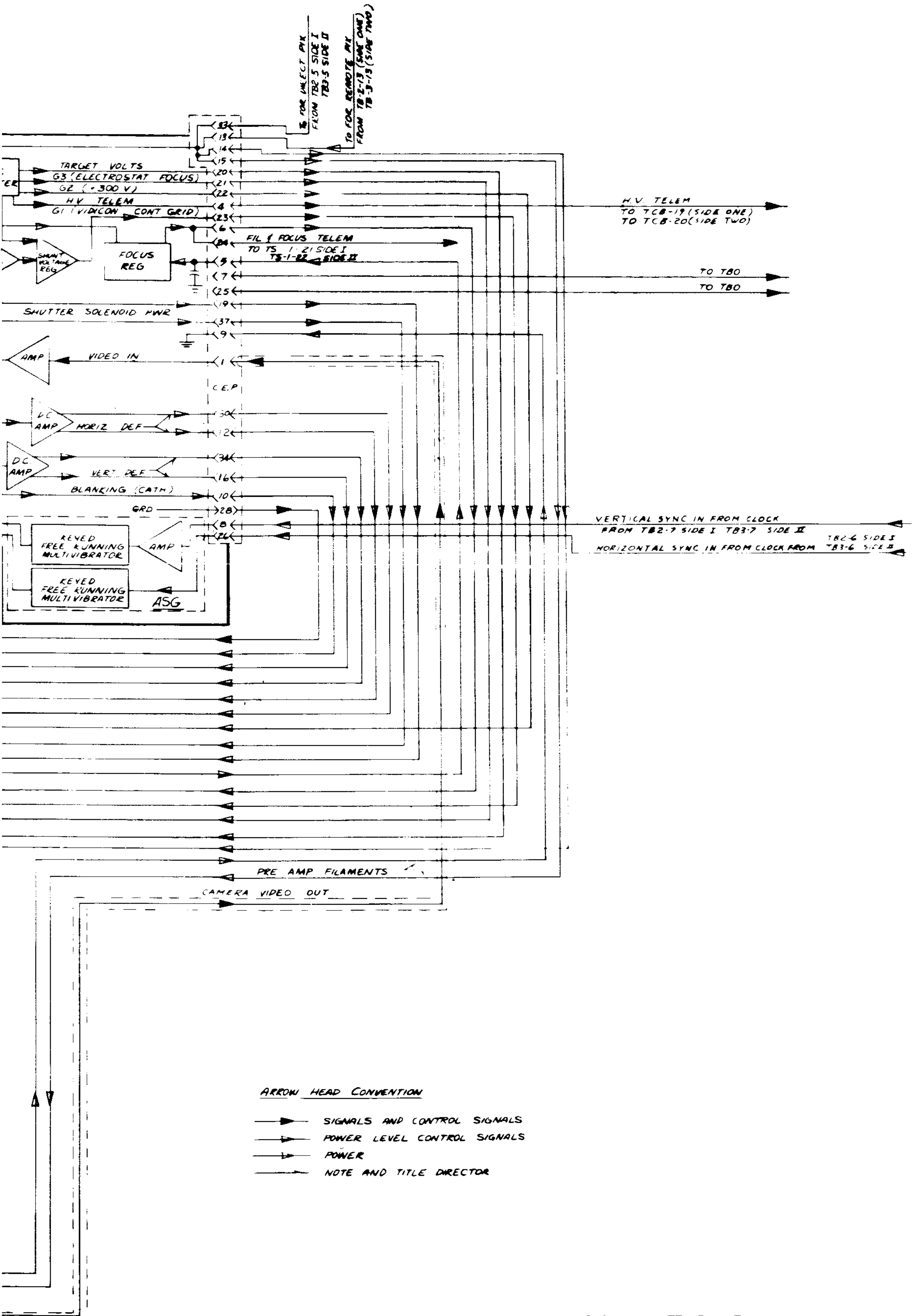
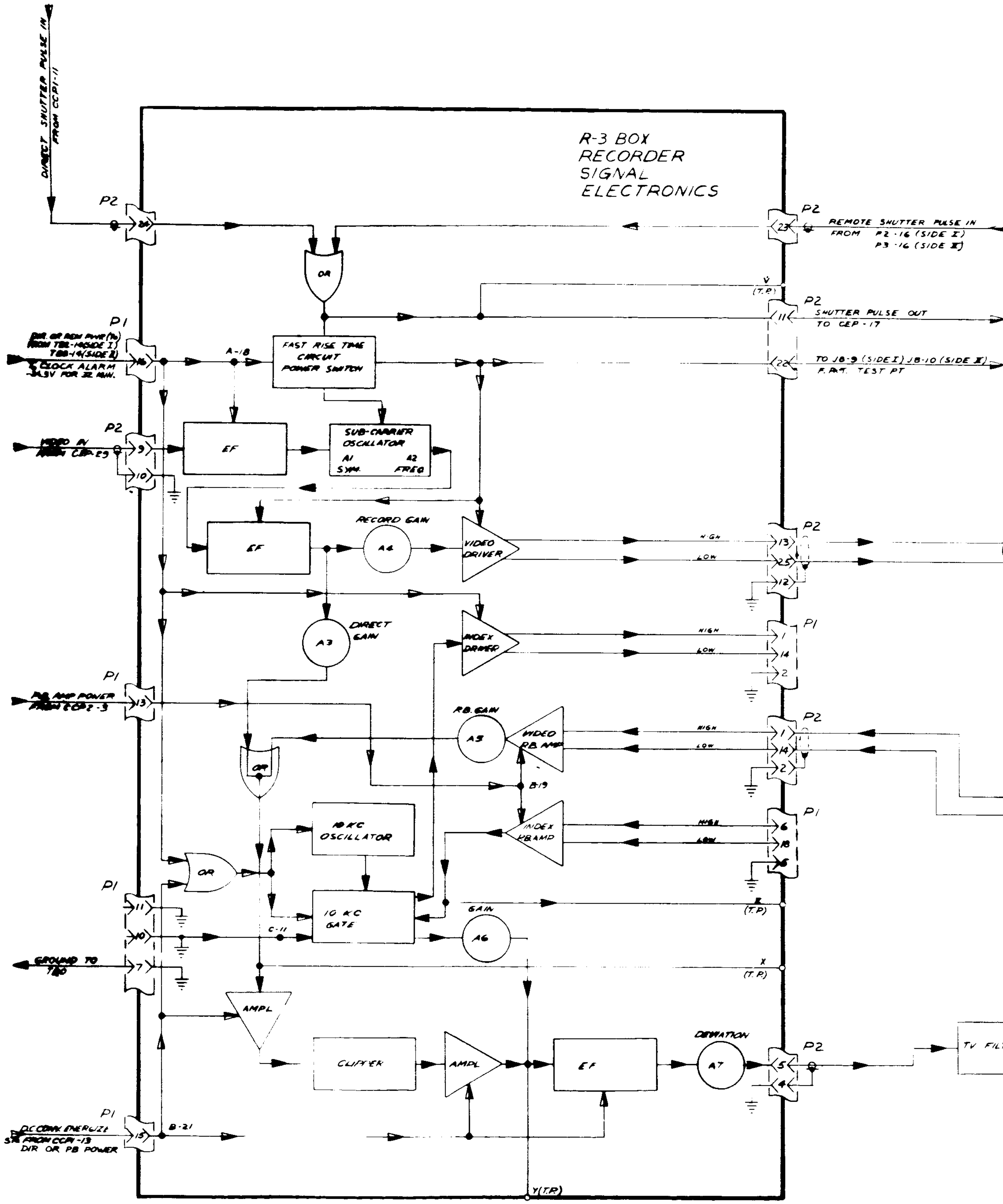


Figure II-3. Logic Diagram of TV Camera and Camera Electronics

A-8 (2)



NOTE
1. ALL RELAYS ARE SHOWN IN THE DEENERGIZED POSITION.

ARROW HEAD CONVENTION
SIGNALS & CONTROL SIGNALS
POWER LEVEL CONTROL SIGNALS
POWER
NOTE & TITLE DIRECTOR

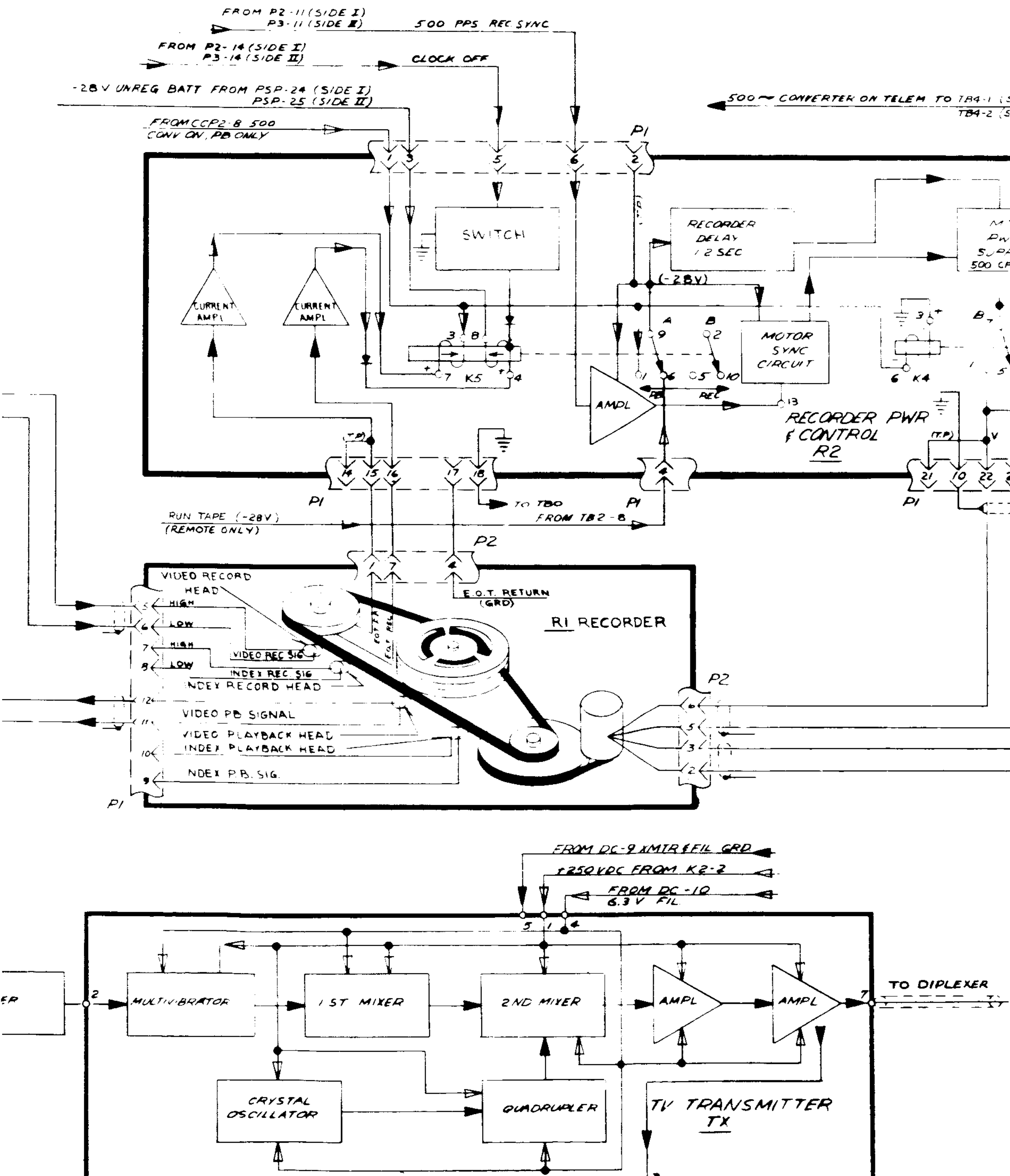


Figure II-4. Logic Diagram of
TV Tape Recorder

A-9